SECOND NATIONAL CONFERENCE on... SPACE MAINTENANCE and... EXTRAVEHICULAR ACTIVITIES

6, 7, 8 august 1968
stardust hotel
las vegas, nevada

sponsored by:

U.S. Air Force Aero Propulsion Laboratory and the National Aeronautics and Space Administration in cooperation with the Space Division of North American Rockwell Corporation

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6, 7, 8 August 1968, Las Vegas, Nevada

February 1969

FOREWORD

The Space Maintenance and Extravehicular Activities Conference serves as a forum in which leading authorities in this area of technology advance new ideas and techniques for critical discussion. The objectives of the conference are (1) to present current research and development contributions in the fields of Space Maintenance and Extravehicular Activities and (2), through the exchange and evaluation of the most advanced concepts, to stimulate further advances of this technology. By publishing the transactions of this conference, the Air Force Aero Propulsion Laboratory hopes to further promote these objectives.

The papers were submitted in reproducible form and were printed as submitted. This accounts for certain variations in treatment and style.

H. A. Lyon

HERBERT A. LYON
Colonel, USAF
Director,
Air Force Aero
Propulsion Laboratory

ABSTRACT

This report contains a presentation of technical contributions summarizing the status of current and significant research in the fields of Space Maintenance and Extravehicular Activities. The report is based upon the discussions at the Second National Conference on Space Maintenance and Extravehicular Activities held August 6-8, 1968 at the Stardust Hotel, Las Vegas, Nevada. The conference transactions have been arranged in the order of presentation during the seven sessions of the conferrnce.

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ADDITIONAL PAPERS

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ACKNOWLEDGMENTS

The Conference Committee would like particularly to acknowledge the efforts of the following Air Force personnel who have made major contributions in planning and arranging the program requirements:

A. J. Cannon	T. Cherwin
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Maj. N. Fairbanks	R. Speelman
Capt. J. Catiller	G. Medisch
F. Forbes	P. Danelak
P. Van Schaik	J. Davis
V. Stephens	B. DeWinter

In addition, thanks go to Captain Gary B. Reid and Lt. Carlan Silha for setting up and carrying through the Underwater Neutral Buoyancy Demonstration on Airlock/Astronaut Capability.

We would like to convey our appreciation to the Space Division of North American Rockwell Corporation for supporting sponsorship of a successful symposium.

A special acknowledgment should be noted for the services rendered by Mr. John Wright of the Nuclear Rocket Test Station for this special efforts in arranging and conducting the tour of the Jackass Flats facilities by the conference attendees 9 August 1968.

KEYNOTE ADDRESS

SPACE BEYOND THE THRESHOLD

By

LT. GEN. J. R. HOLZAPPLE
USAF

Chief DCS R&D
Pentagon

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SPACE BEYOND THE THRESHOLD

Lt. Gen. J. R. Holzapple
USAF
Chief DCS R&D
Pentagon

It is, indeed, a great pleasure to keynote the Second National Conference on Space Maintenance and Extravehicular Activities. This conference can serve the continual needs for fresh perspectives and for cross-stimulation so basic to scientific and technical progress. And there can be no question that the conference co-sponsors - the National Aeronautics and Space Administration and the Air Force Aero Propulsion Laboratory - have developed a program that will well serve those needs. These are bound to be stimulating and productive sessions.

I think all of us tend to get wrapped up in the hurly-burly of everyday problems. So, from time to time, we simply reach the point where we need briefly to stand aside and take a fresh look, and do some fresh thinking. Could there possibly be a better place for uninterrupted and undistracted thought than here in the middle of the desert - with all of these privations?

I suppose that, over the next several days, someone will inevitably observe that mankind is really only at the threshold of space. As the keynote speaker, I might be tempted to be the first to make this observation. The trouble is, I'm not at all sure it's true.

I'm not sure it's true because I'm not sure what it means. If what we really have in mind is the state of our space technologies, then I think that we are already well beyond the threshold. But if we have in mind attitudes toward space, then I think it is entirely possible that we are just barely approaching the threshold.

On the surface, this may appear to be an abstract distinction. And yet it bears vitally on a concern we must all share this morning. Our concern cannot simply be with current technical problems or even with present funding problems. Our inevitable concern must always be with the direction of the national space program over the longer view. And I frankly think we cannot long proceed intelligently in space if attitudes lag technologies. So now is the time to take a hard look at some key attitudes toward space.

Thus far, the national space program has been propelled largely by the momentum of that mass of technology set in motion by the urgent needs for the ballistic missile, and has been speeded by such stimuli as Sputnik. These have been the exciting early years, rich in the drama of suspense and adventure. These have been the threshold years of space technology - the glamor years of public enthusiasm and public funding.

In the short span of ten years, this momentum has carried us from those first tentative unmanned space launches to the point where we are already deeply concerned about the direction of our space program beyond the first manned lunar landing.

This is how far the initial momentum has carried us. And, of course, we've learned much. But this kind of momentum could not forever continue. We have now well passed that phase in the space program - common to any wholly new scientific endeavor - where technical possibility is its own best justification.

It is, of course, still true for space, as for most areas of technical challenge, that many of the more exciting technical possibilities for the future will stem from insights and discoveries we cannot now even imagine.

But it is true, also, that the scientific search for new possibilities can only be sustained for a limited time if there is not at least an equally determined search for beneficial applications. And the search for applications demands that you lead your technologies, as opposed simply to following them.

The idea of funneling scientific and technical effort into specific applications is not always greeted with marked enthusiasm. In fact, the idea runs counter to a strongly held attitude, especially where space is concerned. This is the feeling that you run the risk of missing unsuspected possibilities if you concentrate heavily on those technologies that already have well defined applications.

Thus, there is a tendency to overlook the fact that the process of leading technologies toward specific applications may, itself, have the effect of opening up new possibilities. That is, strong direction need not necessarily inhibit scientific and technical innovation. And, of course, this attitude also tends to forget the fact that our scientific and technical programs in space are not being conducted for their own sake. The ultimate function is to solve problems, or to meet urgent needs, or to open new vistas for human benefit. This is precisely the function of such programs as Apollo Applications and the Manned Orbiting Laboratory. With these programs we are moving somewhat away from basic experimentation and more in the direction of uses and applications in space.

And we can all welcome this as a very healthy trend, it is entirely probable that some of the funding problems for some of our space programs will be bridged when the applications are more fully recognized and understood. Very much contrary to another popular attitude - that Vietnam is the source of most of our funding problems - I think the more significant source is the need for a better understanding of possible applications in space, and their true benefits.

I might add, parenthetically, that almost everyone in government who is confronted with funding problems - and there are few who aren't is likely to point to Vietnam as the source of his difficulty. Well, there is no denying that Vietnam is having some impact. But I suspect that anyone who is expecting a sudden flow of funds into his particular program the moment the war ends, is liable to be disappointed. In any event, this is almost certain to be true where space is concerned.

Where funding is involved, there can be no doubt that the space program is well beyond the threshold. It's not really that the sense of urgency has gone out of the space program, or even that it has lost much of its romance. The problem - if it can properly by called a problem - is that our nation is tackling a wide range of urgent and demanding challenges. What has happened is that the space program has simply reached a new maturity and is finding its proper place in the perspective of <u>all of our national challenges</u>. And this should not be alarming. The fact that the space program will have to compete even more vigorously with other demands on our national resources may, in the long view, greatly benefit it. This kind of discipline forces even more advanced and creative thinking.

In relation to the somewhat more competitive environment in which the space program will likely function for funding, there seems to be some renewed concern that the economic factor will place too many space programs at an unfair disadvantage. The feeling may be that innovations in space are anyone's guess and that if they can't be predicted, then neither can their costs. Costs, of course, have to be based on the proven or tested. There is no way to predict costs on the more "high risk" technology programs where technical problems and system capabilities may be largely unknown. So the effect could be to reduce or eliminate risk taking and this may, in turn, close the door on truly dramatic but unrecognized possibilities.

There is no denying that this is a problem.

But I think the problem is easily exaggerated. Moreover, experience has shown that the place to explore risks is at the basic technology level. You surface your promising possibilities here, and this is where you invest your "risk capital" with a view toward eliminating the risks as fully as possible. You generally don't have to take severe risks by the time you are in a major hardware program.

As for space competing for funding on the basis of economic benefit, there is every reason why it should compete very successfully. There is ample evidence that space applications will shortly - if they have not already-provide economic returns that can match, or perhaps exceed by as much as two to three times, the annual national investment in space, both public and private. And this does not take into account the economic return to the nation of such factors as new space oriented industries. Interestingly, very few people were seriously thinking in terms of space providing an economic return until very recently.

Precise figures are still largely lacking. Nonetheless, the evidence is clear For example, a study conducted for the National Academy of Sciences pointed to the very high value of meteorological data from the Nimbus and Tiros satellites, and the newer ATS satellites of the Department of Commerce. These data have encouraged the development of mathematical models to simulate atmospheric systems on a global scale. The study indicates that weather and atmospheric data relayed instantly to computers programmed on mathematical models could make possible accurate weather forecasts for periods of up to ten days, and perhaps more. It is estimated that the value of such very long range forecasts to agriculture and the construction industry, alone, could be in the range of about $800 million a year. It is not difficult at all to visualize the dramatic value that could accrue in such areas, as perhaps in geological survey, aviation, and shipping. As a matter of fact, as a devoted fisherman, I can see some real possibilities there - or for the entire vacation and resort industries, for that matter.

Weather observation is just one significant economic benefit. Communications is another. There is no question that the cost of new satellite communication systems compares much more than favorably with the costs of new trans-oceanic undersea cables, or cross-continent underground and suspended cable systems. I understand that A.T.& T. is currently estimating that a combined, space based telephone-television system - just for the domestic needs of the United States would result in an investment savings of about $200 million by 1980.

So, any argument that the space program cannot compete on economic grounds is apt to prove woefully short-sighted. In fact, it seems to me that one of the strongest cases for support of the space program in the coming years is going to be the economic case. Like aviation starting in the late 1920s, space in the mid-1960s promises to open a vast new arena for economic growth. And this translates in terms of new jobs, new challenges, and new possibilities.

There is, of course, another aspect of the economics of space. Until rather reçently, all of our space programs advanced in a comparatively uninhibited cost environment. This is not to say that no one was thinking about costs, or worried about them, or attempting to hold them down. On the contrary, this has been a continuing and critical concern. The problem was that we had almost no technical alternatives and even less experience. If we wanted to get into space at all - if we wanted to develop a learning curve - then we had little choice other than to proceed the way we did. From the technical standpoint, the risks really weren't all that great. But we knew little about how to predict some of the costs.

Today we have a solid learning curve. We know a great deal more about how to predict costs, and we also know a number of ways we can cut or eliminate certain costs. Consequently, we are at the point where it is realistic to demand that every program be fully justified in terms of benefit in relation to cost. That is, in preparing proposals and in advocating systems for space, we can address the subject of costs with far greater confidence.

And this is to our advantage.

Moreover, NASA and the Air Force can now move even more in the direction of commonality and of multi-purpose designs with the aim of reducing costs. Not that this is entirely new: The man-rating of the Air Force Titan II for use with the NASA Gemini is a case in point. And, similarly, Gemini is a vital element of the Air Force Manned Orbiting Laboratory (MOL).

But the trend will be even more pronounced. At the present time, in fact, NASA and the Department of Defense are jointly studying possibilities for joint use of manned and unmanned space systems. One possibility, perhaps, might be a multipurpose spacecraft suitable for support of the missions and programs both of NASA and the Air Force. Such dual mission systems if feasible and effective in terms of the needs both of NASA and the Air Force, would result in substantial savings.

Or, looking a bit further ahead, a very large payload space system might serve specialized missions with highly specialized instrumentation, a ferry run to place support systems in orbit for a future mission, and a resupply mission all on the same launch.

As you know, both NASA and the Air Force are especially interested in the re-use of recovered spacecraft. We've already had some success in this area on a small scale. You may know, for example, that a recovered Gemini has been used in testing a heat shield for the MOL program. But on a much larger scale, our preliminary studies indicate that there could be very substantial savings in the reuse of entire spacecraft - even though the initial costs may be higher.

We are really just getting into the potential economies of re-usable systems. We have been considering various propulsion modes for re-usable boost systems for several years. Much of this is vital conceptual and theoretical work.

And, of course, the whole matter of re-entry and landing has been of considerable interest. Current recovery techniques - involving direct impact either on water or on land - appear to involve too many shocks that cannot easily be eliminated. Moreover, air recovery by parachute appears to be limited by weight considerations, at least for the near future. So we are keenly interested in large payload boost systems and spacecraft that would have the capability of landing in much the same way as a conventional aircraft.

In line with this, NASA and the Air Force have been conducting joint lifting body tests, principally using the X-24A. The results have been encouraging. But there are still problems. Frankly, the idea of a night or weather approach at 250 knots in an X-24, which has no go-around capability, doesn't send me to the locker room for my flight suit. What we will want in the actual operational space system is the capacity for conventional landing - combined with the ability to go-around or remain in a holding pattern.

The message in all of this is that economies for space systems are not only possible but technically desirable. And those who still hold to the view that you can't do anything in space without virtually unlimited funds need to adjust their thinking. We can do a great deal in space very economically. And we are going to do just that.

We come, now, to one of the thorniest problems of all. This is the attitude that unmanned near-earth-orbit space systems may have a role, but that man in space has no role. I will not, this morning, recount the many familiar arguments in support of our manned space programs. This touches on a subject closest to you because you, better than most, know how crucial man is for space maintenance, and what some of his potential extravehicular roles are.

In view of the broad range of space possibilities and the limited degree to which they have yet been explored, the question we should logically ask is simply under what circumstances can man do a better job, how much better, and at what additional cost. This is a key question, and one we must answer in the crucial post manned lunar landing decade.

We cannot ignore man in space and still remain technically competitive in space. But this is not the only compelling reason to consider man's role in space.

There is the matter of national security. I think there is no nation in the world more intent than ours on wholly peaceful space uses. But this deep intent must not lead to the fallacy of ignoring one hard fact: There are potential military applications for space. This is not to say that we need, therefore deploy hostile space systems. It is simply to say that we should identify the potential threats - and be able to counter them if need be.

In this connection, at the present time, one of the most urgent needs is to quantify military man's role in space. And I think that more people need to understand that all of our military programs in space - manned or unmanned - have this basic objective: To make any hostile military applications entirely unattractive as a military expedient. More people need to think through the attitude that any military study of possible military applications in space will necessarily invite hostile military applications. The effect is much more likely to be precisely the contrary.

I have talked about some of the current attitudes toward space, and of my own views in their connection. There are, of course, other attitudes that I have not addressed at all. Some of them may be obvious.

I have not, for example, discussed the often expressed feeling that space is essentially the arena of the large corporation and of the large design team. This is a classic misconception. We look back, now, on such "little" problems as eye irritation, the crying need for handholds, and unexpected profuse perspiration during some of the earlier Gemini extravehicular activity missions. And we know that the answers to these kinds of problems are most likely to come from the smaller companies or from a single engineer or scientist.

So no one has an exclusive claim on space. Space is a truly national challenge. We have essentially mastered the technology of space access and have made significant steps toward applications in space.

But the real path through and beyond the threshold of space is not the path of technology itself, but of human daring - of our ability to look well ahead of current technologies, and of current applications, for truly dramatic advances and benefits.

The challenge of space cries out not just for huge, talented, and well managed design teams, as important as these are. It cries out even more for the imaginative and the creative who are not inhibited by the past or intimidated by the future, who are prepared to think unconventionally and positively, and, above all, who are prepared to help drive the space program toward a future we cannot now even foresee.

Thank you.

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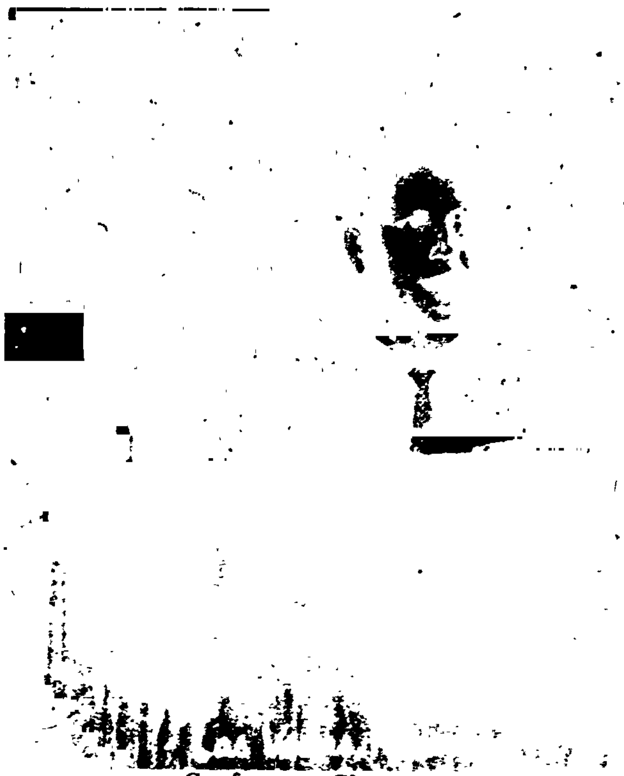

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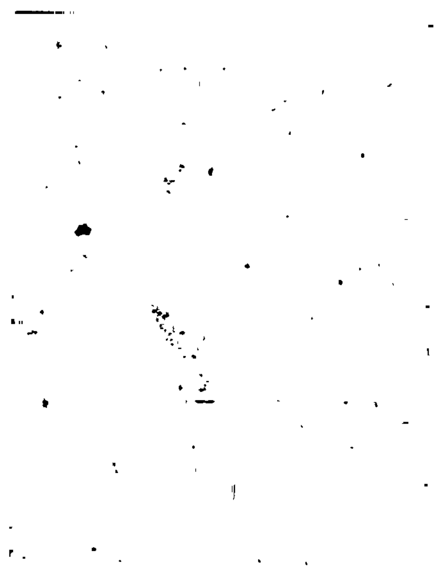

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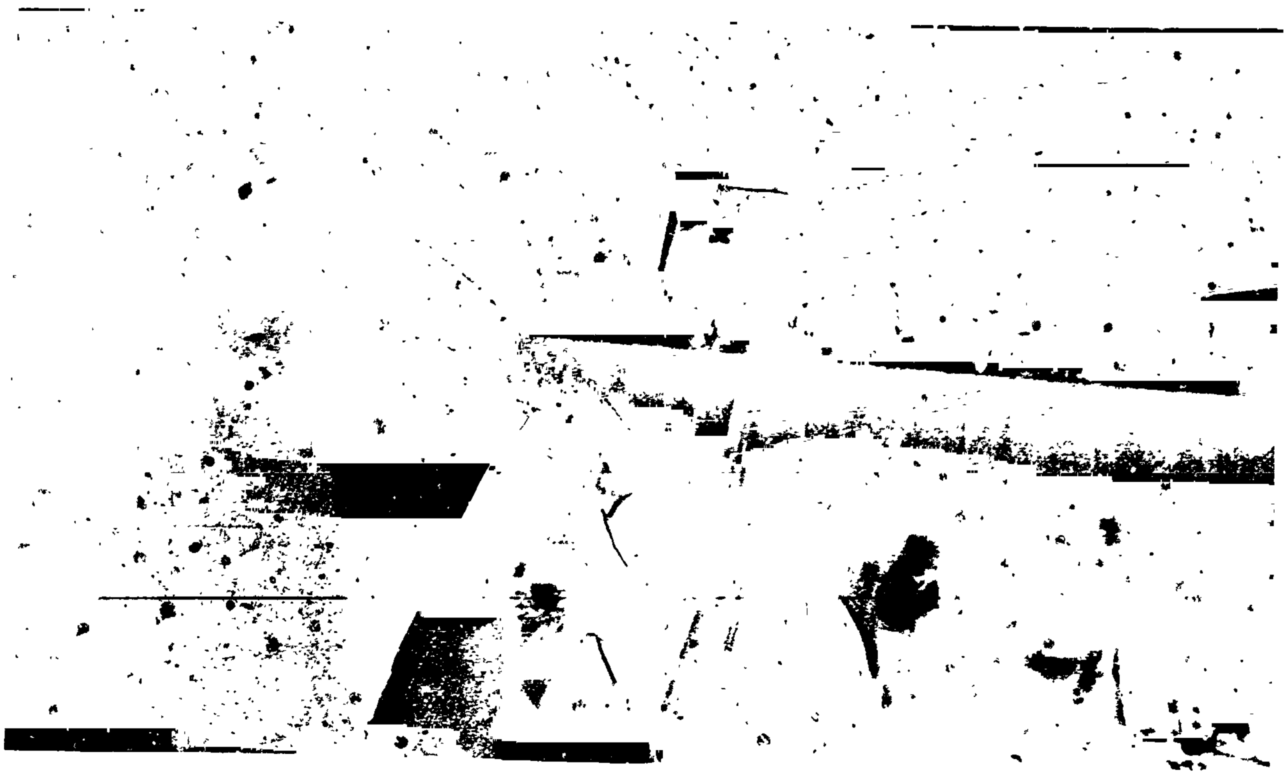
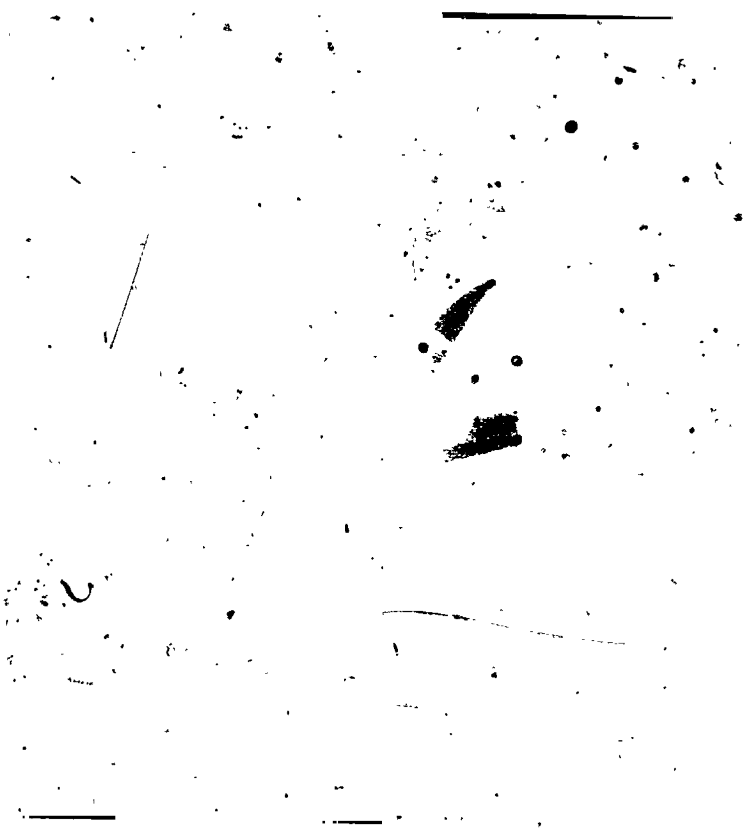


Luncheon



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Luncheon

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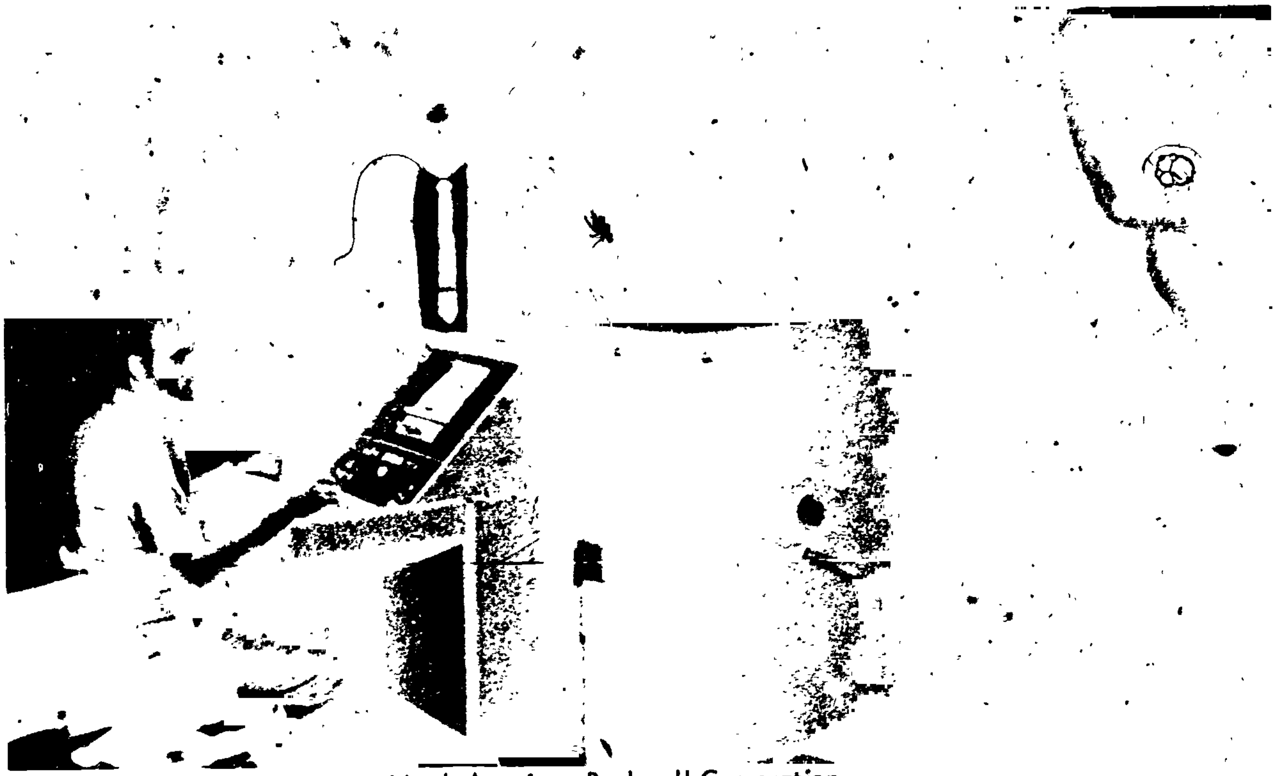
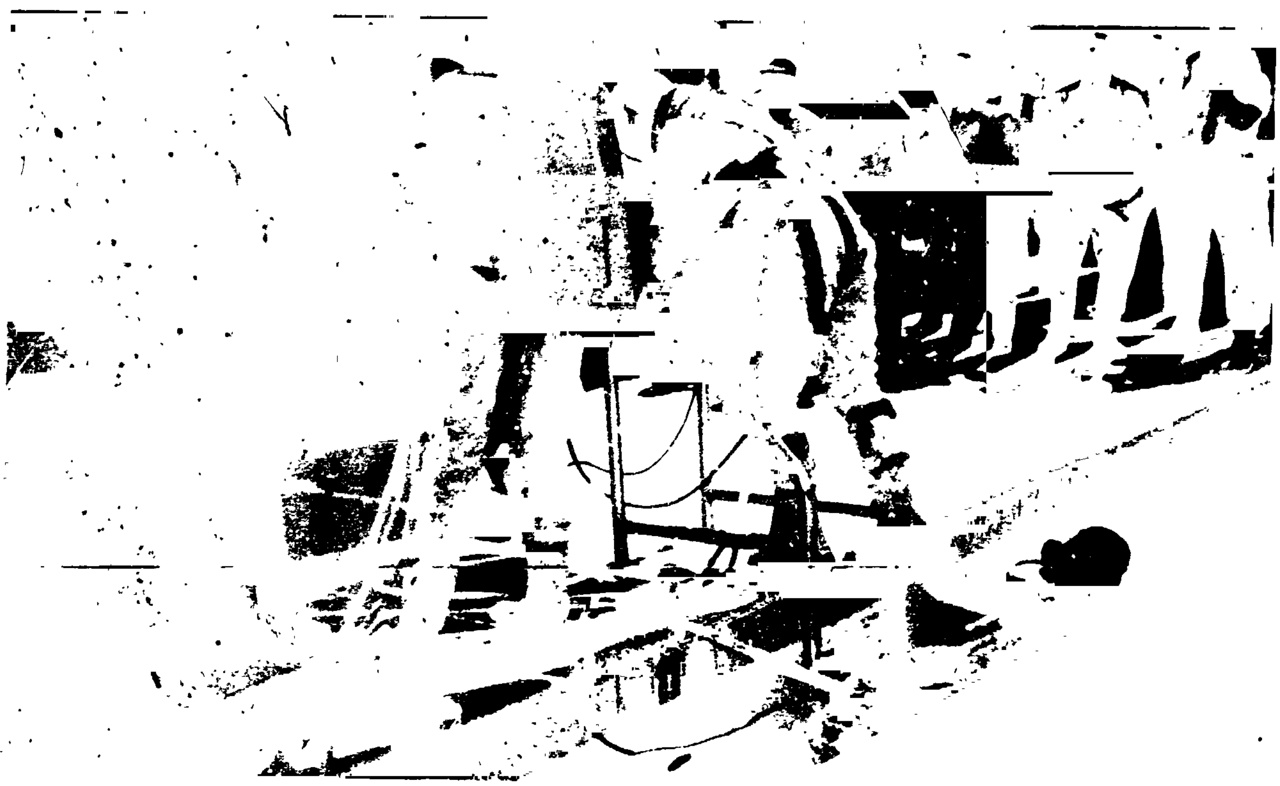
Air Force Aero Propulsion Laboratory

Air Force Exhibit



North American Rockwell Corporation
Exhibit

NASA Exhibit



Underwater Neutral Bouyancy Demonstration

General Electric Exhibit

SESSION I

SPACE MAINTENANCE AND EVA/SPACE MISSIONS

Session Chairman: Dr. Walton L. Jones
NASA Headquarters

SPACE MAINTENANCE AND EVA SPACE MISSIONS

INTRODUCTION

Walton L. Jones, M.D., Session Chairman
Stanley Deutsch, Ph. D., Co-Chairman
NASA Headquarters

Welcome to Session I on Space Maintenance and EVA Space Missions. Thanks to Bob and Larry, we have an extensive program aimed at understanding the capabilities of man to perform useful work outside the pressure vehicle in space and the development of equipment and tools to facilitate this effort. We are also interested in the development of life support systems and space suits required for astronaut EVA efforts as part of our larger effort in environmental control systems and protective equipment.

Our speakers this morning will cover mainline Apollo EVA, in-flight maintenance of space vehicles, and a look at several aspects of man's role in EVA. Dr. Stanley Deutsch, Chief of Man-Systems Integration in my program office, is co-chairman of this session with me. Dr. Deutsch is program manager for the NASA Extravehicular Technology Task Area.

Our first paper is titled "In-flight EVA for Mainline Apollo Missions." It is authored by Mr. Robert Bond and Mr. Jerry Goodman of the NASA Manned Spacecraft Center and Mr. Frank Parker of the General Electric Company.

Mr. Bond, who will present the paper, is an Aerospace Technologist in the Operations Integration Branch of the Apollo Spacecraft Program Office, Manned Spacecraft Center, NASA. He joined NASA in 1965, after a period as an officer in the Air Force. He received an M.S. in Psychology from the University of Mississippi.

Our second speaker is Mr. Donald Barnes. Mr. Barnes has 28 years experience in the aerospace field including 12 years experience with the Douglas Company. Mr. Barnes is responsible for analyzing and defining logistics requirements for earth orbital operations and the development of logistics systems and plans to meet these requirements. He is the Chief of the Space Systems Analysis Section.

Before joining Douglas in 1955, he served with the U.S. Air Force for 16 years in maintenance and operations assignments, the last 5 years of which were spent as a Navigator with the Strategic Air Command. Mr. Barnes will present a paper on In-flight Maintenance of Space Stations and Spacecraft.

Our last speaker this morning is Mr. Peter Van Schaik, who was until recently Technical Area Manager for Space Maintenance and Maneuvering, Air Force Aero Propulsion Laboratory. Mr. Van Schaik is now with the Advanced Planning Office of the same Laboratory.

Mr. Van Schaik graduated from General Motors Institute of Technology in 1953. He entered the research and development on EVA Techniques and Equipment in 1960. He has been responsible for programs on Astronaut Maneuvering Unit, Experiment D-12 on Gemini, the Space Power Tool, which was Experiment D-16, and various other EVA programs in the Air Force. Mr. Van Schaik will discuss man's changing role in EVA Space.

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IN-FLIGHT EVA FOR MAINLINE APOLLO MISSIONS

Robert L. Bond
Jerry R. Goodman

Apollo Spacecraft Program Office
Manned Spacecraft Center
National Aeronautics and Space Administration
Houston, Texas

and

Frank W. Parker

Apollo Systems Department
General Electric Company
Houston, Texas

SUMMARY: The objectives of the paper are to define the requirements for Apollo in-flight EVA, trace the development of an equipment subsystem that would support contingency EV transfer from the LM to the CM, and outline the present plan for a demonstration of the system during space flight.

INTRODUCTION

The objective of the mainline Apollo program is to land a two-man crew on the lunar surface and return them safely to earth. Several approaches were available concerning crew composition and vehicle performance. The method adopted by the NASA will see a three-man crew leave the earth aboard a CSM (command and service module) lofted into earth orbit by a Saturn V booster. In addition to the CSM, the final stage of the Saturn V will contain a LM (lunar module) which will be used for the descent to and ascent from the lunar surface (figure 1). Following a propulsion maneuver by the third stage of the Saturn V which will place the combination of vehicles on a translunar trajectory, the crew will dock the CSM with the LM, and these two units will separate from the booster's final stage (figure 2). The crew will remain in the CM (command module) until the combined vehicles have attained lunar orbit. Two of the three crewmen will then transfer intravehicularly through the transfer tunnel from the CM to the LM (figure 3). The CM, with the one remaining crewman, will remain in lunar orbit while the LM, with its two-man crew, descends to the lunar surface. At the completion of the lunar surface stay the ascent stage of the LM will lift the crew into lunar orbit to rendezvous with the CSM. The two spacecraft will dock (figure 4) and the LM crew will again make an IVT (intravehicular transfer) into the CM; the LM will be jettisoned to remain in lunar orbit; and the CSM will return the Apollo crew to earth.

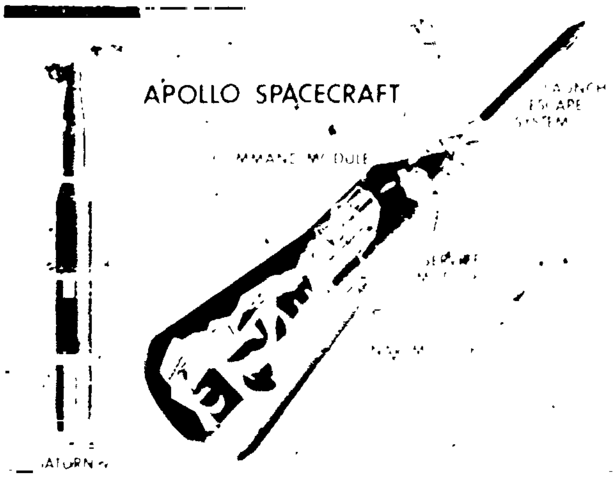

FIGURE 1

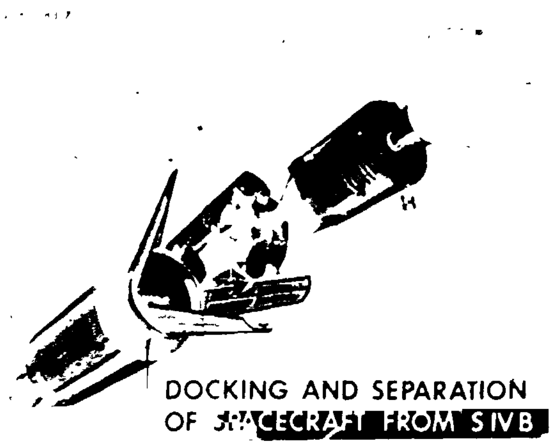

FIGURE 2

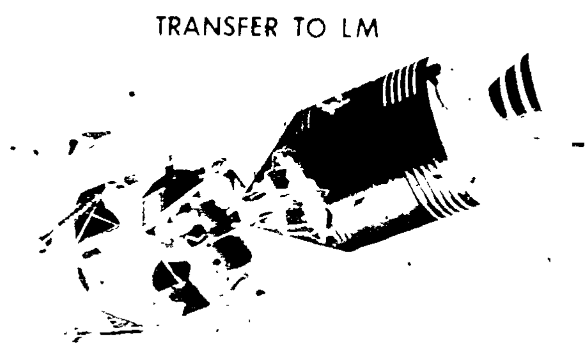

FIGURE 3

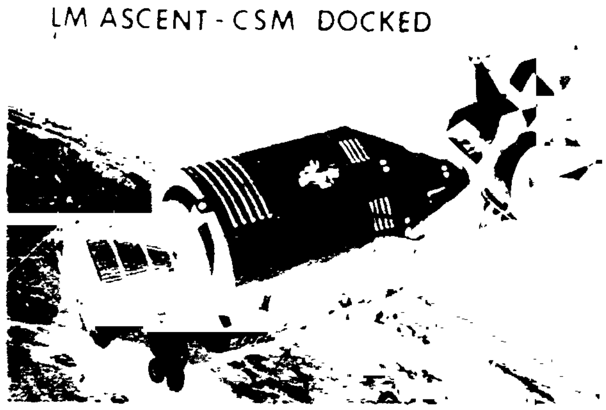

FIGURE 4

REQUIREMENTS FOR IN-FLIGHT EVA

The two-vehicle approach to the lunar landing mission has brought about a new set of manned spaceflight requirements associated with the transfer of crewmen between vehicles. Although extensive provisions have been made, both mechanically and procedurally, to insure the ease with which IVT can be performed, there are two classes of potential contingencies which lead to the need for a backup transfer system should the IVT not be possible. The only remaining mode in such cases is an extravehicular transfer (EVT).

The two classes of problems which could force the LM crewmen into the EVT mode are (1) those relating to the docking and the mechanical devices necessary to insure its success, and (2) those associated with vehicle control. In the former, the primary concern is the possible jamming of equipment in the transfer tunnel since the docking probe and drogue (figure 5) must be

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DOCKING PROBE AND DROGUE

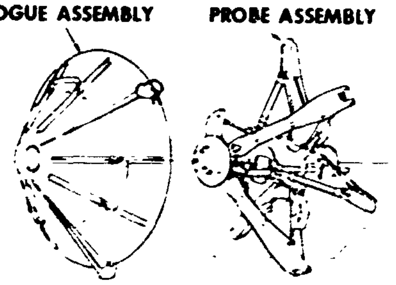

FIGURE 5

removed from the passage to clear the way for the crewmen. In the latter, there is the possibility that excessive consumption of fuel or control stability difficulties could render the LM a moving target rather than a stable docking platform. This family of potential difficulties, however, is considered much less likely to become an overt problem than those associated with the tunnel hardware manipulations.

Although the lunar landing mission has been stressed, the EVT requirement remains unchanged for any Apollo mission in which manned LM operations are to be performed. Since only the CM is designed for reentry into the earth's atmosphere, crew recovery into that vehicle is essential. Thus, the EVT capability had to be operational by the time of the first earth orbital mission in which manned LM operations were to be conducted with the LM separated from the CSM.

TECHNICAL APPROACH

The recognition of the need for an alternate transfer path between the LM and CM led to an early design of hardware to fulfill this need. At this stage of the Apollo hardware design cycle, however, NASA did not have the advantage of the experience and insight gained in the EVA environment during the later Gemini missions, and the initial Apollo transfer system, though seemingly feasible at the time, proved to be inadequate to meet the requirements.

During the EVA's conducted on the Gemini 4, 9, 10, 11, and 12 missions, various techniques and hardware provisions for manual extravehicular locomotion and restraint were evaluated. It was learned that the preferred transfer device was a rigid and continuous handrail along which the EVA crewman could translate. Experience also showed that for any appreciable task to be accomplished, local area restraints were essential.[1]

In view of the recognized requirements for the EVT provisions, and the indications from Gemini experience that the initial concepts provided for Apollo were inadequate, a team was formed to definitize the requirements and develop suitable hardware provisions to meet the requirements. The EVA team recognized that the equipment needed by the Apollo crewman to safely perform the extravehicular transfer from LM to CM would include:

(1) A system of installed translation aids for manual locomotion on both the LM and CSM;

(2) Sufficient artificial illumination of the transfer path to permit a night-time transfer;

(3) A personal safety line and local area restraints, to aid in body positioning where necessary during the transfer;

(4) A tool to permit opening of the CM side hatch from outside in

case the crewman inside the CM had not already done so; and

(5) A spacesuit and portable life support system capable of protecting the crewman from the thermal-vacuum environment during the period of transfer.

The design of provisions for (1) through (4) above was of immediate concern to the group, since it was believed that the spacesuit-life support system being developed for Apollo lunar surface EVA would adequately support the EVT requirement.

General design requirements were then specified for the EVT system on both spacecraft. The prime mode of transfer would be via rigid translation rails, hard-mounted to the surfaces of the vehicles. The system would have to allow a crewman to egress his vehicle, transfer to the other vehicle, ingress the second vehicle, and secure it for further mission operations for cases in which the vehicles were (1) docked, (2) undocked, (3) docked with one vehicle without control authority, and (4) without assistance from a fellow crew member either docked or undocked.

Additionally, the EVT hardware provisions were to be designed to provide for adequate support/restraint to permit: (1) control of body attitude and orientation during translations to insure crew safety and prevent damaging exterior equipment on the LM or CM, such as antennas, thrusters, window coatings, or the spacesuit itself, and (2) the performance of work tasks, such as hatch opening and CM LM ingress/egress.

The translation aids were to be simple in design, of minimum weight, fixed in place ready to use, and have cross section and clearance such that they could be easily felt and grasped by the crewman wearing the pressure suit glove without slippage or loss of rigidity under applied loads. A rectangular form factor was specified for handrail cross section, since Gemini experience showed this configuration permitted good manual control of body attitude. Provision for the attachment of local area restraint tethers was judged desirable to permit the crewman to restrain himself quickly to any translation aid at any point during the transfer. Materials used for the installed exterior translation aids were to be such that high or low temperatures and conductivity were not limiting factors in using the devices.

Since the above criteria applied equally to both the CM and LM EVT installation, one might assume the resulting exterior hardware would emerge identical. This would have been the case, had it not been for the fact that the CM and LM are constructed very differently to accommodate different environmental temperatures and forces encountered during their respective mission phases. For example, the LM rides into orbit inside a protective cocoon which opens only after the earth's atmosphere has been left behind (figure 2). Later, the LM is discarded in lunar orbit long before the Apollo crew reenter the earth's atmosphere. The LM exterior therefore contains neither streamlining nor an ablative heat shield and is functionally patterned to contain and shield the crew during orbital and lunar surface phases of the mission. On the other hand, the CM exterior surface leaves the earth covered tightly by a thin boost protective cover which is jettisoned with the launch escape tower prior to earth orbit insertion (figure 1). During the translunar trip it sees the same thermal vacuum environment as the LM. On the return trip, after the LM has been discarded, the CM surface must withstand the high heat of earth atmosphere reentry and provide the proper dynamic lift and drag for stability and control during reentry. After reentry, parachutes are deployed from the CM exterior, and after landing, swimmers scramble on the surface of the spacecraft to attach flotation devices and a sea anchor. It is apparent that any exterior fittings added to the CM surface would have many more mission phases, possible interferences, and diverse environments to deal with than those added to the LM.

The resulting exterior hardware provisions on the CSM and LM for EVT are shown in figure 6. Details of

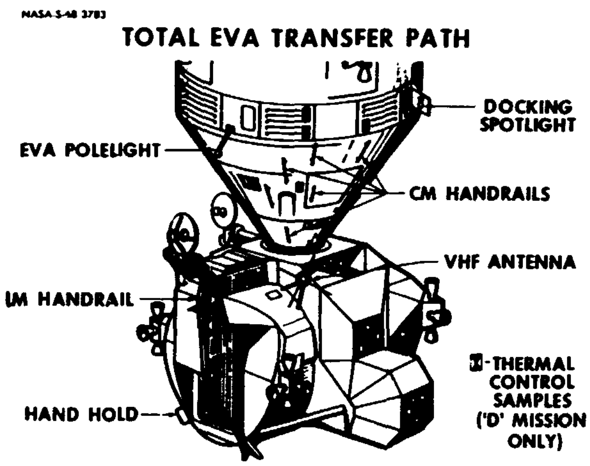

FIGURE 6

the provisions are contained in attachments 2 and 3 to this paper. The CM installation consists of six handrails installed on the conical surface and a seventh circular rail which is located at the apex of the vehicle. A single continuous rail is installed on the LM leading from the front hatch to the docking interface between the two spacecraft. Self-contained radio luminescent lighting is provided on all the handrails and a standoff polelight is installed on the CSM illuminating the transfer area. The following sections will describe the development of the CM and LM exterior provisions and describe the crew personal equipment provided to support EVT.

COMMAND MODULE EXTERIOR PROVISIONS

Late in 1966, EVA team members began technical discussions with North American Rockwell Corporation to determine the feasibility of installing a system of fixed handrails and tether attach points to the outside of the CM. Initial responses were predictably pessimistic since the surface of the CM is covered by about one inch of ablative heat shield material and a boost protective cover fits snugly over the conical surface at earth launch. To add anything to this surface would necessarily disturb the heat shield thermal integrity, the aerodynamic reentry shape of the CM, the fit of the boost cover at launch, the fit of the launch access arm (white room) through which the crew enter the CM on the pad, as well as possible interferences with window vision, parachute riser deployment during low altitude abort situations, operation of the CM uprighting system in cases where the CM comes to rest inverted in the water, or interference with the flotation collar which is attached by swimmers around the CM exterior to help stabilize it after landing.

It was agreed, however, that the job was feasible, and, based upon this, the team began initial placement studies using a full size CM mockup at MSC, Houston. A small portion of the top of the LM spacecraft was constructed and mated with the CM mockup to permit visualization of the interfacing area across which the crewman would make his transfer. Existing locations of windows, launch escape tower attach points, side hatch, and RCS thrusters on the CM necessitated the adoption of a segmented transfer path instead of one continuous rail. Figure 7 illustrates the initial rail locations

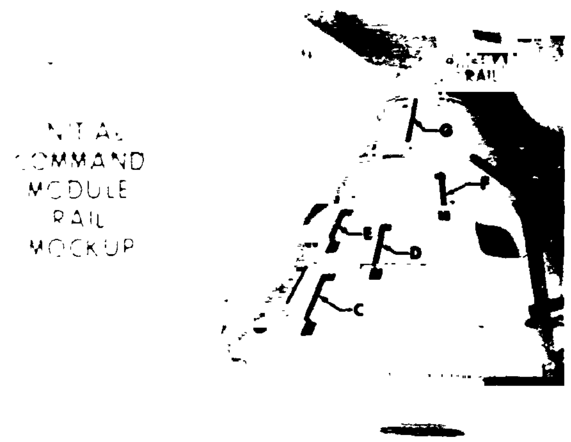

FIGURE 7

chosen for further design study. The rails were designated "A" through "H" on the conical surface of the spacecraft, and a circular handrail was located at the apex, surrounding the docking ring. This circular rail would be used by the crewman to come aboard the CM from the LM should the two vehicles be undocked at the time of EVT. The apex of the CM is located in front and below the field of view of the CM pilot as he looks through the docking window; therefore, the circular rail would not interfere with the pilot's vision during the approach, but he could keep the transferring crewman in sight until he made contact with the apex rail. Rails "G", "F", "D", "C", and "B" provide the path from the apex to the side hatch. Rail "E" would be used to pull the hatch open. Reduced gravity simulations proved rails "A" and "H" to be unnecessary and they were removed from the final system. Attach points for the crewman's personal restraint tethers were recommended at each end of the handrails, providing the crewman a number of choices for tether attachment should he need restraint while translating, opening the hatch, ingressing, etc. The rails were placed within easy reach of each other by a crewman wearing the pressurized Apollo spacesuit, and each rail was required to be of sufficient length to allow the crewman to grasp it simultaneously with both EVA gloves. While rail protrusion off the CM surface had to be minimized, it was determined that a grasp clearance of two inches was essential to permit easy use with the pressurized glove.

At the North American facility, Downey, California, details were worked out to define how the rails could be designed and installed without interferring with the many other spacecraft systems and functions previously mentioned. The rails were designed of aluminum and fastened to the ablative heat shield through heat resistant plugs to prevent thermal shorts to the cabin. The aluminum would burn off during reentry before reaching temperatures which could be of danger in this respect. The plugs were designed to let the rails break off cleanly in the event a parachute riser should contact a rail during chute deployment, and sides and ends were ramped to fair in smoothly with the CM surface and minimize snagging. The boost protective cover was modified to provide bulges to accommodate each rail and faired to prevent snags as the boost cover jettisoned from the CM during launch. For rail "G" the boost cover did not require modification, since in this upper segment a one inch clearance already existed between boost cover and CM surface (figure 8). By designing rail "G" as a deployable "pop-up" rail, it was possible to fit it under the one inch clearance. A fixed design was adopted for the remaining rails, since in the lower portions of the boost cover, no clearance existed, and cover modifications were necessary in any event. The transverse and longitudinal ramps at the end of each rail were designed to double as the restraint tether attach points previously mentioned.

In March 1967, a preliminary design review was conducted to evaluate the proposed installation. As shown in figure 9, Apollo crewmen

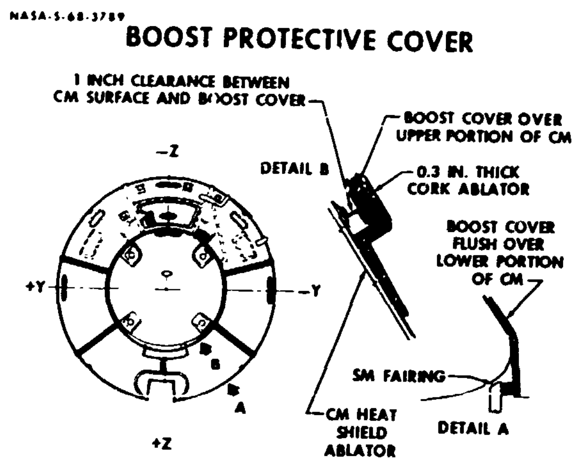

FIGURE 8

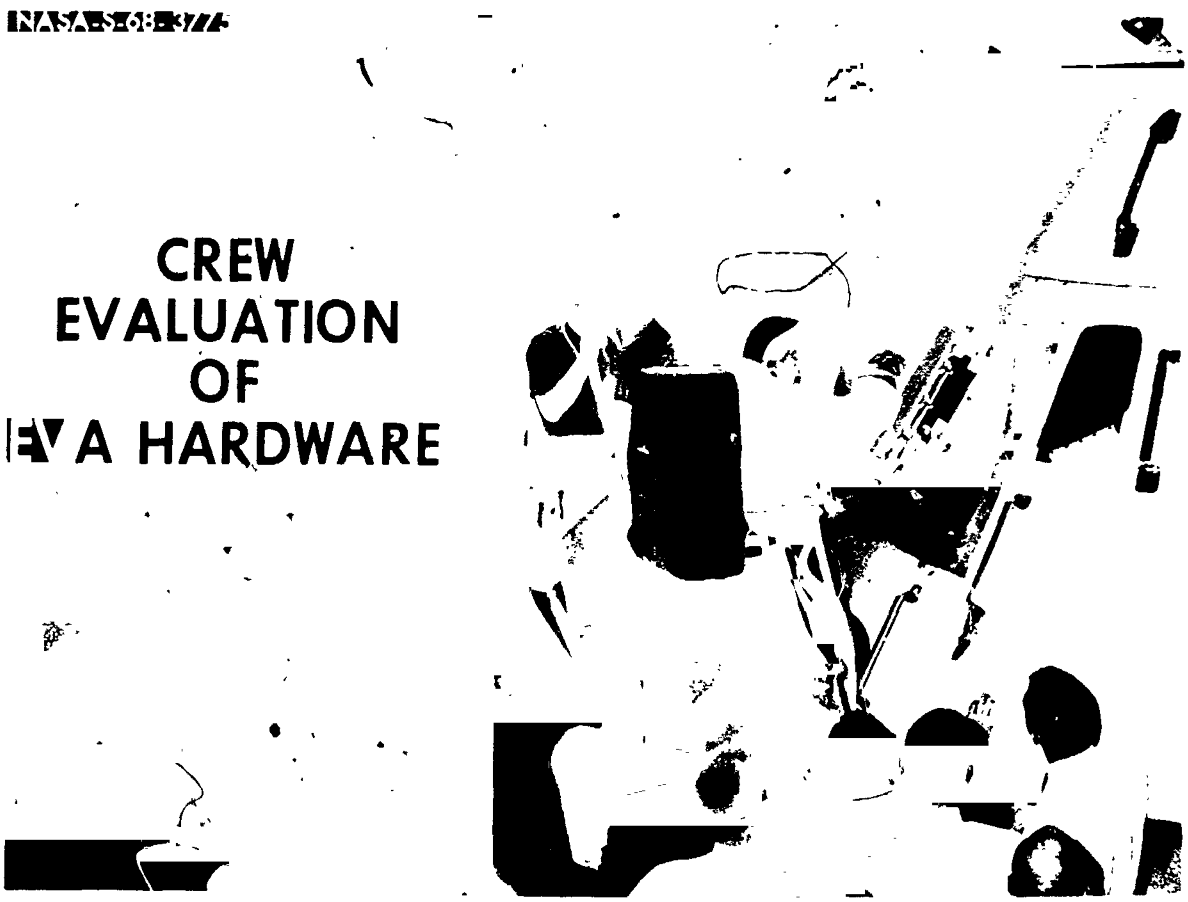

FIGURE 9

utilized the full pressurized spacesuit at this time to check grasp clearances, reach distances, and the hatch opening task.

Having established the basic design and acceptability of the exterior system, the question of how to properly illuminate the CM surface during a night time transfer was investigated. The lighting provisions would illuminate the LM-CM transfer path and facilitate night or shadowed daylight visual detection of the location and orientation of the handrails, tether attach points, and the CM side hatch. Work site illumination for a possible hatch opening task would also be provided. Since the transfer might be with CM and LM undocked, the illumination source would need to be installed on the CM. Two sources of illumination were proposed for evaluation: (1) two deployable EVA polelights which would illuminate the CM with a soft light from either side of the side hatch, and (2) self-contained radio luminescent (RL) discs which could be imbedded in each handrail to provide visual cues to the crewman as he transferred. A lighting review was conducted to evaluate these concepts in June 1967. As a result of this review, only the polelight to the right of the hatch was retained, and RL discs were made a part of each handrail. RL will also be provided on the CM side hatch to indicate the location of the cabin dump valve and the socket for the hatch opening tool. Figures 10 and 11 illustrate the EVA illumination aids.

Attachment 2 to this paper illustrates the complete CM EVT exterior installations, including illumination provisions. A flattened tubular handrail cross section of approximately 1 1/4" x 5/8" was adopted since this configuration had been proved on Gemini missions to be very satisfactory for manually controlling body position during EV transfers. This rail shape presented minimum protrusion off the CM surface, and, by proper selection of tube thickness and attachment method, was made sufficiently strong to withstand the following expected loads:

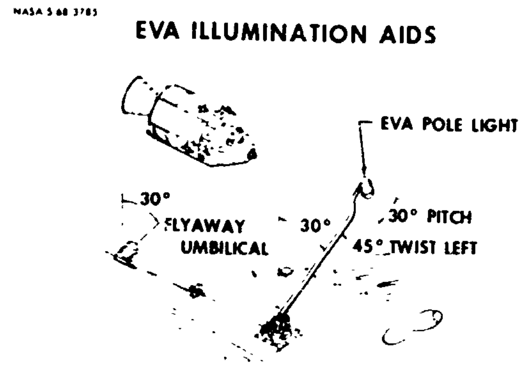

FIGURE 10

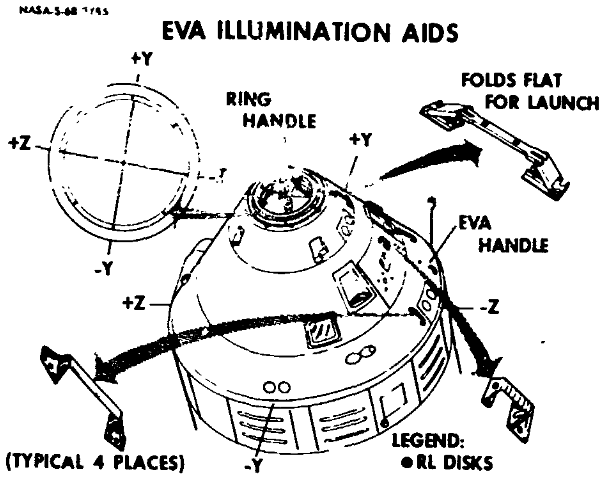

FIGURE 11

(1) Ground handling loads during shipment, installation, and assembly of the CSM on the launch pad. The rails could be expected to provide a natural place for workmen to grasp or use as a step.

(2) Manual loads imposed by the crewman while translating and controlling body positioning.

(3) Reaction loads applied by the crewman's local restraint tethers.

By designing the rails to withstand a force of 600 lbs, an adequate safety margin was provided for the loads identified above. As previously mentioned, the rail attachments were designed to break cleanly under very high loads (2,000 lbs for pilot chutes; 20,000 lbs for drogue chutes) which could be imposed by parachute risers during chute deployment.

LUNAR MODULE EXTERIOR PROVISIONS

Concurrent with initial mockup and feasibility studies for CM exterior EVT provisions, a full scale LM ascent stage mockup was utilized at MSC Houston to investigate possible configurations and routing for a LM handrail. This exterior rail on the LM would provide the transfer path from the front hatch area, across the front face of the vehicle, to the docking interface, where the crewman would make the transition from the LM rail to the CM rails.

Design and placement of the LM EVT provisions was to be decidedly easier than on the CM since the questions of aerodynamic interferences, reentry heating, and earth landing system interference did not apply to the LM outer surface.

From mockup studies using both the CM and LM vehicles it was proposed that a single continuous rail, approximately 130 inches long, be placed adjacent to the front hatch (as shown in figure 12) to

LUNAR MODULE
EVA HANDRAIL
MOCKUP

FIGURE 12

lead directly up the front face beam shield, past the rendezvous radar antenna, to the area of the docking ring.

Detailed design studies were undertaken at Grumman Aircraft Engineering Corporation for the LM rail installation concurrently with previously described CM design studies. The rail location was found to be feasible as proposed, and the engineering effort was concentrated on adopting the rail supports to the relatively lightweight construction LM structure to achieve the required rail strength, while at the same time insuring no thermal leak paths into the crew compartment.

Since valuable fuel is required to land and reorbit every piece of installed equipment on the LM ascent stage, every effort was made to keep the handrail as light as possible. The total final installation weighs approximately six pounds.

Supports for the rail are located at intervals of about one foot. Should the crewman wish to stop enroute to the CM, his local restraint tethers can be attached to either the rail supports or the rail itself. Since the LM rail is continuous, the crewman will be able to make the transfer along the rail to the CM without letting go enroute. However, for added safety, illumination aids to provide nighttime visual cues were provided by installing radio luminescent discs at intervals along the rail.

Attachment 3 to this paper illustrates the LM exterior EVA provisions.

DEVELOPMENT TESTING OF VEHICLE EVT PROVISIONS

The development tests necessary for the exterior EVT installation were of two general ty .s:

(1) Hardware qualification tests to insure that the handrails would perform as designed under the heat, shock, and vibration loads encountered during the mission, and

(2) Functional tests of the transfer path, utilizing fully suited Apollo crewmen, to verify the adequacy of the equipment and develop procedures for use during the missions.

In the first category, the CM rails were of primary concern because of the close interface with the boost cover, heat shield, and earth landing system during launch and recovery phases of the mission. The best test would obviously be to fly the installation on an actual spacecraft, therefore test rails were installed on CM Spacecraft 017 and 020, scheduled for the unmanned flights Apollo 4 and Apollo 6. Rails "B" and "F" were flown on Spacecraft 017 and rails "B", "D", "E", "F", and "G" on Spacecraft 020. Post recovery inspection of the rails showed that they survived the reentry heat with no apparent damage, and were in fact very useful to recovery team swimmers as they worked to install the flotation device. Figure 13 shows Spacecraft 017 just following splashdown, with the swimmer holding rail "B." Figure 14 was taken of Spacecraft 020 by one of the swimmers approaching from the water, and figure 15 shows again the excellent condition of the rails on Apollo 6. The recovery team member is making good use of rail "B" in figure 15.

An additional test was conducted by North American Rockwell Corporation to investigate the failure mode of the rails in case a parachute riser should sweep across the CM conical surface during the chute deployment. This could take place as the CM oscillates during normal reentry and chute deployment or in cases of launch abort in which the CM tumbles as the chutes deploy. Risers were passed over a segment of the conical surface containing typical handrails, at the anticipated tension loads, and the rails were broken off cleanly at their base mountings. No snagging was encountered in these tests.

No special qualification tests were necessary for the LM rail installation other than those normally associated with thermal

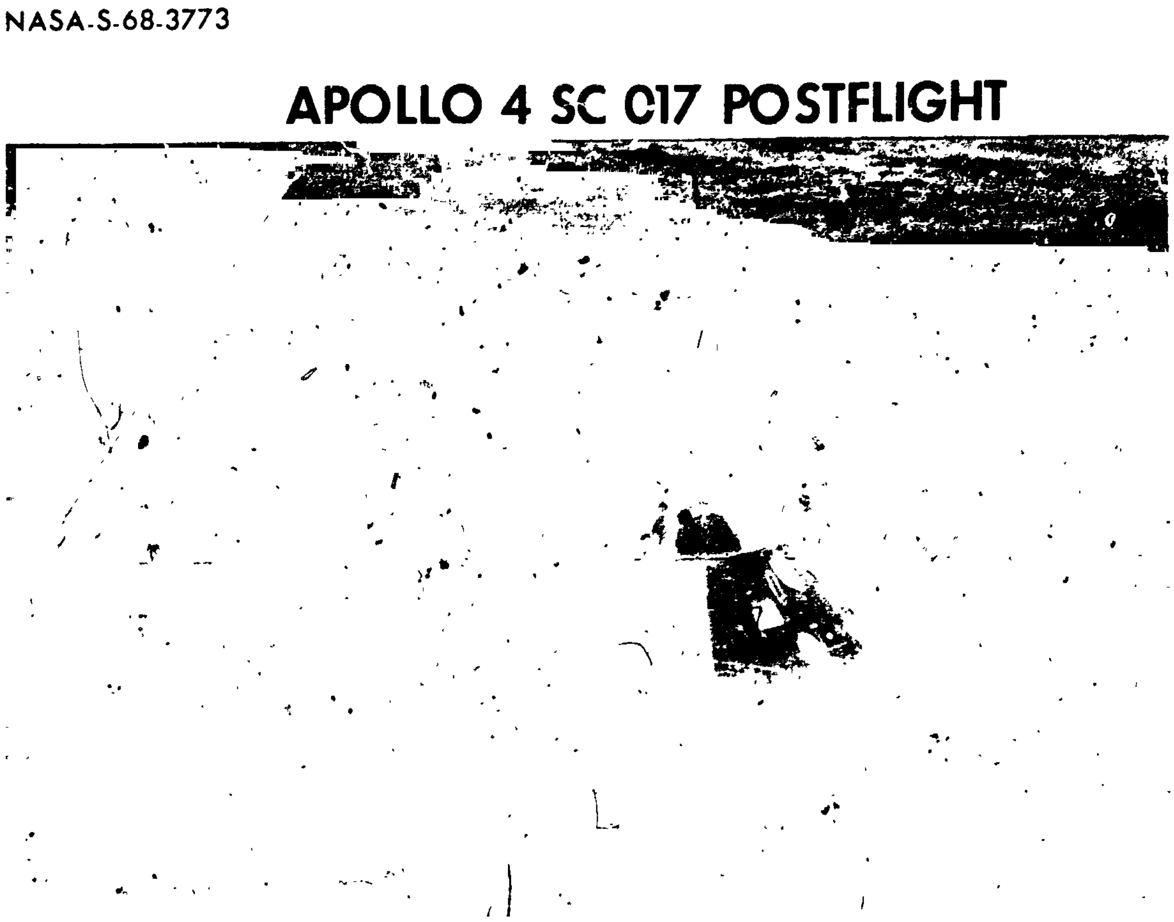

FIGURE 13

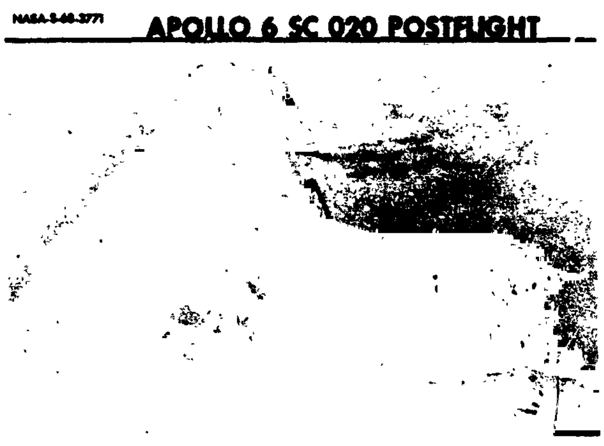

FIGURE 14

and vibration qualification of the LM external surfaces.

In the second category of functional testing, the objective was to exercise the complete LM-CM transfer path in conditions as close to the zero gravity environment of space as possible in order to verify that the placement, size, shape, and strength of the provisions were in all respects adequate to provide the emergency EVT capability. Although no single test medium on earth simulates fully the vacuum and zero gravity of space, two methods have been developed which closely approach these conditions and, when employed together

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APOLLO 6 SC 020 RECOVERY

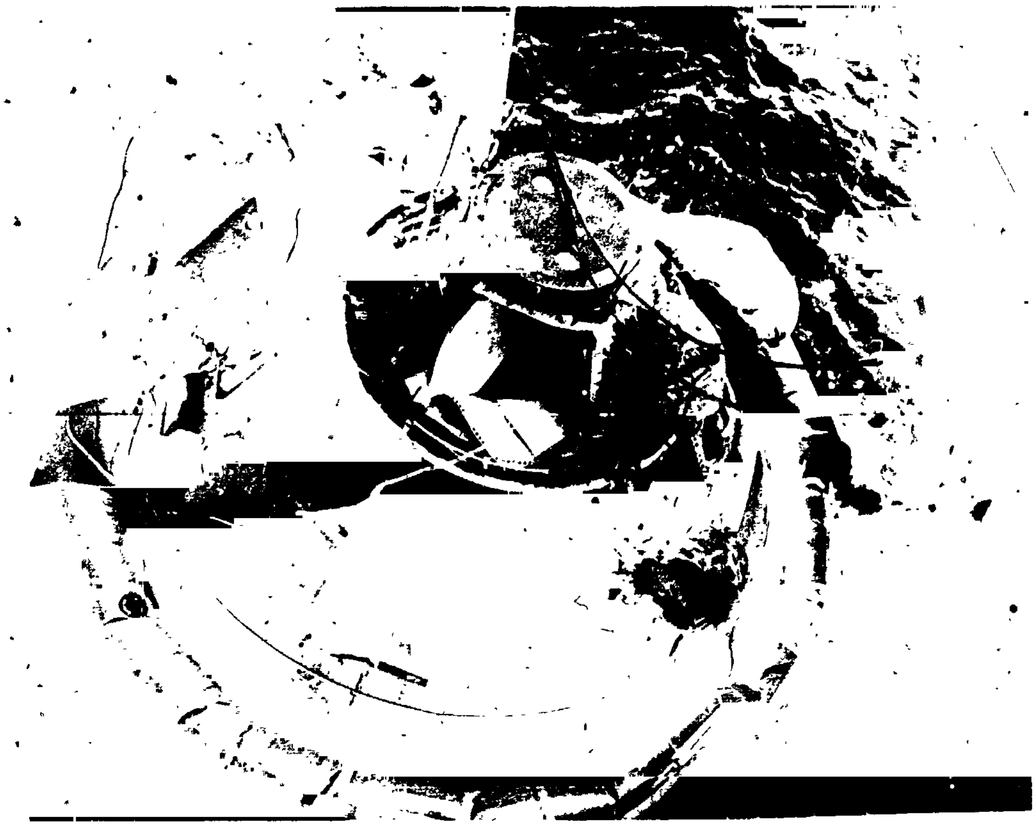

FIGURE 15

in a test program, they give a high degree of confidence that results will be similar to those to be encountered later in space. The first method utilizes a USAF KC-135 aircraft to fly parabolic flight paths to achieve about 30 seconds of zero gravity in the cabin area. The second employs an underwater test facility to provide longer periods of simulated weightlessness, although an obvious drawback is the ever present viscosity of the water which impedes movement and tempts the test subject to swim.

For tests aboard the KC-135, partial mockups of both the CM and LM were installed in the cabin. The LM mockup permitted the Apollo crewmen to egress the LM through the front hatch and translate along the LM rail to the vehicle docking interface. The CM mockup was then utilized to complete the transfer maneuver, attach personal restraints, and perform the hatch opening task. Figure 16 shows Apollo crewman Michael Collins translating along the LM rail after clearing the front hatch area, and figure 17 shows the crewman opening the CM side hatch using rails "D" and "E."

The larger volume of the water immersion facility permitted joining full sized LM and CM mockups underwater for a complete

FIGURE 16

FIGURE 17

uninterrupted transfer simulation from hatch to hatch. Several Apollo crewmen performed this exercise with apparent ease and rated the installation as adequate for its designed purpose. Figure 18 shows an Apollo crewman making the transfer in the underwater facility at MSC Houston.

INDIVIDUAL CREW EQUIPMENT FOR EVA

The extravehicular space environment to be encountered in earth or lunar orbit is similar, though somewhat less severe than that to be encountered during lunar surface EVA. In each case, the crewman will be exposed to the hard vacuum, direct sunlight, and micrometeoroids. On the lunar surface, additional heat will be encountered from reflections off the surface and from craters, and a weak gravity field equal to one-sixth that on earth will give the crewman a vertical reference and the capability to walk over the lunar surface. In orbit there will be no apparent gravity field acting on the crewman and the heat loads on his spacesuit will be somewhat less severe. In view of the similarity between extravehicular environments in orbit and on the lunar surface, the same personal equipment will support both.

The combined spacesuit-life support system developed for the lunar surface EVA is known as the EMU (Extravehicular Mobility Unit) and consists of the following items of equipment:

The PGA (Pressure Garment Assembly) is the basic spacesuit, which like the Gemini suit contains an integrated multi-layer thermal-meteoroid protective garment. To a greater degree than earlier suits, the PGA is an approximately constant volume suit which allows greater general mobility while pressurized.

The EVVA (Extravehicular Visor Assembly) fits over the spacesuit helmet and contains two coated visors which may be selected much like sunglasses to protect from direct sun glare or reflected glare from the spacecraft or lunar surface.

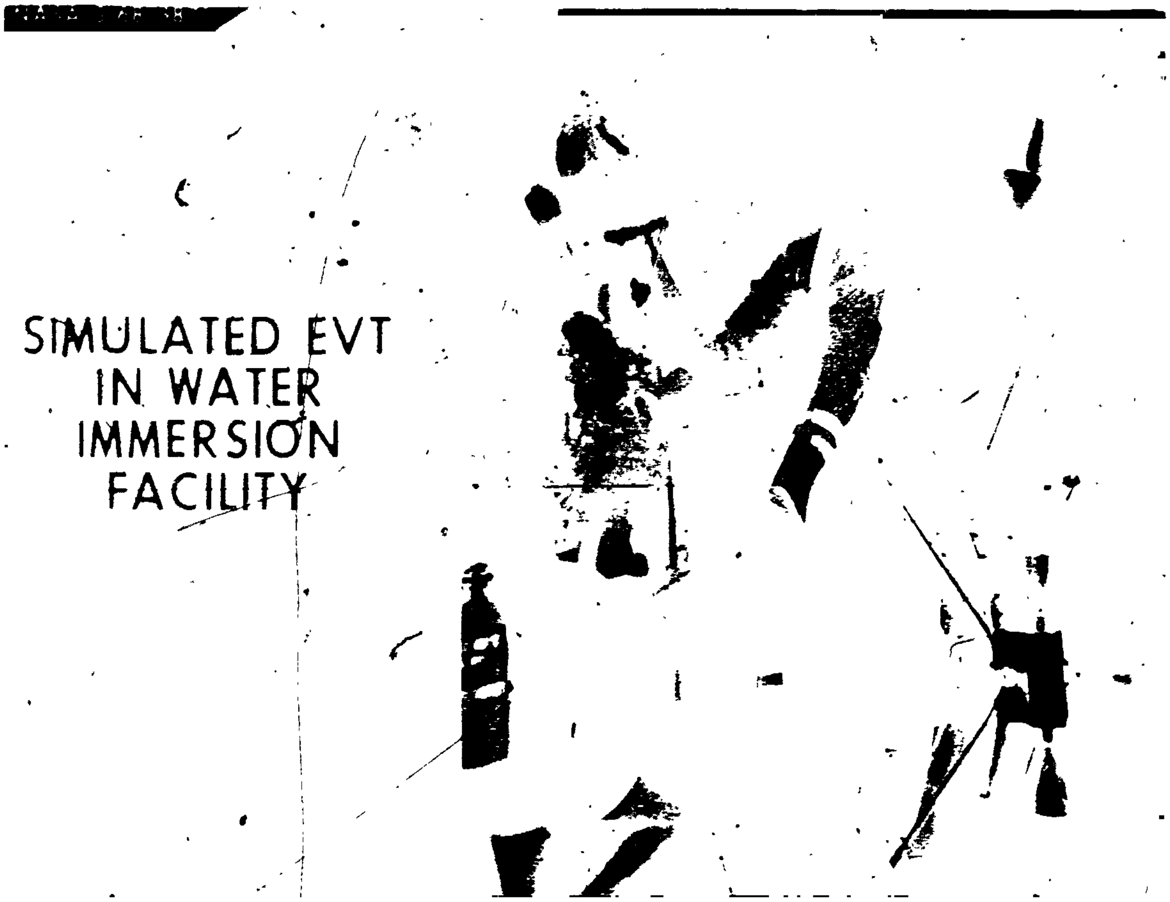

FIGURE 18

A pair of lunar boots is worn over the PGA boots to provide firm footing on the lunar surface and provide additional layers of thermal-meteoroid protection

A pair of extravehicular gloves is substituted for the normal intravehicular gloves. The EV gloves contain added thermal-meteoroid protective layers.

The PLSS (Portable Life Support System) is worn on the crewman's back during lunar surface EVA and is capable of supplying oxygen and cooling for a four-hour excursion outside the LM. A backup emergency device, the OPS (Oxygen Purge System) is worn as an attachment to the PLSS. The OPS provides an independent 30-minute supply of gaseous oxygen as a backup to the PLSS. A multilayer thermal cover for the PLSS gas connectors protects against the lunar heat loads.

Current plans are to leave the lunar boots, gas connector covers, and PLSS on the lunar surface at lift-off. The OPS would be carried into lunar orbit to serve as the source of extravehicular life support for the contingency EVT, should it be necessary.

The personal equipment items to be provided the crewmen uniquely for the EVT consist of a single safety tether line and four personal restraint tethers, two for each man. The safety line will be approximately 25 feet long and will be utilized to provide a positive connection between the EV crewman and the spacecraft at all times. This line would be attached inside the LM cabin by the first crewman making the transfer to the CM. After entering the CM, the first crewman would attach his end of the safety line inside the CM cabin. The other end of the line would then be disconnected from within the LM cabin by the second crewman and attached to his spacesuit prior to his making the transfer. The short restraint tethers are fastened at each side of the crewman's waist and are adjustable in length. Their purpose will be to provide local area restraint while performing work tasks such as hatch opening or to permit the crewman to rest and relax without the necessity of continuously grasping a handhold. These small restraint tethers will also be useful for temporary restraint of loose items.

The final item of personal equipment for the EVT is the small hatch opening tool which is supplied to the first crewman making the transfer to the CM. It is to be used only in case the third crewman inside the CM has not already opened the side hatch in preparation for the EVT.

Figure 19 shows the full range of personal equipment to be used in Apollo EVA.

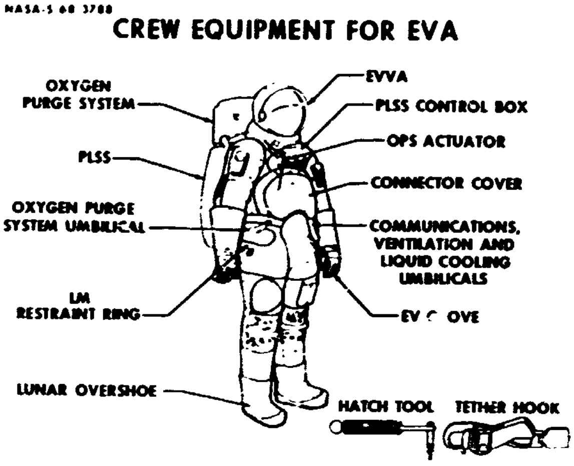

FIGURE 19

MISSION "D" PLANNED EVA

The types of missions leading up to a lunar landing mission have been designated as types "A" through "G," where "A" type missions are unmanned CSM's, "B" type are unmanned LM's, etc., culminating in "G" types being manned lunar landing missions. In this sequence, the "D" type mission is an earth orbital mission in which both the CSM and LM will perform manned maneuvers. Since the manned LM undocks from the CSM in a mission of this type, the EVT capability must be operational and ready to support a possible contingency transfer by flight time. Extensive training will be conducted in various simulated zero gravity facilities prior to the time the "D" type mission is flown, although the total operational environment of thermal vacuum, zero gravity, and flight hardware and vehicles will not be encountered until the actual flight. Thus, the EVT will be demonstrated in the mission timeline prior to manned undocking of the LM.

The planned in-flight EVT demonstration will occur during the fourth day of the mission, following a full day of LM checkout and systems exercising with the LM and CSM docked. Undocking, separation, rendezvous, and redocking will occur on the mission day subsequent to the EVA.

The EVA day will begin with a short period of LM systems checkout to insure that those vehicle subsystems necessary to support the activity are functioning properly. A period of EVA preparation will follow the vehicle checkout, and this period will be devoted to donning and functionally checking out the EVA life support equipment and generally configuring the vehicle and personnel for the ensuing activity. Following this preparation activity, the LM will be depressurized and the EVA crewman will egress the vehicle through the front hatch and mount a 16 mm sequence camera on the boarding platform handrail. This camera will face the transfer path and will view the crewman's activity during transfer and CM ingress and egress. The expected view is shown in figure 20. The crewman will then ingress the LM and prepare to take a timed transfer to the CM.

NASA-S-68-3781

CM AS SEEN BY LM EVA CAMERA

FIGURE 20

Concurrent with LM EVA preparation, the crewman remaining in the CM will prepare that vehicle to support the EVA. He will mount a 16 mm sequence camera to the CM side hatch in such a manner that when the hatch is fully opened the camera will view the transfer path and the LM egress and ingress. The expected view is shown in figure 21. The CM side hatch will be opened prior to the first egress from the LM so that photo coverage will be available for all activities performed during the EVA. Both the 16 mm cameras will be remotely controlled by the crewmen remaining in the vehicles, thus camera running time can be limited to those periods when the EVA crewman is within camera range.

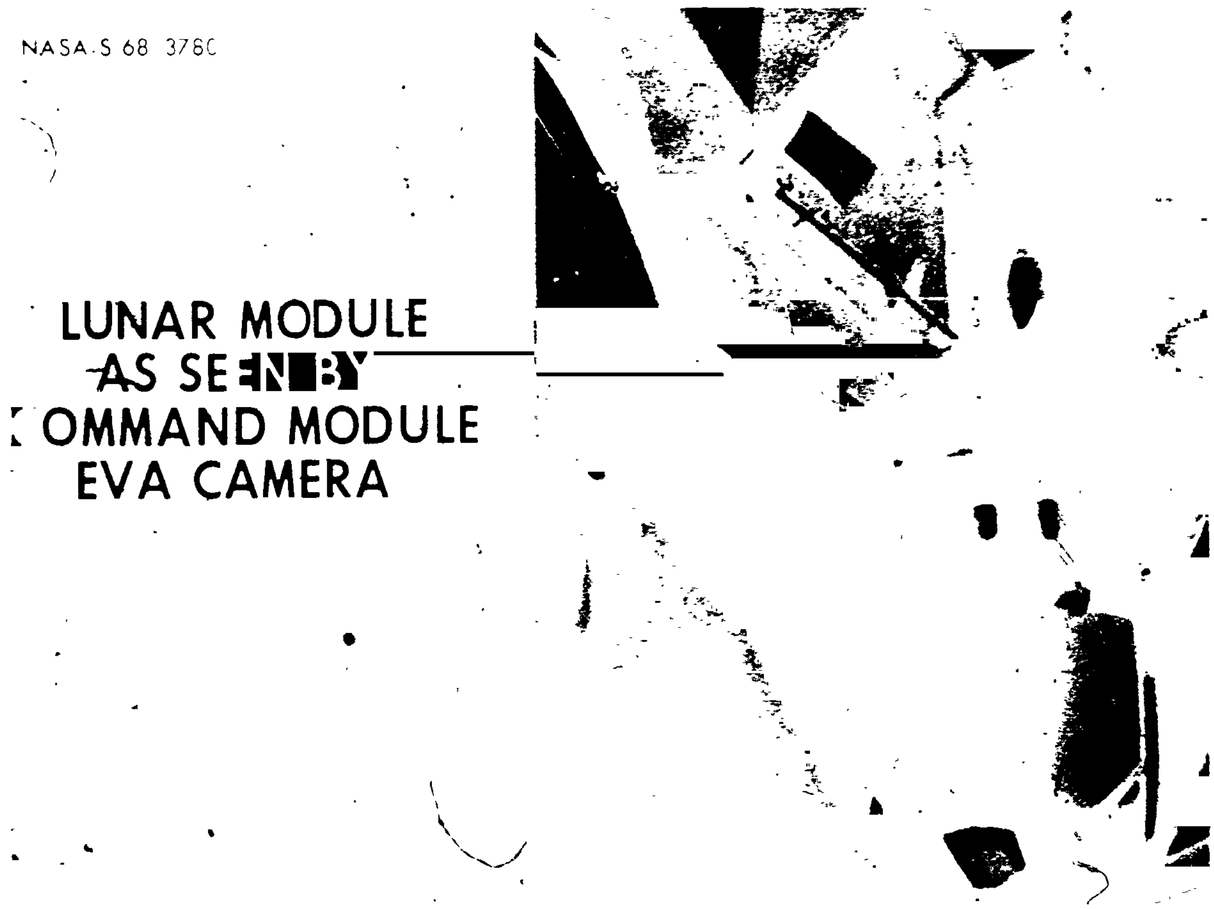

FIGURE 21

The EVT will be performed as a timed end-to-end activity, uninterrupted by other activities, such as the mounting of the camera. The crewman will egress the LM, translate to the CM side hatch via the LM and CM handrails, and ingress the CM. Following a rest period, the EVA crewman will egress the CM and retrieve several thermal control samples located on the surface of the CM and SM in the area near the hatch. The samples will be passed to the crewman in the CM for stowage, and the EVA crewman will return to the LM via the transfer rails. This first sequence of activities will commence at the beginning of a daylight pass and terminate with the crewman back at the LM boarding platform before the vehicles enter the dark portion of the revolution.

The EVA crewman will be restrained in a standing position on the LM boarding platform during the nightside pass and will evaluate the various lighting aids associated with the EVT during that time. The crewman in the CM will actuate the various lighting modes, as necessary.

Following the sunrise subsequent to the lighting evaluation, the EVA crewman will retrieve a thermal control sample located on the forward surface of the LM near the commander's window. This sample will be passed to the crewman in the LM and the remainder of the EVA will be devoted to activities in the vicinity of the LM forward hatch. As the total EVA time approaches two hours, the EVA crewman will ingress the LM and the EVA period will be terminated. A period of post EVA activities will conclude this phase of the mission and will include doffing the EVA equipment, stowing the items used or collected during the EVA, and configuring the vehicles and personnel for an eat and sleep period.

Since current mission planning calls for this to be the only planned EVA prior to the surface excursions of the lunar landing mission, the timeline has been extended beyond that necessary to demonstrate the EVT in an attempt to gather data from, and operational experience with, the hardware intended to support the surface excursions. The crewman conducting the EVT will wear the full array of life support equipment to be worn during lunar surface EVA with the exception of the lunar surface boots and a thermal protective cover for exposed spacesuit gas connectors. He will have available both the PLSS and OPS, although only one of these units would be used during an actual contingency EVT. By having available both units, the crewman has a redundant life support system in case of emergency, and he is additionally gathering life support system performance data pertinent to the lunar surface mission.

CONCLUSION

This paper has been addressed to the recognition of the need for an alternate return path from a space vehicle which has for some reason lost its capability to dock with its sister ship or, having docked, is incapable of supporting an intravehicular crew transfer. In either case, the only path of return for the crewmen is via extravehicular transfer. The efforts brought to bear on this problem were aimed particularly at the mainline Apollo contingency transfer, but the technology developed is a step toward EVA as an operational capability. The authors hope that the hardware and procedures developed for Apollo EVT will become part of an ever growing technology aimed toward future programs and missions in which EVA will be an operational requirement rather than a contingency procedure.

[1] Gemini EVA Summary Report, MSC-G-R-67-2

ACKNOWLEDGMENT

The development and implementation of the EVA provisions and procedures presented in this paper are a result of the combined efforts of the Apollo EVA team, the associate contractors, and all those associated with the previous Gemini EVA exercises. Rather than attempt to list each individual, the authors wish to gratefully acknowledge the dedicated efforts of all who have helped to provide an Apollo in-flight EVA capability, and express thanks for their contributions.

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IN-FLIGHT MAINTENANCE OF SPACE STATIONS AND SPACECRAFT

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SUMMARY: There has been considerable discussion of whether or not in-flight maintenance is a valid requirement. A review of past experience clearly shows that human error and equipment failure are ever-present, and have a definite effect on mission success; therefore, the major issue is not whether an in-flight maintenance capability should be developed, but rather how this capability can be developed in terms of technical and economic feasibility. Current and future space maintenance requirements are presented, showing the dynamic relationship of these requirements with reliability, maintainability, equipment design and human factors criteria, together with the ultimate effect on mission success. Mission-oriented maintenance concepts are established for each mission category, which include earth orbital, lunar orbital and planetary fly-by missions. Each concept considers: what can be replaced or repaired, what can fail but cannot be replaced or repaired in-flight, what effect replacement and repair action (or its omission) has on mission success or crew safety, and how often replacement and repair is required. The four basic maintenance modes presented are: remove-replace-discard, repair without removal, remove-replace-repair, and remove-repair-replace. Simplicity, improved equipment reliability, crew safety and improved probability of mission success are among the criteria established for each maintenance task. Other considerations include: crew time available for maintenance, fault detection and isolation systems, maintenance aids, the requirement for resupply missions and maintenance techniques necessary for intravehicular activity (IVA) and extravehicular activity (EVA). This approach results in in-flight maintenance policies and plans which further assure attainment of mission objectives.

INTRODUCTION

The rapid pace of ever-expanding technologies coupled with the high cost of research and development (R&D) activities in the environment of space make it both logical and realistic to conclude that future manned space-flight missions will be of extended or long duration in order to obtain the maximum return on the R&D dollar investment and to minimize the gaps in technology gains.

The planning of these missions is a very complex task with many decision factors to be considered. The selection of mission objectives with their resultant tasks requires careful and thorough analyses to be compatible with and support the national space program objectives. Also, technical and economic feasibility, along with crew safety, are prime decision factors which are employed throughout the mission planning process.

In planning the operation of any hardware system, whether it be for a space or terrestrial environment, it is essential to identify the factors/ influences which can have a significant effect on mission success. When adverse conditions are identified, it is necessary to provide a means of compensating for them. Thus, some form of contingency protection must be made available.

The mission planner has four (4) basic options in the selection of a contingency protection approach which will assure the required degree of operational availability. They are:

- To overdesign the mission hardware systems
- To incorporate redundancy in the mission hardware systems
- To accept some degree of degraded performance
- To provide an In-Flight Maintenance (IFM) capability

These options are interrelated and one approach need not be pursued to the exclusion of the other three. In most cases a decision will be made to employ a mixture of all four (4), depending on technology level (state-of-the-art), system design complexity, mission duration, reliability criteria, crew safety, etc.

It has been theorized that IFM is not required because of the high reliability and system redundancy which will be incorporated in space stations and spacecraft. This might be true when considering short term spacecraft such as Mercury and Gemini; however, when dealing with manned spaceflight, that is measured in terms of months and years, the inescapable fact remains that equipment does fail and the probability of human error does exist. Therefore, logical and sound judgement must be exercised to strike a balance between two (2) philosophical extremes - optimism and pessimism. The balance point between these extremes is realism.

This paper will discuss the development and implementation of the most realistic approach to contingency protection for long term manned spaceflight missions - an IFM capability. Simply defined, IFM consists of all actions necessary for retaining an equipment item in or restoring it to a specified condition during manned spaceflight missions.

In-Flight Maintenance is a part of an integrated Space-Terrestrial Logistics (STL) System. Figure 1 shows the interface and interdependancy of IFM with the other elements of that system Heretofore, the conventional support methods have been feasible, largely due to the short duration of current orbital flights. However, with increased emphasis being placed on manned orbital missions of extended duration and on preparation for earth-lunar missions, it is necessary to plan the development of a system capable of supporting these operations. The criticality of earth-orbital, lunar, and interplanetary operations, in terms of crew safety, support response times, allowable downtimes of flight hardware, etc., together with the requirement for a precision interface between space and terrestrial logistics functions, make it mandatory that an effective STL system plan be developed and implemented.

Past experience has also shown that effective support does not just happen; it is the result of a reiterative planning process. An IFM capability must be planned, tested, evaluated, and replanned through many iterations before a satisfactory capability will emerge. Therefore, it is logical to assume that an IFM capability will be developed as an evolutionary process. The initial capability will be relatively simple; however, as empirical data is accumulated, the scope and depth of IFM technology will be increased.

A number of space maintenance (IFM) studies have been conducted in conjunction with the prototype

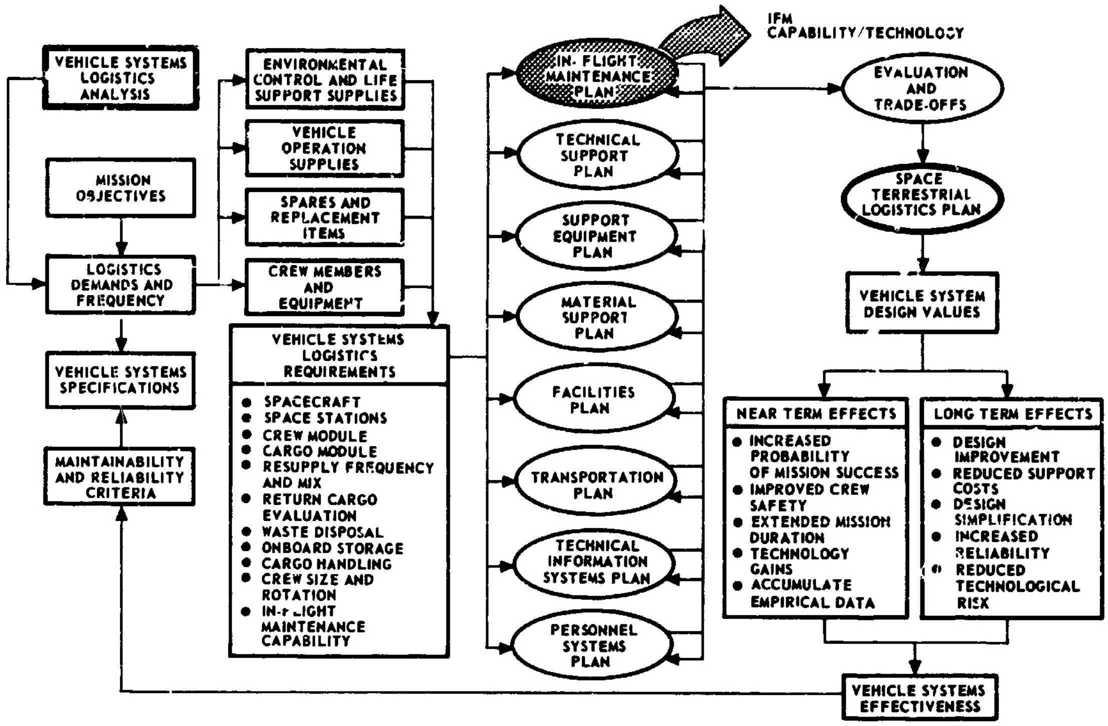

FIG 1 - THE SPACE-TERRESTRIAL LOGISTICS PLAN

development and testing of a variety of space maintenance tools. These efforts have generally produced good results; however, they have been fractionated and have not provided for the development of an integrated IFM technology and capability which considers the total spectrum of manned spaceflight missions. Although some missions may not require every facet of an integrated IFM capability, the IFM capability should not be fractionated until it first has been analyzed and defined as a complete entity. The development sequence for an integrated IFM technology and capability is shown in Figure 2.

Early in the development of any support function, existing capabilities should be examined to determine whether they can be used "as is," if they can or should be modified, and lastly, if a complete new capability is required. All too often an attempt is made to "reinvent the wheel" instead of using the technology and other support resources that are already available and capable of fulfilling support requirements. Therefore, new capabilities should be acquired through an evolutionary process which provides testing and evaluation milestones that progressively measure the effectiveness of the development approach. Figure 3 shows the analytical path followed to determine the initial scope of the development effort.

The approach taken, in this paper, for the acquisition of a space maintenance capability is unique in that it:

- Considers the total spectrum of manned spaceflight missions.
- Provides for the development and implementation of an integrated IFM technology and capability.

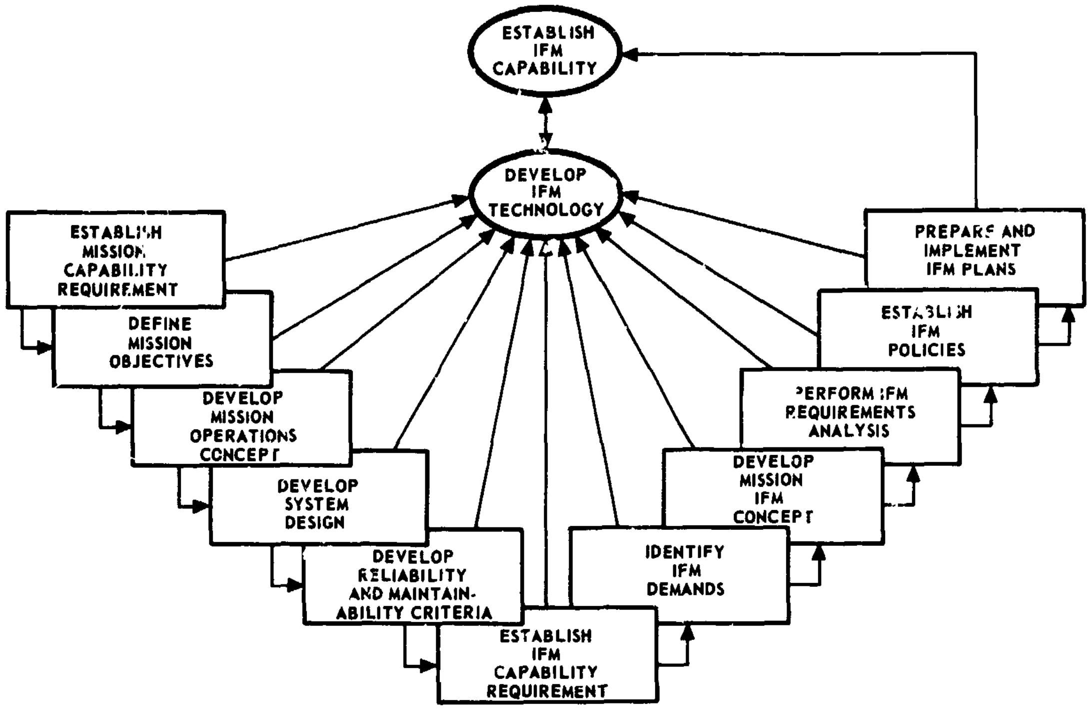

FIG. 2. - DEVELOPMENT OF AN IFM TECHNOLOGY AND CAPABILITY

- Employs modular techniques for mission-oriented IFM concepts.
- Combines simplicity and sound judgement with sophistication and automation.
- Integrates with an STL Plan.

The remaining text of this paper is arranged as follows:

- IFM Demands
- IFM Concepts
- IFM Requirements
- Policies and Plans
- Conclusions
- References

GUIDELINES

The number and complexity of tasks involved in the development and implementation of an IFM capability make it necessary to first establish the scope and depth of IFM. The following guidelines are realistic, and provide a frame of reference or parameters wherein the ensuing discussion is presented.

- A requirement either does or will exist for short term, extended duration, and long term manned spaceflight.
- Crew safety is a prime consideration in all phases of development and implementation of an IFM capability.
- Maximum use must be made of the technologies and materials which presently exist.
- An IFM capability will be developed to bridge the gap between required and achieved system reliability.
- The development of an IFM capability requires an integrated effort of the Design, Reliability, Maintainability, Human Factors,

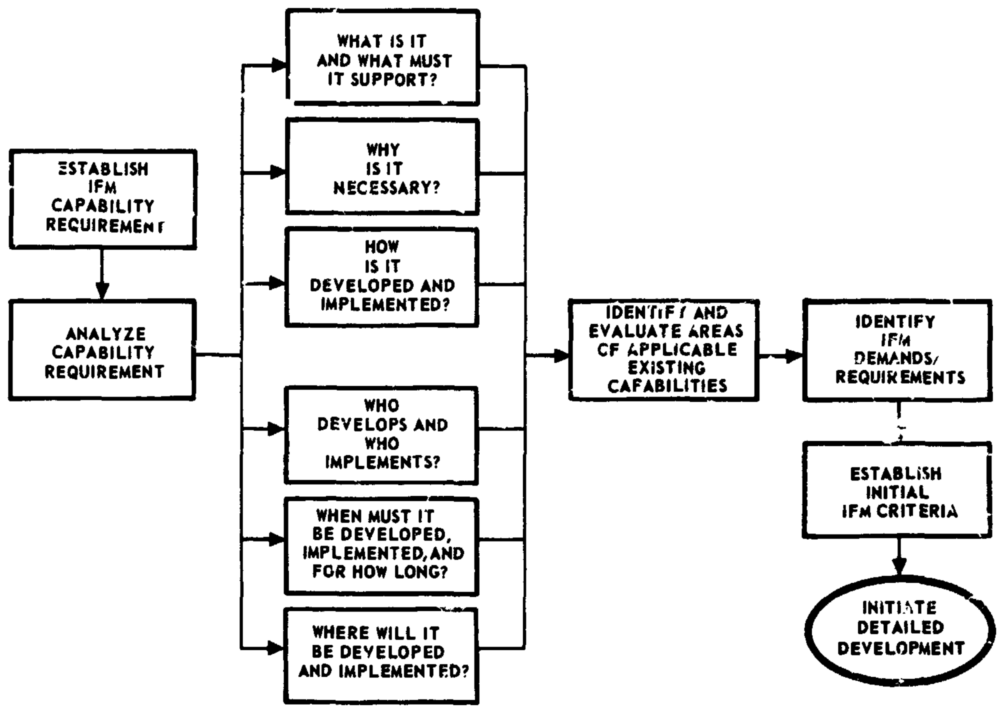

FIG. 3. INITIAL IFM CAPABILITY ANALYSIS

Systems Safety, and the Logistics disciplines.

- An IFM capability must be based on realistic requirements and not desires.
- IFM planning is an integral part of the mission planning process.
- A Zero-Gravity environment is assumed for all IFM actions discussed in this paper.
- For purposes of this paper all maintenance performed during lunar landing and lunar exploration missions, and during interplanetary landing and exploration missions is initially considered IFM.

OBJECTIVES

The objectives of this paper are:

- To present a realistic and comprehensive approach to In-Flight Maintenance in a space environment.
- To identify the need for an integrated IFM capability and IFM technology.
- To describe the rationale and development process utilized in their acquisition.
- To establish the interdependancy of IFM, Hardware Design, Reliability, Maintainability, Crew Safety, Human Factors, and Logistics: and how this integration of engineering disciplines relates to mission success.

IFM DEMANDS

The first step in the development of an integrated IFM capability is the definition of mission objectives which create certain IFM demands. These demands are mission-peculiar (MP) and result from a variety of space station/spacecraft mission operations. The MP demands, in turn, create maintenance resource (MR) demands. IFM demands possess the following characteristics:

- Type or Kind
- Magnitude
- Frequency
- Location

A more detailed breakdown of the demand characteristics is shown in Figure 4.

Mission-Peculiar Demands

MP demands are generated by the operation of the following vehicle systems:

- Manned Space Stations
- Manned Spacecraft
- Unmanned Space Stations/ Spacecraft

These demands are functionally divided as follows:

- Space Station/Spacecraft Primary Flight System (i. e., Structure, Guidance and Control, Propulsion, etc.)
- Environmental Control and Life Support (EC/LS) Systems
- Scientific and Biomedical Experiments
- IFM Technology Experiments

A summary of mission-peculiar demands is shown in Figure 5.

The generation of mission-peculiar demands for IFM has been presented in general terms of vehicle system operation. More specifically, these

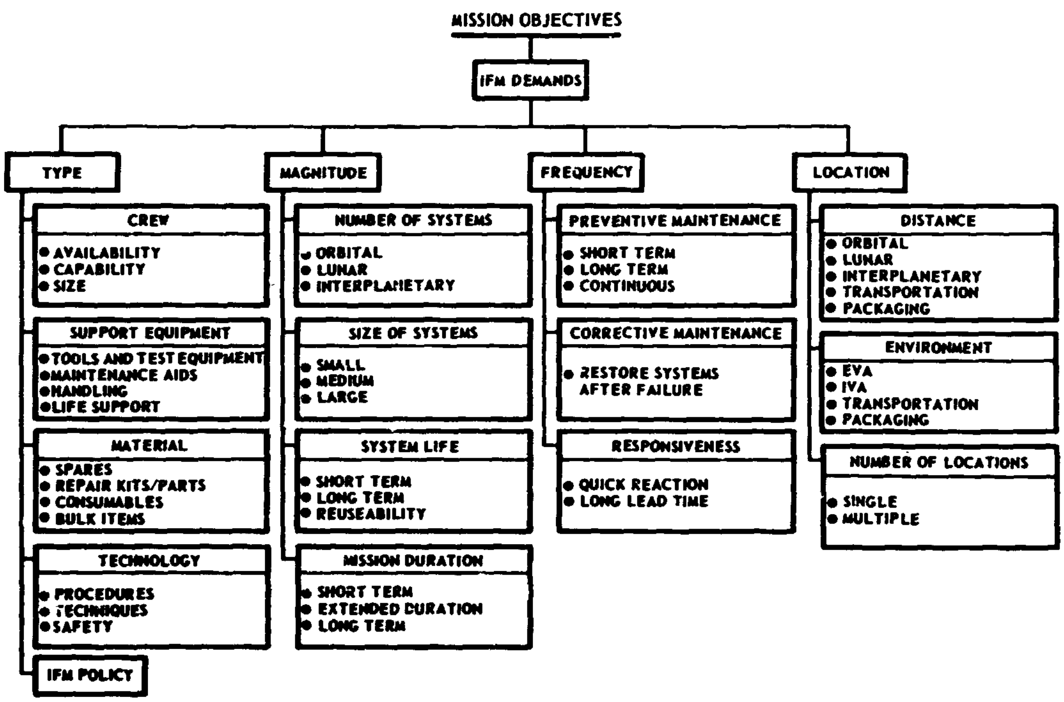

FIG. 4. - IFM DEMANDS

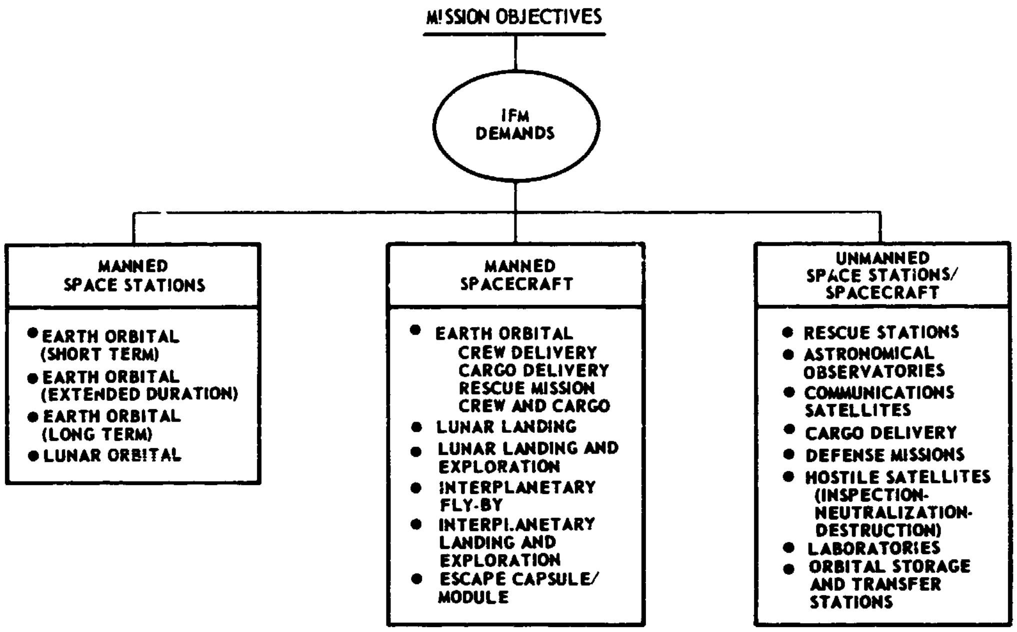

FIG. 5. - SUMMARY OF MISSION-PECULIAR IFM DEMAND GENERATORS

demands are caused by one or more of the following conditions:

- Random Failure
- Damage (Human or Environmental Induced)
- Wearout
- Resource Consumption (Normal and Emergency)

Maintenance Resource Demands

Having identified the mission demands, the next step is to identify the resource demands and to determine how they will be applied. Maintenance resources consist of technology, people, material, and services which are applied individually and collectively to the right place, at the right time, in the required amounts and in an operable condition. These resources possess a variety of characteristics that materially affect their application. Examples of these characteristics or qualities include whether the resource, being considered, is:

- Capable of meeting the demand
- Dependable
- Available
- Consumable
- Expendable
- Restorable
- Reuseable
- Modifiable
- Storable
- Multi-Utility
- Transportable
- Cost Effective

Translation of Demands

The means by which mission-peculiar and resource demands are translated into finite requirements, and ultimately into mission-oriented

IFM plans, are discussed in subsequent sections of this paper.

IFM CONCEPTS

Prior to the development of a specific support capability, it is essential to define (in as precise terms as possible) the nature and structure of the mission to be performed, together with the factors and influences which impact on such a mission, and the hardware systems required for its performance.

Mission Categories

Each mission will not require an IFM capability of the same type and magnitude; therefore, to provide the required level of support, it is necessary to establish mission categories which possess a general commonality of IFM demands and required technology levels. The three (3) basic categories are:

- Earth-Orbital
- Earth-Lunar
- Interplanetary

Typical missions in each category are shown in Figure 6.

Mission Durations

To provide a reference point, the following mission durations have been established. The earth-orbital mission durations are in general agreement with current planning; however, those in the earth-lunar and

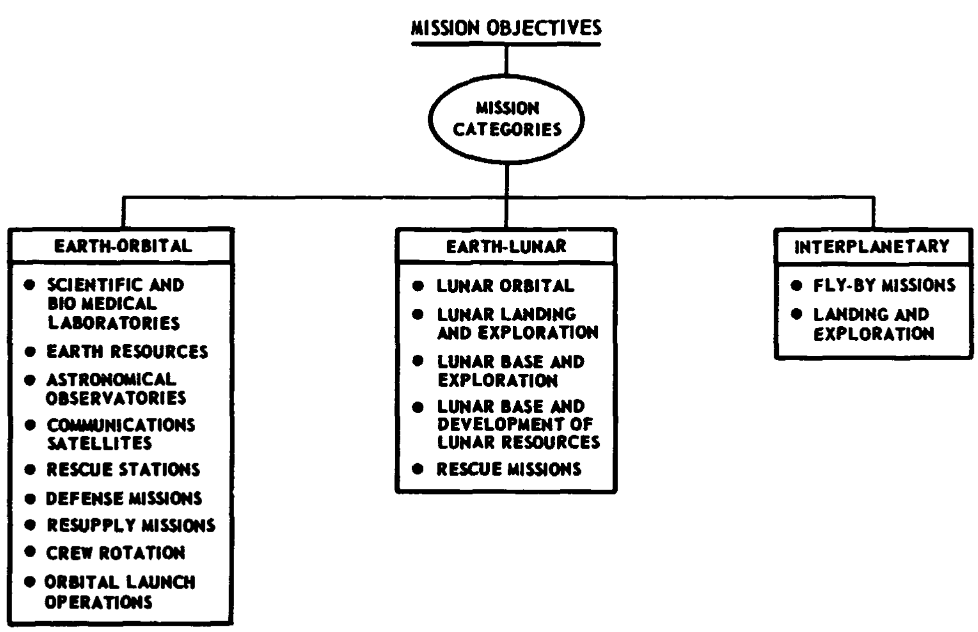

FIG. 6. - SUMMARY OF MISSION CATEGORIES

interplanetary categories are the author's approximations.

- Earth-Orbital
  - Short Term:
    - Up to 3 months
  - Extended Duration:
    - 3 months to 1 year
  - Long Term:
    - 1 to 5 years
- Earth-Lunar
  - Lunar-Orbital:
    - TBD
  - Lunar Landing and Exploration:
    - Up to 3 months
  - Lunar Base and Exploration:
    - 3 months to 3 years
- Interplanetary
  - Fly-By Mission:
    - 1 to 2 years
- Landing and Exploration:
  - TBD

Mission Effects Analysis

A myriad of decision factors have either a direct or indirect effect on IFM (see Figure 7), and must be analyzed to identify the nature and extent of their impact. A mission effects analysis is performed to make this determination. The results of this analysis provide input data to the more detailed process of developing a mission-oriented IFM concept shown in Figure 8. One of the most critical mission effects occurs in the development of an IFM concept for interplanetary fly-by and landing/exploration missions. The resupply of these missions is not deemed feasible, from either an earth or lunar source, once the spacecraft has been launched from earth or earth orbit.

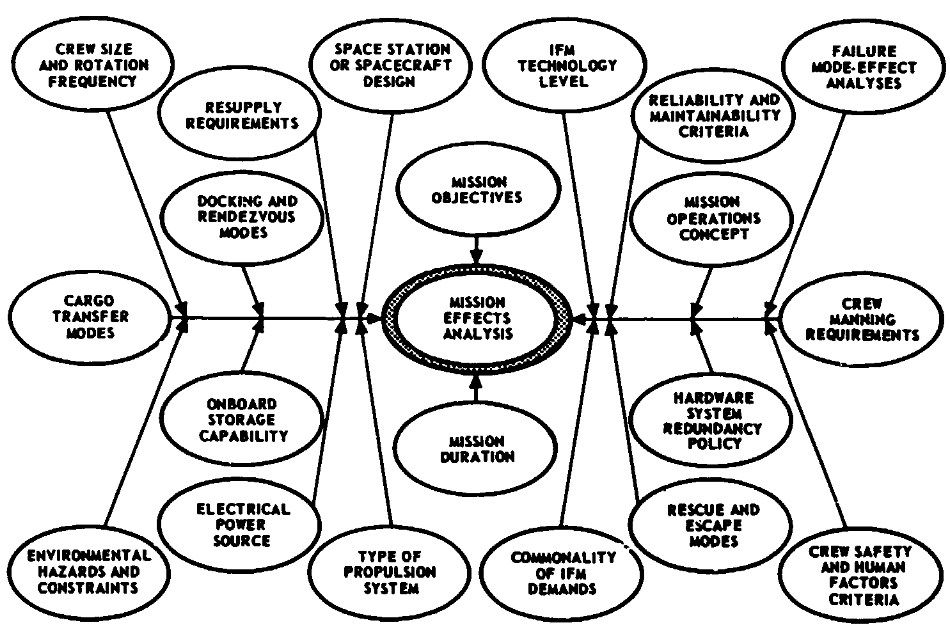

FIG. 7. - IFM CONCEPT DECISION FACTORS

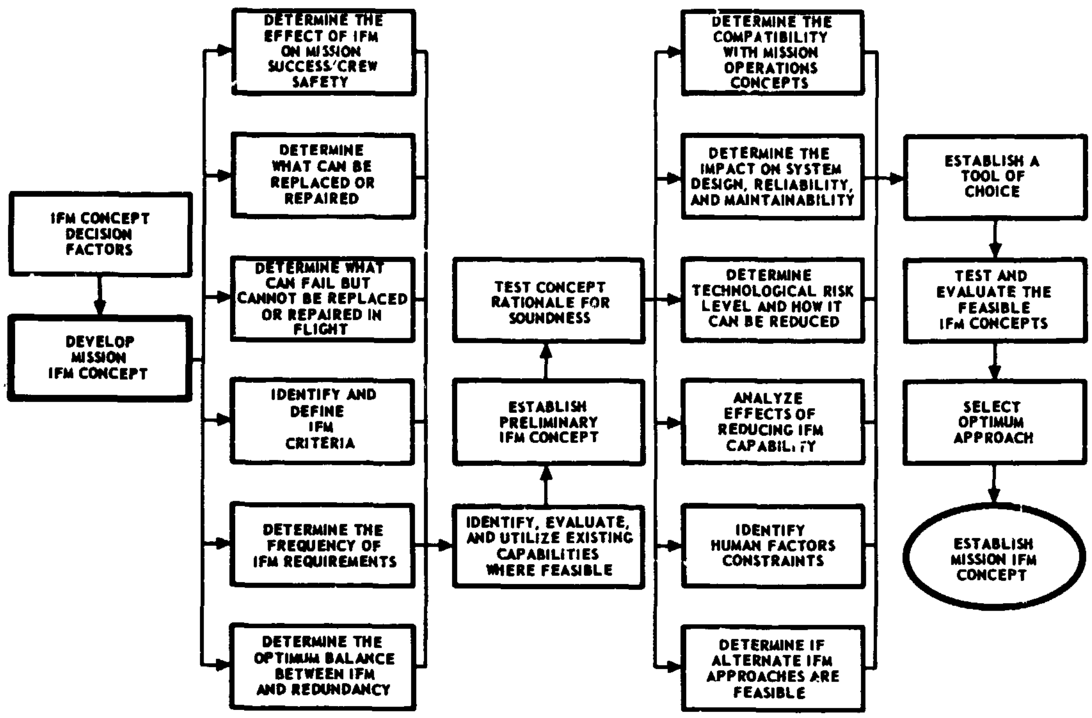

FIG. 8. - MISSION IFM CONCEPT DEVELOPMENT

Therefore, all support capability for these missions must be self-contained, and the planning and provisioning of 1- to 2- year missions, without resupply, will require the ultimate in effective logistics support.

Concept Rationale

The principal decision to be made in the development of an IFM concept is the relationship of IFM and hardware system redundancy, and the proportionate responsibility to be assigned to each of these forms of contingency protection. An optimum balance between these two (2) approaches is necessary to assure a high probability of mission success.

Neither IFM nor redundancy can do the job alone. For example, if a life support system failure occurs, it would be of little value to possess only an IFM capability. The operation of the life support system is an essential maintenance resource, and without it, the IFM capability would be reduced to nil. Conversely, nothing man has ever built is immune to failure and human error. Furthermore, when the redundant system fails, the remaining system is reduced to inherent reliability and should another failure occur (and Murphy's Law usually prevails), not only has the safety factor (provided by redundancy) been removed, but now a critical system is inoperative and mission failure is imminent.

IFM and redundancy are examples of the two (2) philosophical extremes mentioned in the introductory portion of this paper. Redundancy, a form of reliability, is at the optimistic end of the mission reliability spectrum, and IFM is at the opposite or pessimistic end. It is readily discernible that both capabilities are required, and the individual merits and limitations of each must be analyzed and weighed carefully to arrive at the balance point - realism.

The initial IFM capabilities will be relatively simple, and will probably be limited to the removal and replacement of certain modular equipment items identified, during a failure mode effect analysis, as being flight critical items. A certain degree of redundancy will be provided for these critical items; however, since redundancy is not an absolute guarantee of crew safety and mission success, some IFM capability must also be provided. A limited in-flight repair capability will be required to augment the removal and replacement capability. These initial repair tasks will employ repair kits containing bulk materials to patch and seal leaks in cabin pressurization, food and waste management areas, and to repair damage to electrical wiring.

The performance of these IFM tasks will provide an important means of enriching IFM technology with empirical data which will provide a more realistic baseline for extending IFM capabilities to more demanding applications. Extrapolation is more effective than pure prediction; therefore, we must investigate the possibility of using existing capabilities and resources "as is" or modifying them before we attempt to develop a totally new capability. For instance, we can draw useful parallels from "Polaris" type nuclear submarine missions. These "inner space" vehicles have much in common with space stations and spacecraft. Some of these similarities are:

- The requirement for an EC/LS system
- Environmental protection is required for EVA.
- Special tools are required for EVA.
- Missions are of extended duration.
- Electrical power requirements
- Limited storage area available for spares.

The principal differences between these two (2) types of vehicles, from a maintenance standpoint, is the presence of normal gravity in the submarine and the availability of maintenance specialists.

A basic IFM concept will be developed for each mission category. Each of these concepts will be modularly constructed to facilitate incorporation and blending of those elements of IFM required to meet specific mission demands within each category. This approach provides an IFM capability which is commensurate to the needs of a variety of missions within a particular category. Furthermore, all mission categories and missions within each category are analyzed to determine the type and degree of commonality of mission IFM demands.

IFM Criteria

It would not be technically or economically feasible to attempt to establish an IFM capability which would duplicate a terrestrial capability. Therefore an order of ranking must be established for all IFM actions. This ranking is based on the criticality of the vehicle system failure/damage as follows:

- Failure/damage which has an immediate or imminent effect on crew safety
- Failure/damage which affects vehicle performance and abort probability
- Failure/damage which would result in degraded mission performance
- Failure/damage which would result only in creating a nuisance

Only the first three orders of criticality should be considered in developing an IFM capability.

The major vehicle systems included in each order of ranking are listed below.

Crew Safety Impact

- Structure
- Electrical Power
- EC/LS
- Escape Capability
- Biomedical and Behavioral Monitoring

Vehicle Performance/Abort Probability

- Stabilization and Control
- Propulsion
- Navigation and Guidance
- Displays and Controls
- Communications

Degraded Mission Performance

- Instrumentation and Telemetry
- Experiments

The following criteria are recommended for the performance of IFM:

- IFM actions must make a significant contribution to: (1) improved reliability, (2) crew safety, and (3) mission success.
- Tasks must be simplified to permit their satisfactory accomplishment by crew members with limited maintenance training and experience.
- IFM data, procedures, and techniques must be developed to be compatible with both the IFM requirement and the maintenance skill levels of the crew.
- Spares and repair kits must be readily accessible to crew members.
- Requirements for special tools and test equipment must be minimized.
- Hardware systems should incorporate fail-safe features.
- All IFM procedures and techniques must be in accordance with crew safety criteria.
- All IFM procedures and techniques must satisfy human factors criteria.
- Hardware systems must be designed to facilitate rapid and positive detection, isolation and correction of in-flight failures and damage.
- IFM performance times must be equal or less than the maximum allowable downtime for hardware systems.
- All candidate IFM actions must be evaluated in terms of technical and economic feasibility.
- IFM will primarily consist of corrective maintenance actions. Where a requirement for preventive maintenance is identified, the preventive maintenance tasks should be performed in conjunction with normal mission operations functions.

IFM Modes

All IFM actions are performed by utilizing one (1) or more of the following four (4) IFM modes:

- Remove-Replace-Discard
- Repair Without Removal
- Remove-Replace-Repair
- Remove-Repair-Replace

These modes, individually and in combination, form the basis for structuring all IFM tasks. The application of these modes to the hardware systems generation breakdown is shown in Figure 9. The decision to use a particular mode has a definite effect on the requirements for specific maintenance resources (i.e., Spares, Repair Kits, IFM Procedures and Techniques, IFM Times, etc.).

IFM Technology

An IFM capability is a composite and complex structure which

HARDWARE GENERATION BREAKDOWN	REMOVE-REPLACE-DISCARD	REPAIR WITHOUT REMOVAL	REMOVE-REPLACE-REPAIR	REMOVE-REPAIR-REPLACE
SYSTEM(GENERAL)		X		
SYSTEM(SPECIFIC)		X		
SET		X		
GROUP		X	X	X
UNIT	X	X	X	X
ASSEMBLY	X		X	X
SUBASSEMBLY	X		X	X
PART	X		X	X

FIG. 9. - TYPICAL APPLICATION OF IFM MODES

incorporates information and criteria from many engineering disciplines. The impact of IFM on the following disciplines, together with their need to interact with each other, make it both necessary and advantageous to accumulate and integrate all areas of IFM information and experience into one formalized technology.

- System Safety
- Hardware Design
- Reliability
- Maintainability
- Human Factors
- Logistics

It is important that, once developed, the initial IFM capability be sufficiently flexible to permit its extension to more demanding applications in support of future manned spaceflight missions. The development of a technology relating specifically to IFM will provide a more realistic IFM baseline for new missions and programs by tempering theory and predictions with actual experience. A typical development flow for IFM technology is shown in Figure 10.

IFM REQUIREMENTS

IFM requirements are the specific needs for support resources based on the identification of IFM demands and the development of mission-peculiar IFM concepts. The requirements are identified and quantified as the result of performing an IFM requirements analysis. The outputs of this analysis form the basis for the development and allocation of resources, which are applied individually and in combination to produce an effective IFM capability. The large variety of manned spaceflight missions, together with

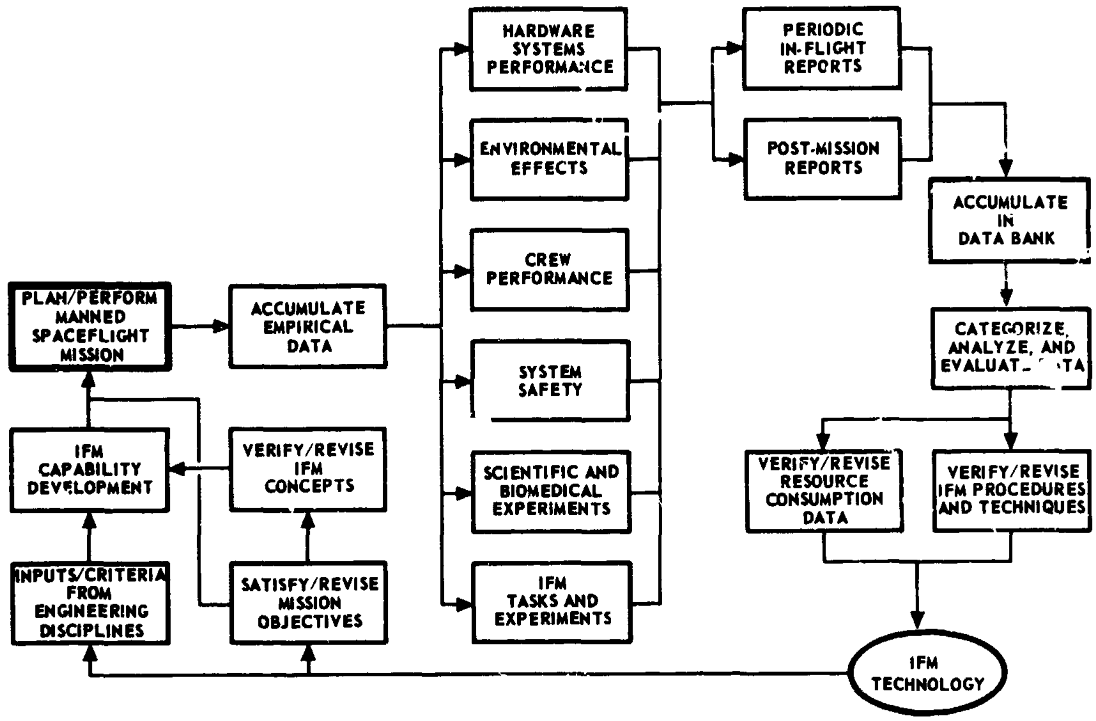

FIG. 10. - IFM TECHNOLOGY DEVELOPMENT

the many uncertainties of manned missions in the space environment, make the establishment and fulfillment of these requirements a very complex task. To further emphasize the magnitude of this effort, the primary outputs of the requirements analysis and the major elements of each are presented in subsequent discussions. The analytical pattern for determining these requirements is shown in Figure 11. The requirements determination starts with an analysis of vehicle systems that results in a detailed definition of IFM tasks, which were initially defined, in gross terms, during the concept development phase. This definition initiates the identification, determination, and development of a variety of resources presented in the following paragraphs.

IFM Modes and Environment

A decision is made with respect to utilization of one or more of the following modes, and the applicable environment is identified:

- Remove-Replace-Discard
- Repair Without Removal
- Remove-Replace-Repair
- Remove-Repair-Replace
- IVA
- EVA

Maintenance Category

There are two (2) basic categories of IFM - Preventive and Corrective. Although primary emphasis will be placed on corrective maintenance; where a requirement for preventive

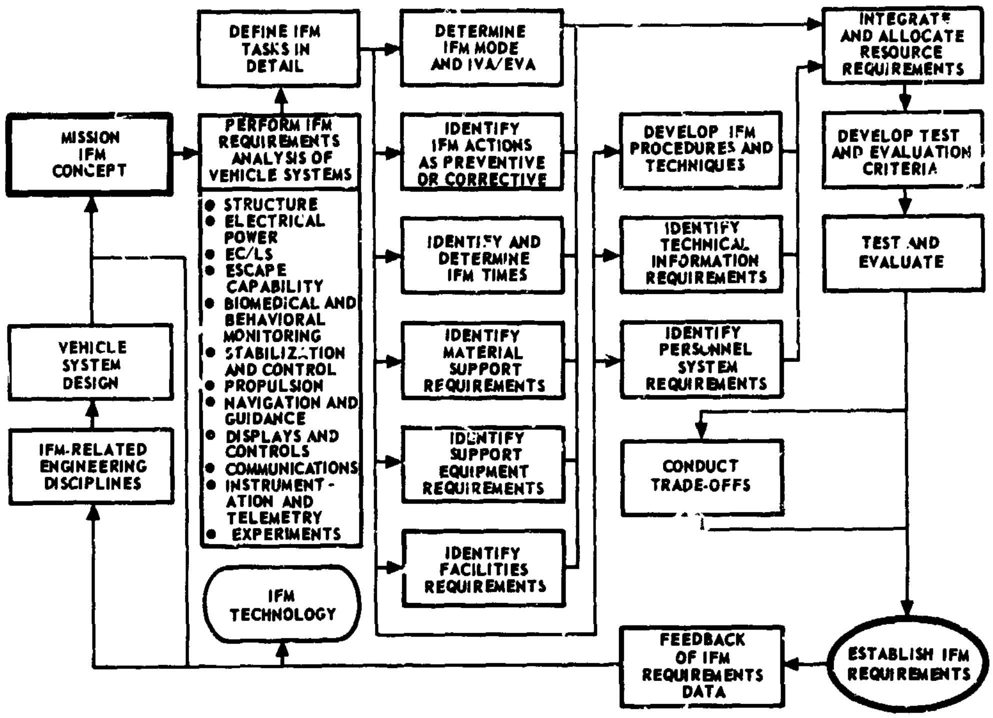

FIG. 11. - DETERMINATION OF IFM REQUIREMENTS

maintenance is identified, it will be integrated with routine mission tasks. Examples of typical preventive maintenance tasks include:

- Removal and replacement of limited life items
- Inspection
- Servicing
- Calibration
- Periodic vehicle systems GO-NO GO checks

IFM Time Considerations

Four types of IFM times are identified and determined.

- Elapsed time required to perform preventive or corrective maintenance tasks.
- Frequency of requirement for IFM tasks
- Crew time available for maintenance tasks
- Safe allowable time for maintenance tasks

Material Support

The following considerations are analyzed and resultant requirements are identified.

- Spares and repair kits/parts, consumable items, bulk items, EC/LS supplies, vehicle propellants, pressurants, and routine operating supplies
- Quantities of material support items required
- Frequency of requirements based on failure rates, damage probability, wearout, and shelf life in a space environment
- Weight, volume, and mass constraints

- Economic tradeoffs of replace/repair versus throw-away items
- Onboard storage capacity
- Effect of onboard storage capacity on resupply requirements
- Environmental sensitivity of support item
- Packaging protection
- Compatibility of packaging protection with existing packaging technology

Support Equipment

The following criteria has been established for support equipment requirements:

- Eliminate support equipment requirements, wherever feasible, by designing the function or capability into the vehicle systems equipment.
- Utilize existing equipment wherever technically feasible.
- Modify existing equipment wherever technically and economically feasible.
- Establish new equipment requirements only where required, and no capability presently exists.

Examples of support equipment requirements include:

- Cargo Transfer Equipment - Arms, booms, tethers, conveyors, intervehicle collapsible and extendible tunnels, pumps, fuel, propellant, pressurant disconnects, etc.
- EVA Maneuvering Equipment - Astronaut maneuvering units, hand held propulsion devices, jet shoes, tethers, etc.
- Life Support Equipment - Spacesuits, portable life support systems, radiation detection devices, etc.
- Mobile Power Units for EVA maintenance, orbital assembly and checkout of space stations and interplanetary spacecraft; strap-on propulsion devices for use in case of retrorocket failure, de-orbit of waste materials and space junk and for emergency maneuvering
- IFM Tools - The requirements include: zero-reaction tools, multipurpose tools, welders, minimum torque and internal wrenching tools, and tools for use with nuclear devices
- Maintenance Aids and Devices - IFM requirements include: personnel and equipment restraints, portable lighting equipment, visual reference indices, portable radiation shields for use while performing IFM on nuclear devices, and electrical recharge devices for IFM tool power packs
- Handling Equipment - The requirements are limited to that equipment necessary for holding or maneuvering supplies and vehicle system equipment.
- Special Purpose Vehicles - Maintenance shuttles and orbital tugs
- Diagnostic, Checkout and Monitoring Equipment - Requirements for this equipment are analyzed to determine: (1) optimum mode of operation (automated, semi-automated or manual), (2) system level to which equipment should function, (3) types of audio and visual signals and displays required, and (4) should this equipment be incorporated into vehicle systems hardware.
- Calibration Standards - Requirements for standards are analyzed to determine: (1) types required, (2) transportability, (3) environmental protection requirement, and (4) certification requirements
- Orbital Repair/Escape Decision System - This system provides a rapid means of determining (in gross terms) if sufficient time is available to repair a malfunction in a critical system or should the crew evacuate to another part of the space station or evacuate the

space station to the escape module (logistics spacecraft docked at the station). The system would utilize sensors for heat, pressure, atmospheric contamination and nuclear build-up, and would operate as a function of rate of change or threshold. Programming will be developed to provide a repair or escape decision based on the severity of the input from the sensors, and would terminate in audio signals and flashing visual displays.

Facilities

The facilities requirements deal principally with two (2) considerations:

- Onboard storage capability for material support and support equipment items
- Areas for performance of IFM repair tasks

Procedures and Techniques

IFM tasks are performed by following an action sequence which generally consists of three (3) steps for preventive maintenance and five (5) for corrective maintenance. Procedures and techniques are required for the following sequential actions:

- Detect and Isolate Failure
- Determine Maintenance Action
- Prepare for Maintenance (Preventive and Corrective)
- Perform Maintenance (Preventive and Corrective)
- Perform Post-Maintenance Checkout (Preventive and Corrective)

A typical IFM cycle is shown in Figure 12.

It is recognized that maintenance functions are more critical in a space environment in terms of tolerable system downtimes, availability of crew members for maintenance tasks, the ability to perform these tasks in a weightless condition, and in some instances while wearing a spacesuit. However, as in the case of aircraft maintenance in arctic areas, and that of undersea maintenance operations, these obstacles can be overcome through the development of procedures and techniques, together with training and experience.

Procedures and techniques will be developed to permit crewmembers to perform the necessary IFM tasks. The means of implementation include the following:

- Fault detection and isolation
- Recognition of hazards and hazardous conditions
- Use of tools, maintenance aids, and other support equipment
- Control of tools and spares/repair kits during maintenance actions
- Economy of motion and energy while performing maintenance tasks
- Motion and force translation
- Removal and replacement actions
- Repair actions
- Post-maintenance verification of vehicle systems status

A variety of maintenance action functions are involved in the performance of IFM. Over 100 of these potential functions are shown in Table 1. Initially, both the maintenance tasks and the required procedures and techniques will be relatively simple. However, as we proceed from earth-orbital missions to earth-lunar missions, and ultimately to the interplanetary category, the mission durations will increase and so will the complexity of vehicle systems equipment. This trend to longer missions will place more exacting demands on the time and skills of the crew members to achieve and maintain a high probability of mission success. In the case of interplanetary missions, the point-of-no-return or abort limit

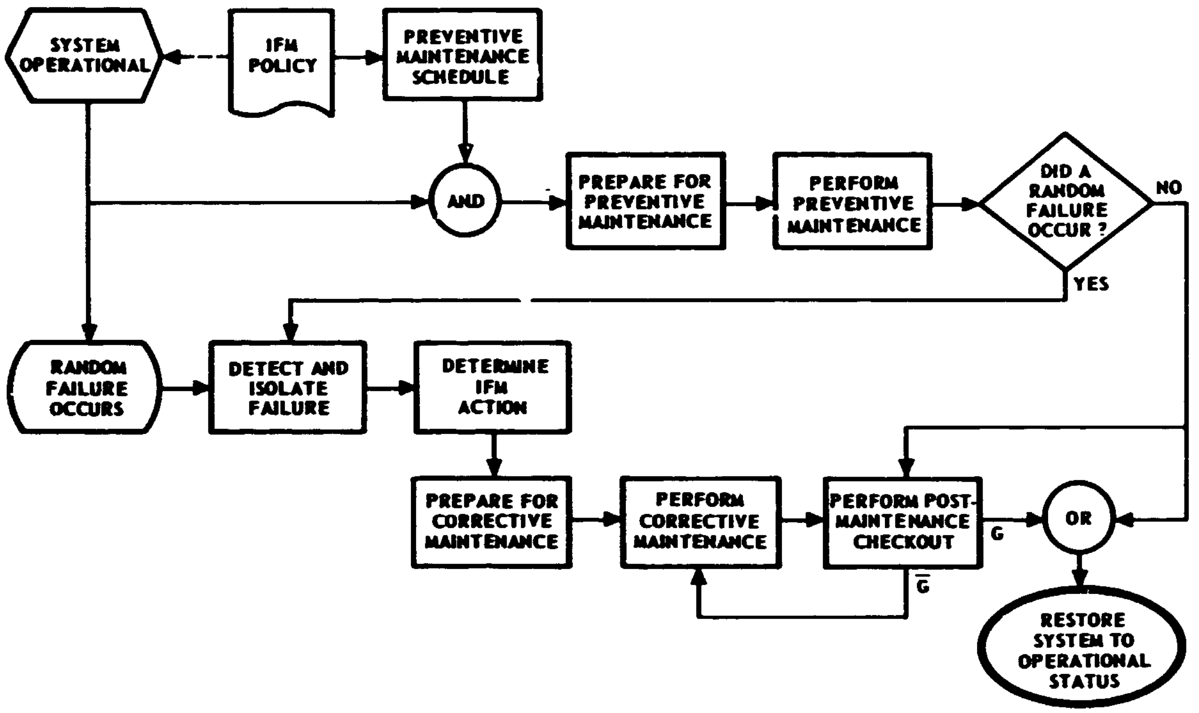

FIG. 12. - IFM ACTION SEQUENCE

will be only a short period of time after departing earth-orbit. Therefore, since no resupply capability will be feasible for these missions, after leaving earth-orbit, all support capabilities will, of necessity, be self-contained. This situation creates a clear-cut requirement to develop and implement IFM procedures and techniques which maximize the crew member's IFM capability and minimize his limitations. This subject is discussed further in the Personnel Systems paragraph.

Technical Information

The volumetric limitations of a space station or spacecraft combined with a weightless environment, make it necessary to depart from the conventional terrestrial formats of technical information for support of manned space flight missions. A primary and secondary in-flight system for providing technical information are recommended. The primary system will consist of a micro-viewer supported by an onboard file of microfilm chips or tape. The secondary system will consist of television-audio equipment. Both systems would provide in-flight operations and maintenance procedures augmented by semiconventional checklists for operations, maintenance, and emergency procedures.

The following factors are considered in the development of an in-flight technical information system capability:

- Type of in-flight information required (e.g., Procedures, Schematics, Reference Tables, etc.)

ACTUATE	DEVISE	HANG	ORIENT	REMOVE	STOW
ADJUST	DISASSEMBLE	HOLD	OVERHAUL	RENOVATE	STRIP
ALIGN	DISCONNECT	INITIATE	PACKAGE	REPAIR	SYNCHRONIZE
APPLY	DISMANTLE	INJECT	PATCH	REPLACE	TAG
ASSEMBLE	DISPOSE	INSPECT	PIERCE	REPLENISH	TEST
ATTACH	DRILL	INSTALL	PLACE	RESTORE	THREAD
BOLT	ENGAGE	ISOLATE	POSITION	REWORK	TIGHTEN
CALIBRATE	ENTER	JOIN	POUND	RIVET	TORQUE
CHANGE	ERECT	LEVEL	PREPARE	ROTATE	TRANSPORT
CHECK	EXAMINE	LOCATE	PRESET	SALVAGE	TRIM
CLAMP	FABRICATE	LOOSEN	PROBE	SEAL	TROUBLESHOOT
CLEAN	FASTEN	LUBRICATE	PROCESS	SCREW	TUNE
CONNECT	FILE	MATE	PUNCTURE	SECURE	TURN
CONSTRUCT	FORM	MEASURE	PUSH	SERVICE	TWIST
CONTROL	GLUE	MODIFY	PULL	SET	UNBOLT
COVER	GRASP	MOUNT	REASSEMBLE	SETUP	UNCOVER
CUT	GUIDE	MOVE	RECONDITION	SOLDER	WELD
DETACH	HAMMER	NEUTRALIZE	RECORD	STIMULATE	ZERO
DESTROY	HANDLE	OPERATE	REGULATE	STORE	

TABLE 1. TYPICAL MAINTENANCE ACTION FUNCTIONS

- Type of display format
- Type of primary system equipment
- Automation requirement for primary system
- Functional independence of primary system (with the exception of updating)
- Means by which primary system can be kept current
- Source of in-flight technical information
- Data size reduction techniques required and available
- Feasibility of using conventional checklists
- Type of secondary or backup system required
- Method of cataloging, storing, and retrieving information
- EVA information requirements
- EVA information system equipment requirements
- Support requirements for primary and secondary system

The constant change in all technologies made it essential that the technical information systems be developed with a built-in flexibility to permit rapid and efficient response to changes in missions and vehicle systems equipment.

Personnel Systems

Personnel systems have been called "the configuration management of people," and it is through these systems that man is analyzed as a resource, his capabilities are defined and utilized, and his limitations are identified and compensated for, either by training or the substitution of automated machine functions. The introduction of man into the loop of space flight operations provides a decision making capability with greater flexibility than that possessed by inanimate systems, coupled with the ability to relate decisions to mission success.

After mission objectives are defined and an optimal sequence of events is established, a task/equipment analysis (TEA) of the operational requirements, system functions, performance standards, and operations and maintenance concepts is conducted. This analysis defines the man/machine interface, and is the basis for assigning each task to either man or systems equipment, depending on which possesses the greater performance capability and practicality. Those automated functions, which affect crew safety, will be further analyzed to determine the feasibility of incorporating a manual override feature and complete independence from ground control.

Once the operations and maintenance tasks have been defined, they are time-oriented in terms of sequence, duration, and frequency by performing a time line analysis. This analysis results in the identification of problem areas in time-phasing and allocation for operations and maintenance functions. These problem areas are subjected to further analysis, and necessary tradeoffs are made.

Examples of the factors considered during these analyses are:

- What tasks must be performed and why?
- What are man's abilities to perform mission tasks and how are they constrained with respect to his role in the space environment?
- What are the performance standards, and how are they measured?
- What are the man/machine relationships, and how do they relate to mission success?
- What are the operations and maintenance concepts, and what influences are considered during their development?
- What are the requirements for assigning crew members multi-task responsibilities?

A detailed analysis of personnel requirements (e.g., Qualitative and Quantitative Personnel Requirements Information-QQPRI) is performed which identifies the types, levels, quantities, and performance standards for skills which are necessary to accomplish the tasks defined in the TEA. In addition to skill information, the personnel requirements analysis provides a basis for manning recommendations for user operations and support organizations. These recommendations include need dates, user locations, work loads, and work shifts.

Effective job performance in a weightless environment for extended periods of time depends on a crew member possessing the following capabilities, characteristics, and tolerances:

- Technical capability in his duty assignment
- Motor capabilities and anthropometric characteristics which are compatible with the operating environment.
- Physiological and psychological characteristics and tolerances which are compatible with the mission requirements and the operating environment.

In-flight operating techniques are developed to enable crewmembers to perform their assigned duties effectively. The capabilities, listed above, apply to both IVA and EVA; however, EVA imposes the following additional constraints on job performance:

- Reduced motor capabilities and greater susceptibility to fatigue caused by wearing a space suit
- Degradation of visual reference
- Criticality of illumination
- Reduced accessibility of equipment while wearing a space suit
- Increased radiation hazard

Training requirements are determined by comparing existing skills with those required by the mission tasks defined by the TEA. After the requirements are established, it is necessary to develop a means by which they can be satisfied. The personnel training function is analyzed and evaluated so that an equitable approach can be established. Examples of the factors considered during these analyses are:

- The prerequisites for crew members, and the means by which they are determined (e.g., present skills, educational background, physical condition, age, emotional stability, character, leadership qualities, etc.)
- The personnel sources which are available (e.g., military and naval services, industry, colleges and universities, etc.).
- Means by which the personnel sources can be expanded (e.g., new curriculum for high schools, colleges and universities, etc.).
- The estimated rate of attrition for mission operations.
- Determination of screening methods which should be established for the selection of crew members.
- The training evaluation methods and procedures which are required.
- Training course requirements (e.g., basic courses for new job classification, crew training, proficiency retention, upgrading, in-flight operating techniques for crew members, etc.).
- Training aids and equipment requirements (e.g., dynamic and static mockups, simulators, films, etc.).
- The effect the total training effort has on IFM, human factors, vehicle design, reliability, maintainability, and systems safety.

The personnel systems must be tested and evaluated on the same basis as any other resource. The introduction of man into the loop of space missions provides many distinct and unique advantages; however, in evaluating his effectiveness with respect to IFM, man must only be considered as a resource. The margin by which man's abilities outweigh his limitations will be the measure of his contribution to mission success.

Integration and Allocation of Resources

After the determination, identification, and development of IFM resources has been analyzed, it is necessary to integrate and allocate the required resources. The methodology by which this action takes place requires the use of several analytical tools which include:

- Simulation Models
- Cost Models
- Systems Analysis Techniques
- Judgement in Recommendations

In recent years, there has been an increasing tendency to oversophisticate analytical techniques. This is a natural occurrence due to the increasing complexity of vehicle systems; however, like any tool, each has its capabilities and limitations which must govern its use. Some systems analysts seem to have overlooked the fact that while computer program models provide a rapid means of handling large volumes of data, a thorough understanding of the problem by the analyst, coupled with the ability to define input requirements in precise terms, are essential in producing a valid and effective output.

Little IFM experience in an actual space environment is available; therefore, it is necessary to synthesize or simulate "real world" conditions. Systems synthesis is both an art and a science. The art is selecting the combinations to be studied, and establishing the criteria by which final system decisions can be made. The science lies in demonstrating, by analysis, the physical realizability and detailed characteristics of selected combinations of subsystems, and in this case, IFM resources. In this analytical process, the tool of judgement in recommendations should receive the most weight.

The prime purpose of this paper is to recommerd an approach to achieving an IFM capability which is realistic, effective, and economically feasible. Broadly, the elements involved in this analysis include system restraints, decision factors, ranking, evaluating, grading, and selecting. All of these factors are analyzed statically, as shown in Figure 13, and then dynamically with respect to customer goals and cost effectiveness to answer such questions as:

- What should an IFM capability consist of?
- When should it be initiated?
- What should the schedule be?
- What funding level is suitable?
- What development and testing policy should be followed in order to meet IFM requirements at the least cost?

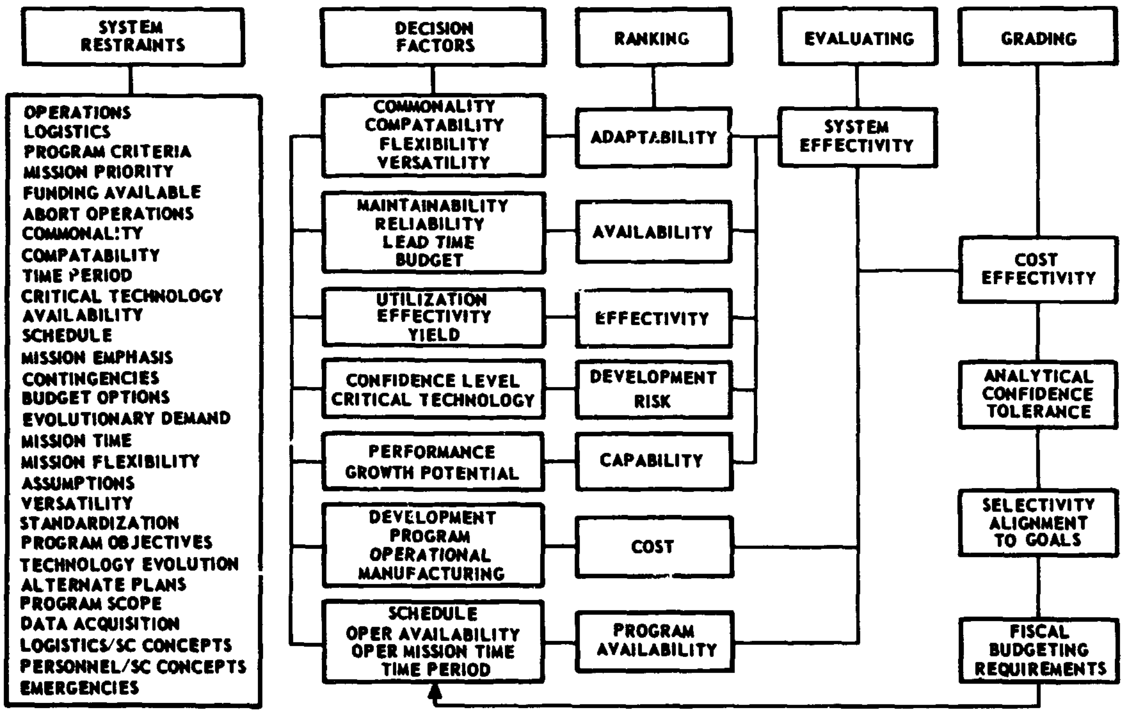

FIG. 13. - SYSTEM EFFECTIVITY - UTILITY

- What equipments should be developed or improved, and in what sequence to obtain the greatest capability in the least possible time?

The final analytical step in the development of an effective IFM capability is the definition of an analytical confidence tolerance. This tolerance will determine the sensitivities of IFM so that any selectivity may be confidently aligned with space program objectives. It is apparent that if the approach, presented herein, were considered unity or the total sum of related parts, the primary resolve of this analysis is to determine: (1) the cost of less than unity, and (2) the savings that would result from such a reduction. Since an integrated IFM capability will probably remain on an experimental basis for some time to come, it is necessary to determine. the minimum degree of IFM which is satisfactory, the least path development requirements to make the transition from experimental to operational status, and to what degree unity is required.

POLICIES AND PLANS

The final phase in the acquisition of an IFM capability consists of establishing the necessary IFM policies that will govern the performance of all IFM actions, and the development and preparation of plans which provide for policy implementation.

IFM Policies

To achieve a logical progression towards a common objective - mission success, it is necessary to standardize performance methods, and thereby create a baseline for measuring the effectiveness of the approach being taken. The establishment of IFM policies, which are technically and economically feasible, produce the following results:

- Uniformity in IFM actions which permit the accumulation of meaningful empirical data.
- System downtime will be minimized by providing crew members with predetermined maintenance modes for system failures or damage.
- Improves cost effectiveness
- Minimizes expenditure of resources
- Maintains crew safety
- Provides the most effective utilization of the crewmember in his role as a maintenance man.
- Minimizes human error during the performance of IFM tasks
- Provides an effective interface with IFM - related technologies and disciplines
- Maximizes the contribution of IFM to mission success

IFM Plans

There are three (3) evolutionary phases through which a new capability must pass before becoming a reality. To assure an orderly progression, a plan is prepared for each of the following phases:

- Development
- Test and Evaluation
- Implementation

Each plan should contribute to the development of an overall IFM technology and an integrated IFM capability. They should be sufficiently flexible to respond to changes in mission requirements and vehicle system hardware without major perturbations. Furthermore, IFM plans must always be directed towards improving vehicle system reliability and the probability of mission success. A summary of plan objectives is shown in Figure 14.

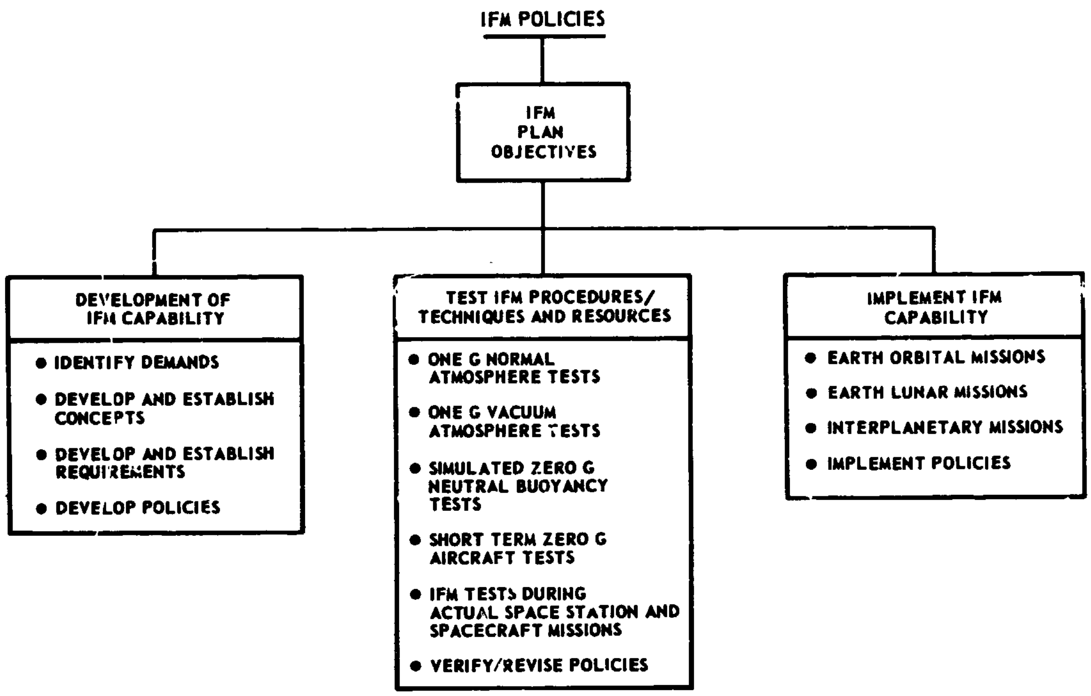

FIG. 14. - A SUMMARY OF IFM PLAN OBJECTIVES

Finally, it is important to remember that although system redundancy and a high degree of reliability will be incorporated in space stations and spacecraft, human error and equipment failures will occur. We have little experience in the actual space environment; yet, an IFM capability must be developed which will be responsive to mission needs in an environment more hostile to man and equipment than has been previously experienced. IFM planning poses a tremendous challenge to the mission support planners of industry, NASA, and the Department of Defense. The effectiveness with which this challenge is met will, to a large degree, determine the feasibility of long term manned spaceflight.

CONCLUSIONS

Perhaps the most compelling reason to develop an IFM capability is the historical fact that a constant demand for vehicle maintenance has existed for well over 2, 000 years, and it is not logical to assume that we are now immune from human error and equipment failures. The goals that have been set in the national space program can be met only by achieving a high incidence of mission success. Whenever error, failures, or damage occur (and they will) during manned spaceflight missions, some form of onboard contingency protection must be available. The development and implementation of an IFM capability appears to be the most realistic answer to this problem.

In summary, the following conclusions are presented:

- An IFM capability is required for missions of extended or long duration to bridge the gap between achieved and required system reliability.
- The development of an IFM capability is both technically and economically feasible.
- Manned spaceflight missions are divisible into three (3) categories: (1) Earth Orbital, (2) Earth Lunar, and (3) Interplanetary.
- Mission-oriented IFM concepts should be developed for missions in each category.
- An IFM capability will be developed as an evolutionary process.
- The planned interaction of the design, human factors, reliability, systems safety, logistics, and maintainability disciplines are essential to an effective IFM capability.
- An IFM technology must be developed and formalized.
- An IFM testing program conducted during actual spaceflight missions must be incorporated in mission planning.
- Future vehicle system hardware must be designed for IFM.
- IFM is just one element of an overall Space-Terrestrial Logistics System.
- Enrichment of IFM technology will provide an empirical data baseline for a number of other crew-related tasks and activities (i.e., (1) Orbital assembly and checkout of space stations, interplanetary spacecraft, and astronomical observatories; (2) Orbital and lunar fabrication procedures and techniques; (3) Lunar assembly of structures; and (4) Rescue missions).

The actual development and implementation of an integrated IFM capability, such as the one described in this paper, will be a difficult task. It will not happen all at once; instead, it will be a progressive effort which will undergo many iterations due to advances in technology, changes in mission objectives, and other technical, political, and economic considerations. Therefore, to maintain a continuity of effort, it is necessary to develop a framework or blueprint from which a support capability for earth orbital, lunar, and ultimately, interplanetary missions can be constructed. The approach to IFM presented in this paper is an initial step towards achieving that goal.

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CREDITS

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MAN'S CHANGING ROLE IN EVA SPACE

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SUMMARY: The results of EVA on the Gemini program were spectacular "firsts" demonstrating man could leave the spacecraft, survive with no apparent difficulty, and move between various points. However, toward achieving the dedicated role of space mechanic, orbital worker, etc., the results were quite discouraging. Man became tired very rapidly and literally exhausted after 1-2 hours work indicating that he may not be able to achieve the capability we originally thought. Man's role will drastically change over the next 10 years. We will not expect the astronaut to construct antennas, telescopes, spacecraft structures, etc. from "nuts and bolts". Instead, he will assemble only major components, retrieve film and experiments, maintain critical subsystems, etc. This paper will describe the results of an analysis of a typical 100 foot diameter antenna which will be assembled in a one G environment to show the task difficulty.

INTRODUCTION

Although during the last few years the spacecraft being launched are becoming larger and larger, it is still impossible to launch space stations, large antennas, telescopes, etc. in the completely assembled position. Based upon some of the future requirements, it appears that antennas may be 100-200 feet in diameter. During the next decade, the diameter of space vehicles will be about 10-20 feet in diameter and therefore require these large structures to be folded or be assembled in space. Looking at antennas in particular, there are three basic methods being analyzed to package and launch the large antennas into the smaller spacecraft. These vary from inflatables, to foldable trusses, to manual assembled modules. This paper will concentrate on the module approach to show man's changing role in EVA space.

During the past ten years of space exploration, man has been envisioned as a space mechanic, orbital worker, and handyman repairman in addition to the basic functions as pilot. It appeared that man, if properly trained and equipped, could do most jobs better than mechanical systems. He possessed high intelligence, both physiological and psychological talents, and versatility to undertake and complete a variety of tasks in space. In pursuit of this objective, many extravehicular development programs were undertaken including the Hand-Held Maneuvering Unit, Astronaut Maneuvering Unit, Space

Power Tools, EVA Space Suits, etc. Most of these development efforts culminated with space flights on Gemini to demonstrate the astronaut's new capability.

The Gemini Experiments first encouraged EVA with the success of Colonel White, however, later flights have severely dampened the enthusiasm for EVA. The published results (Ref 1) of the Gemini EVA program states "EVA should be considered for future missions where a specific need exists, and where the activity cannot be accomplished by any other practical means."

The intent of this paper therefore is to show that a specific need does exist, and that if the EVA task is properly planned, tested, and the astronaut trained, it can be an effective tool.

ANTENNA DESIGN

The modular antenna concept appears to be most attractive to an EVA assembly procedure. This is based upon the idea that the modular honeycomb panels are assembled like a prefabricated home. It requires alignment and fastening of similar size panels; therefore, requires a minimum of tools, skills, and time which is already at a premium.

Many antenna studies already have been conducted to date, each using a different size panel as being the basis for analysis and comparison. For instance, the LTV Study on EVA Effectiveness (Ref 3) utilized a 5' x 8' size panel. Obviously the size of panel is very important as it determines the total number of panels and therefore affects the total assembly time. In order to establish a

time. In order to establish a
firmer basis for the panel size selection, the results of a USAF Study (Ref 2) with General Dynamics/Convair are summarized as follows:

Fig 1 - Underwater Assembly at Convair

The mass handling part of the study handled masses considerably heavier than is contemplated for this paper study. The panel size determination, however, is quite applicable to this assembly analysis. Actually, the optimum panel size was never derived since the three panels chosen were handled relatively easy by an astronaut. The panels were chosen arbitrarily to span those sizes which may be applied to modular antennas of the near future. The sizes are as follows:

Each panel was constructed with a wood frame and partially covered by mesh cloth to provide constant drag on the panel no matter which way it was turned in the water. The experiments comprised a test subject (suited) who reached the panel nearby turned it 90° and elevated it 90° to fit into the pie shaped holding bracket. The astronaut was restrained by the Gemini "Dutch" shoes during the task. The fasteners (not part of study) were simply a dowel pin and hole. The results of this study showed the average time to acquire the panel, rotate it, and install it as follows:

Panel (Approx)	Area (ft)	Time (Avg)
5x5	27.5	.54 min
6x8	54.0	.65 min
7.5x10	85.0	.80 min

These experimental results are the basis for establishing a panel size for this paper. The intent was to divide a 150 foot diameter antenna into large, equal size sections so the number of panels to be assembled were held to a minimum. Initially, a panel 10 ft x 10 ft was chosen to fill the needs of the antenna in Fig 2.

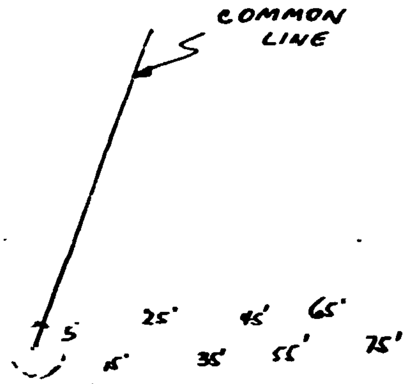
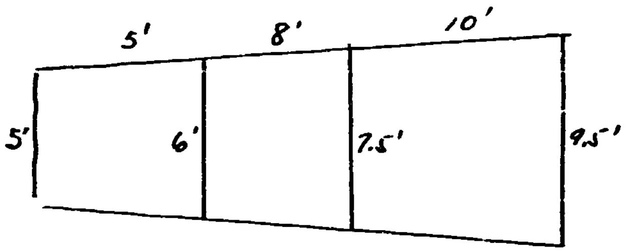


Fig 2 - Antenna Design (Initial)

The detailed data generated from this design is as follows:

Location	Circumference (Ft)	Panels (Nr)	Error (Ft)
5	31.4	3.14	.14
15	94.2	9.42	.42
25	157	15.7	.70
35	220	22.0	0.
45	283	28.3	.30
55	345	34.5	.50
65	408	40.8	.80
75	471	47.1	.10
	TOTAL	200.56	

After many trails and errors, it appeared a slightly smaller panel was optimum. This was based on the fact that each row ended in a fraction of a panel remaining. So to make each panel the same, and each row to come out even, a slightly smaller panel was studied and shown in Fig 3.



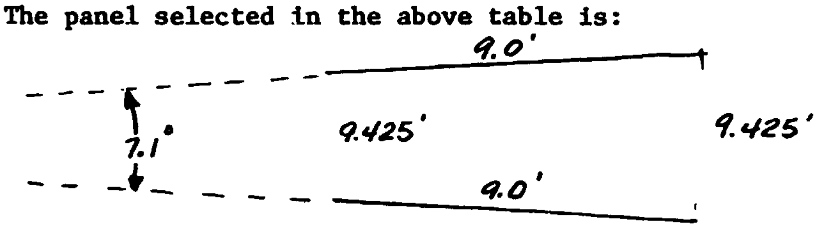
PANEL SIZES USED
IN CONVAIR TESTS

13.5 31.5 49.5 67.5
4.5 22.5 40.5 58.5 76.5
9 PAN
15 PAN
21 PAN
27 PAN
33 PAN
39 PAN
45 PAN
51 PAN

Fig 3 - Final Antenna Design

Row	Diameter (ft)	Circum (ft^2)	Nr of Panels	Length of Panels	Error ± (ft)
1	9.0	28.274	3.0	28.275	+ .001
2	27.0	84.823	9.0	84.825	+ .002
3	45.0	141.372	15.0	141.375	+ .003
4	63.0	197.921	21.0	197.925	+ .004
5	81.0	254.470	27.0	254.475	+ .005
6	99.0	311.02	33.0	311.02	0.
7	117.0	367.567	39.0	367.575	+ .008
8	135.0	424.116	45.0	424.125	+ .009
9	153.0	480.665	51.0	480.675	+ .010

The panel selected in the above table is:



It is pointed out that the actual number of panels would be 240 since the three (pie shaped) panels would remain as one solid piece. It is 9 foot diameter and would fit in most spacecraft as a single item.

PANEL SELECTION

The panels selected are slightly pie shaped as the angle would be 1/51 x 360 or about 7.1 degrees taper. The top and bottom of each panel will have a slightly different curvature depending on the location on the on the antenna. Therefore, it will probably be necessary to color code them for easy assembly by the astronaut.

The panels will be two (2) inches thick (Fig 4) and be constructed of either aluminum or paper honeycomb with about .005 inch thick mylar or aluminum surface. Each panel will have metallic edges to facilitate locking into place.

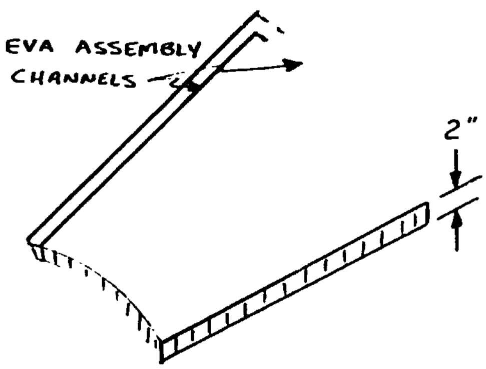


Fig 4 - Selected Panel

ASSEMBLY METHODOLOGY

The actual assembly and construction of large space structures should be based upon fact and experience rather than optimism. This is necessary since the structure by its very nature will be expensive and mission dependent. In other words, commitment to undertake the development of a large antenna structure will lead to a functional operating antenna that can be completed once the assembly is completed. Therefore, assembly will become a requirement for mission success leaving behind any desire to assemble "to learn what man can do or not do". Hence, the assembly technology must be highly reliable, and therefore simple.

To keep a system simple and reliable, especially an EVA method, it will be necessary to design very simple fasteners and keep the number of components to a minimum. The antenna described in the previous paragraphs will involve 240 trapezoidal panels plus a 9 foot diameter base section. If all panels could be exactly the same size and shape, it would simplify the assembly task. However, this is not possible as each layer or circular rows will have a different radius of curvature requiring specific assembly procedures. However, each row in this study does have identical panels requiring only that all the panels be brought to a given ring but the panels in that row require no specific order of assembly.

To further simplify the assembly process, no attempt will be made to utilize maneuvering and propulsion devices, portable life support systems or power tools. This is not to minimize the potential or value of these systems as they do have their place. However, the method to be discussed here is a different approach than some of those previously analyzed.

The much awaited antenna (Fig 5) is shown in the assembled condition first.

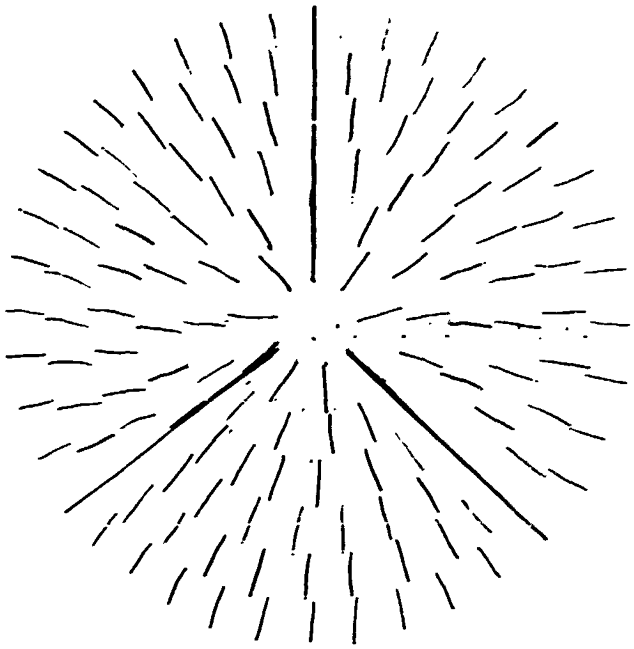


Fig 5 - Assembled Antenna

One important feature of this antenna is that one seam is continuous from the 9 foot radius to the outer rim at 76.5 feet. This will be a key feature in the assembly process. All other joints will follow an irregular pattern whether another single seam appears is unimportant.

Built into this radial seam and each concentric seam will be a lightweight channel track as shown in Fig 6.

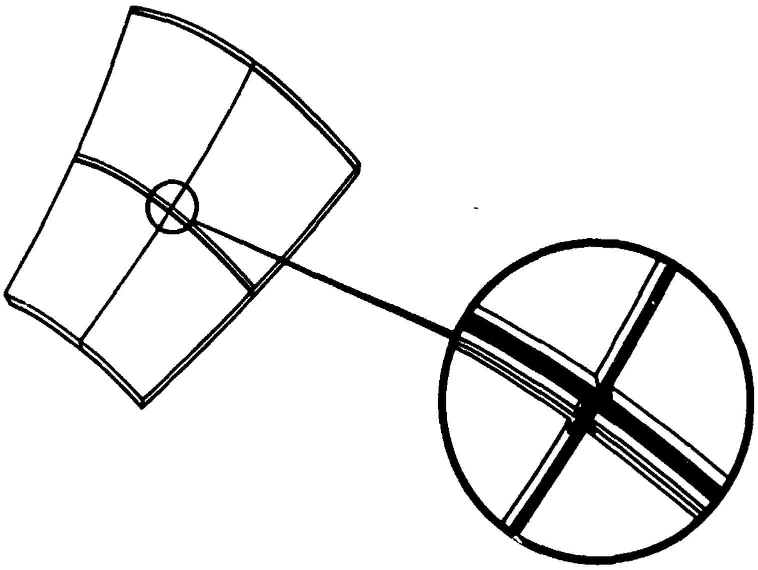


FIGURE 6-SEAM DESIGN

In addition to this track system, it will be necessary to build a very lightweight work platform as shown in Fig 7. This scafford is a platform to hold the NASA "Dutch" shoes, a light-weight guard for safety of the astronaut, wheels which hook into the track system, and a sponge rubber like support and drive wheel.

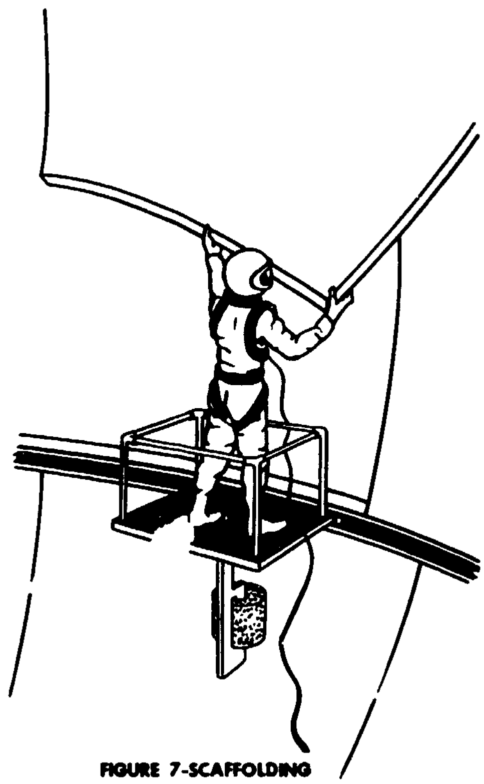


FIGURE 7-SCAFFOLDING

Fig 7 - SCAFFORD

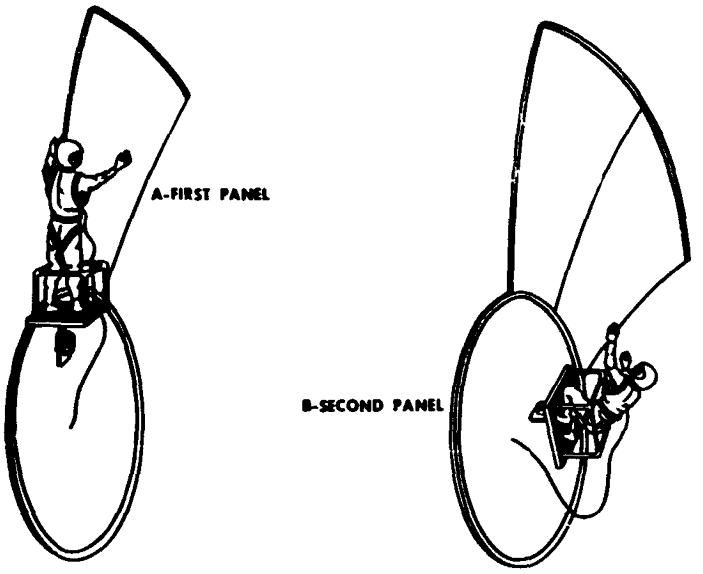


FIGURE 8-ASSEMBLY PROCESS

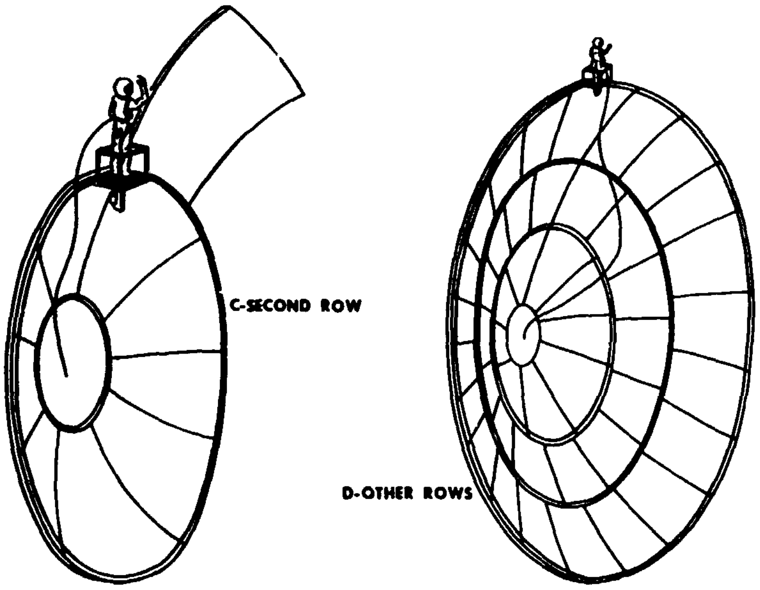


FIGURE 8-ASSEMBLY PROCESS

The detailed assembly method is shown in Fig 8. In Fig 8(A), the astronaut attaches himself to the 4.5 foot radius track system and begins assembly by installing the first panel in place. Each panel will probably use guide pins and a locking fastener to hold it in place. He moves the station for panel two and installs it. He continues as in Fig 8(B) around until the 9 panels are installed. He then moves vertically (or radially) to the 13.5 foot radius and assembles the next 15 panels (Fig 8(C) and so forth until all 240 panels are in place. All assembly will proceed from the backside of the antenna to protect the mirror surface and reduce the burn hazard to the astronaut.

There are two important features in this assembly process which has not been discussed so far. One, how do we support the assembler with ECS; and two, how do the panels get to the circumferential rows? The environmental control and life support is simpliest to answer. The astronaut would use an umbilical with the source at the pivot point of the antenna. This would allow 360° rotation on the circumferential rings without being entangled. The length of the umbilical would be only about 80 feet long which is not unreasonable since the Gemini program used 50 feet without too much difficulty.

The method for raising and lowering the panels to the appropriate ring appears relatively simple. Since the panels are weightless, we only have to worry about the inertias. This will be controlled or minimized by "raising" a stack of panels at low velocities. It is then only necessary to guide the stack in the proper direction by ropes either tied to the work scaffold, or by a simple bracket hooked to the circumferential track. It is not the point of this paper to make any design of each component of the system as long as it appears generally feasible.

EVA ASTRONAUT TIME REQUIREMENTS

With the rationale of assembly established and a design fairly well fixed, the next step will be to determine the astronaut requirements. This should include fatigue, ECS, time and difficulty. Obviously, if the assembly task becomes a physiological impossibility, then man would have no role. So it will be necessary to scale the task (complete antenna) to be reasonable with these factors. First, let's make an estimate of how long it will take to assemble the 240 panel antenna. The General Dynamics study showed that the average time to grasp, rotate and install an 85 square foot panel was 0.80 minute. This was achieved with a relatively simple fastener concept, a Gemini space suit, and while underwater. Data has shown (Ref) that space will be 1.5 times as slow as underwater simulation data. Therefore, the assumptions for this evaluation will be as follows:

EVENT	TIME	REMARK
Time Per Panel	1.20 (Min)	0.80 x 1.5 (50% Over Estimate)
Time to Translate	0.50 fps	
Rest	5 min/30 min	

The over-all estimate to assemble the antenna will therefore be:

240 x 1.2 min	=	288 min
1881 x 72 x 0.5 ft/sec	=	62 min
Rest = 11.0 periods		55 min
		405 min

This figure of 405 does not seem too unreasonable as it means he will install 20 panels, rest 5 minutes, etc. The detailed background is as follows:

TABLE

ROW	DISTANCE (ft)	TIME (sec)	PANELS	TIME (sec)	TOTAL TIME (sec)
1-2	9	18.0			
2	29	58.0	9	648	706
2-3	9	18.0			
3	85	170.0	15	1080	1250
3-4	9	18.0			
4	141	282	21	1500	1782
4-5	9	18			
5	198	396	27	1920	2316
5-6	9	18			
6	254	508	33	2340	2848
6-7	9	18			
7	311	622	39	2820	3442
7-8	9	18			
8	367	734	45	3240	3974
8-9	9	18			
9	424	848	51	3660	4508
TOTAL DISTANCE	1881 (ft)			TOTAL TIME -	20826 Sec
				or	347 Min
				Rest	58
				TOTAL	405 Min

MAN'S CHANGING ROLE

The earlier roles for the extravehicular astronaut has been to construct and assemble large structures, repair and replace malfunctioned components, and conduct scientific experiments. In the construction and assembly area, one may ask "How big of a job can an astronaut be expected to accomplish in space?" Also, "At what point does man become effective over more automatic techniques?" These are very difficult questions to answer with any degree of validity. The experience from the Gemini flights demonstrated that man will be tired or exhausted in a short time from what appeared to be not too difficult of tasks. Major Aldrin's final flight on Gemini XII demonstrated improved EVA performance if paced at a slower rate and provided with frequent rest periods.

Reviewing the antenna design analyzed in this study, it appears man can assemble this large of a structure without too much difficulty. It will, however, require the astronaut to physically handle 240 separate panels, translate about 1/2 mile, and be out for 6.75 hours of time (optimistic estimate). This is all based upon a 9.0 ft x 9.425 ft panel size. Suppose we make the handling job a little easier by reducing the panel size. For instance:

Size	Nr of Panels	EVA Line (hrs)
9x9.4	240	6.75
9x4.7	480	13.50
4.5x4.7	960	27.0
4.5x2.35	1920	54.0

This makes it obvious that although we can reduce the panel size to make individual installations easier, the total time required for EVA increases rapidly. This indicates that the "nut and bolt" concept or fabrication in space will be totally unreasonable due to the amount of EVA time required and the associated weight penality.

The cost of an EVA operation of this sort involves two basic elements - one - the pounds per hour of life support expendables, and two - the weight involved in recycling of an airlock. For the average case of a semiopen life support system and a non reclaimable airlock is as follows:

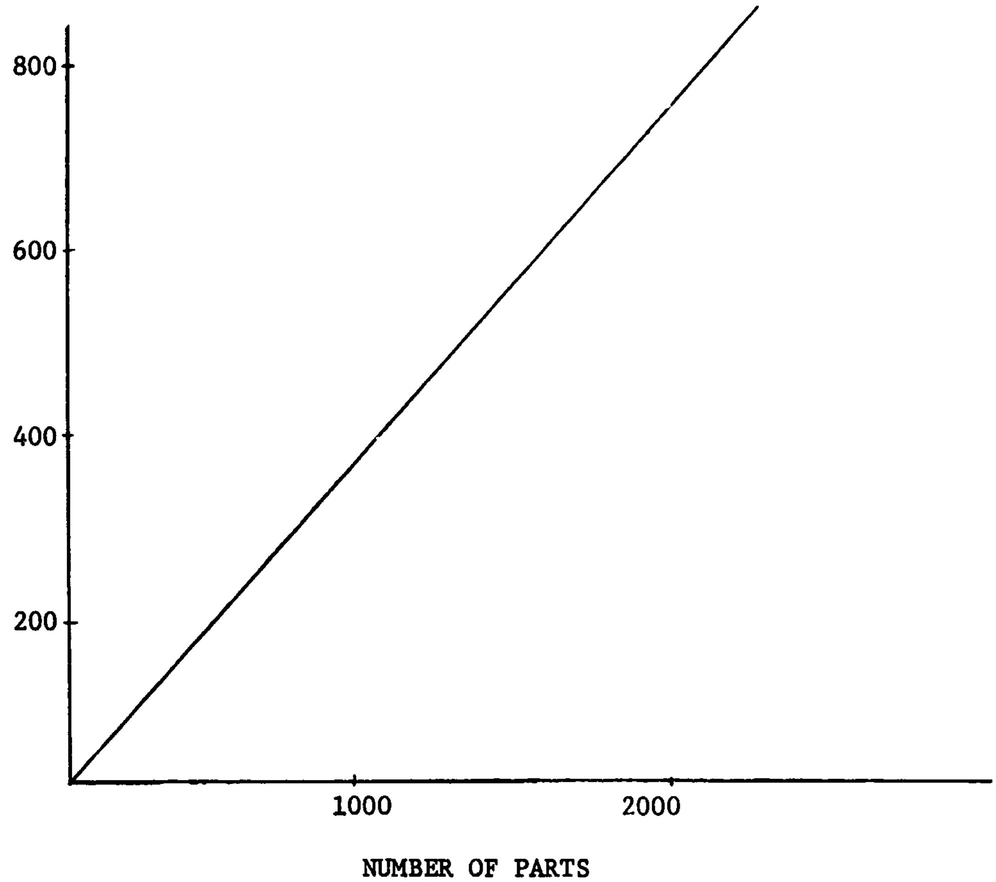
ECS - 8.0 lbs per hour

AIRLOCK - 25 pounds per recycle

In addition, assume the maximum work length is 3.5 hours. This is based on Gemini (2-1/2 hours) and maximum back-packs (4 hours). Using this data as before, we obtain:

SIZE	NR	TIME	WT CAL	WEIGHT
9 x 9.4	240	6.3	50.4 + (25x2)	100.4
9 x 4.7	480	12.3	98.4 + (25x4)	198.4
4.5 x 4.7	960	24.3	194.4 + (25x7)	369.4
4.5 x 2.3	1920	48.3	483. + (25x14)	736.4

With the types of weight penalities, it starts to become obvious why it will be necessary to reduce the number of components and tasks.



Based upon the previous discussion, it is now possible to make some conclusions relative to the EVA assembly task. These are:

a. The number of parts to be assembled must be kept to a minimum to reduce time and weight penalities.

b. EVA support equipment (ECS, tools, locomotion aids, etc) must be designed to be as light as possible to reduce total system weight.

c. Considerably more experimental data is required to establish man's capability to install large panels on a continuous basis. The scaffolding technique can be tested adequately in underwater simulation.

The primary advantage of an EVA assembled antenna, at this time, appears to be the stowage or package volume during launch. However as the panels become larger, the ability to store panels in each available square foot of spacecraft volume becomes questionable. Therefore, work is needed in the following areas:

1. A unique panel design and materials to obtain an extremely lightweight design which is competitive with automatic deployables.

2. Unique fastener and EVA equipment design to reduce assembly time and system weight.

3. Experimental tests to establish firm data on assembly technique and time required.

4. Space tests verify experimental data.

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SESSION II

SPACECRAFT MAINTAINABILITY AND RELIABILITY

Session Chairman: Dr. M. I. Yarymovych
Deputy for Requirements
Office of Assistant Secretary
of the Air Force

REDUNDANCY VS. MAINTAINABILITY
SOME CONCLUSIONS ON THE CROSSOVER FOR MANNED MISSIONS

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SUMMARY: Short duration missions that use automated redundant systems have demonstrated a high degree of success. Manned maintenance may be weight effective for long duration missions. A technique for determining the crossover, the mission time beyond which a maintainable design is more effective, is described. The crossover time's sensitivity to variations in maintenance and automation parameters are discussed. Analyses show that manned maintenance becomes weight effective for a typical Earth-orbiting space station when the mission exceeds a few months.

INTRODUCTION

It is generally recognized that space vehicles require manned maintenance if they are to have a high probability of success for long duration missions. In fact, without man, high probability of success may be unachievable, due to the inability to foresee all possible malfunctions and system interactions. It has been demonstrated that manned maintenance is unnecessary for relatively short missions; that designs incorporating such maintenance would impose inordinate penalties. What has not been determined is the particular time, or crossover; at which a maintainable system is more effective. The need to identify this crossover was established during studies of an earth-orbiting space station conducted by the Grumman Aircraft Engineering Corporation. Since man is on-board to perform experiments, monitoring and control functions, it would be beneficial to determine whether the vehicle design should allow for man to perform the maintenance tasks. The space station involved is an earth orbiting platform that provides accomodations for the experimentors, and a stable platform for the astronomy, solar, meteorology, earth resources and biology experiments. A vehicle of this size, which supports scientific experiments, is much more complex than prior space vehicles. From a reliliability point of view, it is at least an order of magnitude more complex, with an equivalent piece part count of more than four million parts. To protect the high investment of such a program, the mission success goal will have to be very high, probably .95 or greater.

Inherent reliability cannot meet these requirements. Optimistic projections of equipment (assemblies, black boxes, etc.) failure rates are of the order of one to ten failures per million hours. To achieve a system mission success goal of 0.95 for a five year mission the sum of all equipment failures rates must total about one failure per million hours. Since

CONCLUSION ON THE CROSSOVER

Analysis of the results show that a mission duration of a few weeks is enough to make a maintainable design "yield dividends". Figure 8 is an isometric representation of the results showing the bounds and trend of the crossover time. The surface represents the mission length at which a maintainable system becomes identically equal in weight to a "non-maintainable system". If the mission duration is to exceed this time then a maintainable design will result in a lighter vehicle.

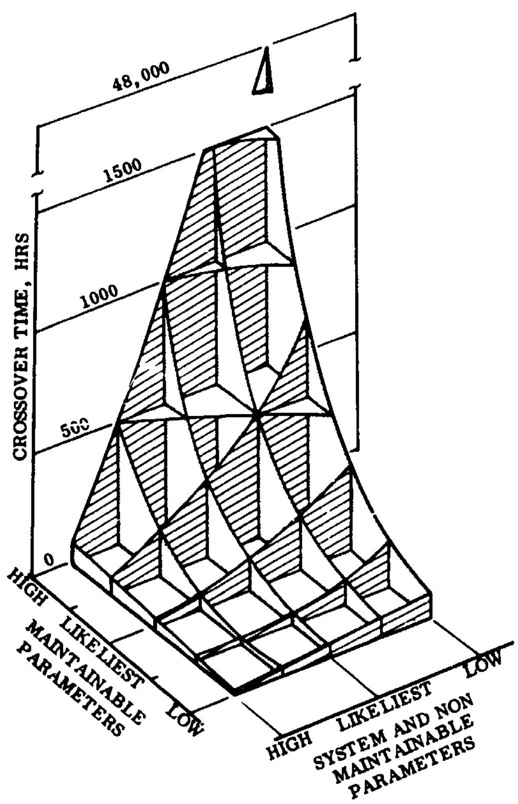

FIGURE 8 CROSSOVER SURFACE

If the mission duration is more than a few months the only combination of parameters which would invite a non-maintainable design is as follows:

System - Low reliability goal and complexity
Maintainable Parameters - High
Non-Maintainable Parameters - Low

A sensitivity analysis conducted by varying a single parameter at a time concluded the following:

A. Crossover time decreased if any of the following parameters are increased:

System Reliability Goal
System Complexity
Switch Weight
Switch Failure Rate
System Failure Rate

B. Crossover time increases if any of the following parameters are increased:

Connector Weight
Connector Failure Rate
Tools and Technical Data
Isolation Valves
Induced Failures
Isolation Valve Failure Rate

The authors conclude that for any manned spacecraft, a maintainable system is "optimal", if the mission is to be for more than a couple of months. A best long duration vehicle will include some automated redundancy in critical systems with short allowable down times to overcome man's deficiency of requiring time to make repairs.

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APPENDIX

PROCEDURE FOR DETERMINATION OF CROSSOVER TIME

As noted in the paper, a general technique was developed to determine the approximate time at which an automated non-maintainable system weighs more than a man maintained system. The procedure, together with the assumptions and approximations are outlined below:

The following "basic inherent functional system" must be identified from preliminary analysis.

a) The total number of equipments or potentially replaceble units required to perform the objectives of the mission. A replaceable equipment could be a single component such as a Sabatier Reactor in an EC/LS subsystem, or a group of components such as V.H.F. transmitter in a communication subsystem.

b) Approximate failure rate of each equipment and the sum total of all equipments.

c) Mission success or reliability goal

d) Weight of equipment defined in a) above.

As shown previously, due to bimodal failures, the best form of automated redundancy is generally standby where the spares are off line and non operating. Thus, the next step is to define the penalties in the form of auxiliary or additional equipment required to implement manned maintenance or standby redundancy.

Following is a tabular summary of auxilliary equipment:

Automated Standby

Switches
Wiring and Plumbing
Checkout Instrumentation
Spares Environmental Control

Manned Maintenance

Tools and Manuals
Connectors
Checkout Equipment
Isolation Valves and Relays
Man (Time and Skills)

In order to derive the crossover time it is necessary to develop a plot of system weight versus time for each of the above mechanized systems and obtain the point of intersection. The above lists were reduced by deleting items that were common and/or of a single equipment, at best, will meet this goal, it is obvious that a system containing several thousand equipments will have a much lower reliability. Figure 1 shows the inherent reliability forecast for a single equipment and a total system. Obviously, there is a large gap between inherent reliability of equipment, and the reliability requirements of a long duration mission. Fortunately, there are techniques available to bridge this gap. Redundancy, in the form of either parallel operating redundancy, or automatic switch-in standby redundancy can significantly improve system reliability. In manned missions, a maintainable system could be provided using man to perform the repair actions.

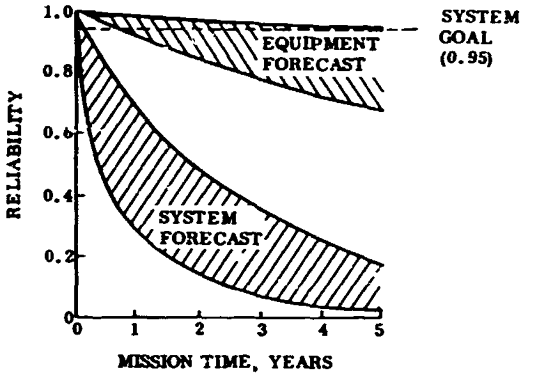

FIGURE 1 RELIABILITY OBJECTIVE/FORECAST

Evaluation of redundancy techniques must be accomplished with regard to a specifiec design constraint. At the early design stage, many of the design constraints, such as cost, power, volume, and manpower, are not too clearly defined. They are subject to variation, depending on mission objectives and vehicle design. However, weight of a vehicle is a well defined constraint, determined by the capability of the launch vehicle to inject a payload into the required orbital altitude and inclination. Therefore, tradeoff studies of alternative redundancy techniques use weight as the dominant constraint.

REDUNDANCY

Parallel operating redundancy requires that all redundant equipment be on line, simultaneously, with the basic equipment. The system does not fail as long as one of each required equipment is operative.

Standby redundancy does not require the redundant equipment to operate until the basic equipment fails. This reduces the redundant equipment's exposure to failure.

Figure 2 shows the "theoretical" weight/reliability curves for parallel and standby redundancy. Assuming perfect switching, standby redundancy is a more efficient means of achieving the reliability goal. The abscissa, λt, is the product of the equipment failure rate (λ) and the mission time (t).

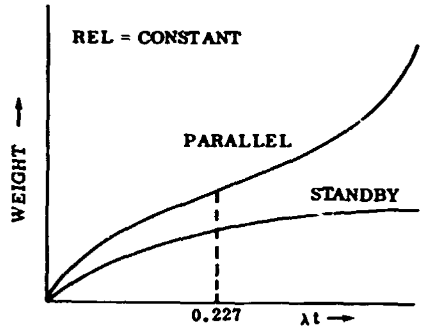

FIGURE 2 THEORETICAL REDUNDANCY CURVES

The parallel redundancy curve has an inflection point (minimum slope) at a constant value of λt (0.227). The slope is continuously decreasing from 0 to 0.227 and then continually increasing beyond 0.227. The standby redundancy curve has a continuously decreasing slope as λt increases. Thus, for λt values greater than 0.227 the two curves diverge very rapidly. This point suggests a practical limit to the economical use of the parallel redundant technique.

Practical application of these redundancy techniques departs from the theoretical. Parallel redundancy involves problems of multi-failure modes of equipment. Such equipment must be arranged in a series parallel configuration to protect against each mode of failure and results in complex networks of equipment. Figure 3 shows a typical series-parallel configuration.

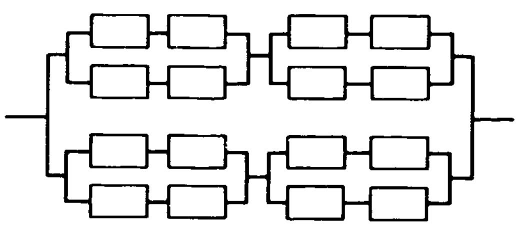

FIGURE 3 SERIES-PARALLEL CONFIGURATION

In addition, not all equipments lend themselves to operating redundancy techniques. In many cases, system characteristics change when one or more of the parallel redundant equipments are inoperative.

Standby redundancy is highly dependent on the reliability of the switch and sensor mechanism. The switches (or valves), considered to be continually operating, have a time dependent reliability. Increasing the reliability of an automatic standby system results in a complex network of switches and sensors controlling the standby redundant equipment, as shown in Figure 4.

Standby redundancy can be invoked, if the switch can be replaced by a mechanism whose re-

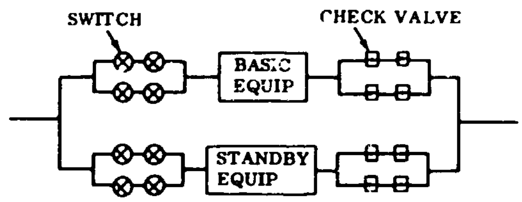

FIGURE 4 STANDBY REDUNDANCY CONFIGURATION

liability is not time dependent. Man may be considered as such a mechanism. By installing separable connectors and some system isolation equipment, a maintainable system can be designed in which malfunctioning units are simply removed and replaced with spare parts.

In practice, a combination of techniques may be required. Redundancy may be required, if the allowable downtime is insufficient for the maintenance tasks confident performance. However, the redundant system can be designed to permit replacement of the primary equipment while it is operating in a redundant mode.

ANALYSIS TECHNIQUES

Determination of the crossover between a maintainable and a nonmaintainable system for a specific design requires a detailed analysis of each component of the system. The analysis consists of calculating the best (lightest weight) nonmaintainable configuration for each component for various time periods. This implies a trade-off between operating redundancy and automatic standby redundancy, at a fixed reliability or mission goal. The total weight for the nonmaintainable configuration is obtained by calculating the individual weights of each component configuration for each time period, resulting in a weight versus mission time curve.

A similar curve is computed for the maintainable case by determining the weight of spares, isolation valves and connectors, to achieve the same reliability goal. Figure 5 shows the general shape of the curves and crossover point.

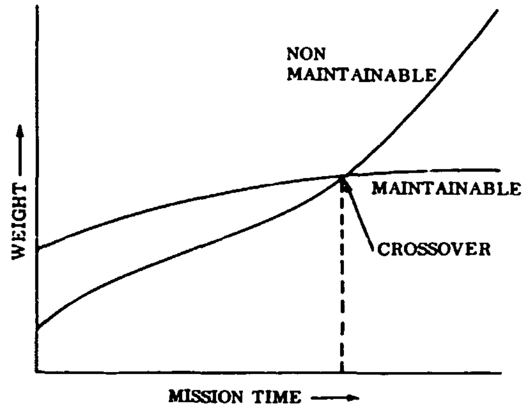

FIGURE 5 MAINTAINABLE VS NON-MAINTAINABLE WEIGHT VS MISSION TIME

This component level analysis requires that the system under study be completely defined, which is not feasible during the early phases of a space vehicle's design. This complex analysis is also time consuming, therefore a simpler and more general technique was developed, which is outlined briefly below, and described in detail in the Appendix. This technique determines the approximate time at which an automated non-maintainable redundant system weighs more than a man maintained system.

A gross system level analysis is performed by identification of functional equipments and their approximate failure rates and weights. From this basic system definition, an average failure rate ($\bar{\lambda}$) and maximum number of replaceable units (M) is determined. Using $\bar{\lambda}$, M, switch weight and failure rate, a nonmaintainable system weight versus time curve, Figure 6, is generated assuming standby redundancy with automatic switching. To derive the maintainable system weight versus time curve, auxilliary equipment such as manuals and tools, isolation valves or relays, and connectors are defined. The spares required are calculated based on $\bar{\lambda}$ and M values. A weight penalty is also introduced to account for imperfect repairs. The composite curve is shown in Figure 7.

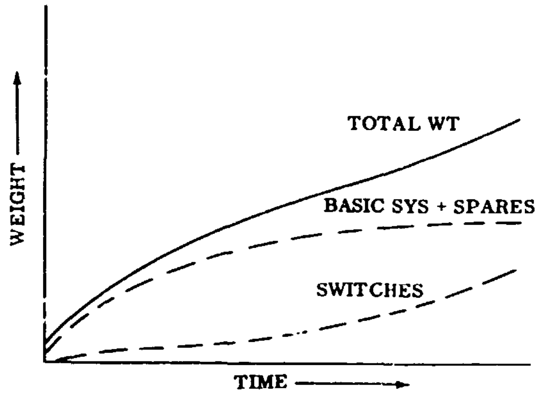

FIGURE 6 NON MAINTAINABLE WEIGHT VS TIME

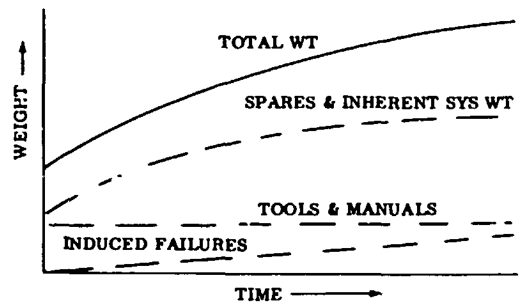

FIGURE 7 MAINTAINABLE SYSTEM WEIGHT VS TIME

The maintainable and nonmaintainable curves are then both plotted as in Figure 5 to determine the crossover time.

Each crossover curve is obtained by fixing the following system parameters:

Both

Reliability Goal
System Weight
Average Inherent Basic System Failure Rate

Nonmaintainable

Valve or relay failure rate
Valve or relay weight

Maintainable

Connector weight
Number of isolation valves
Connector failure rate
Weight of manuals and tools
Percentage of faulty repairs
Isolation valve failure rate

These analyses considered man normally on-board the vehicle as part of the mission or experiment package. If man were not on-board, his introduction solely for maintenance purposes would have a pronounced effect on the crossover time. He and all his support equipment would have to be included in the maintenance penalties. In this case, a thorough analysis to determine the crossover must consider such factors as:

- o Interchangeability - One spare part can replace many similar parts throughout the vehicle, thus reducing spares weight.[1,2]
- o Support by logistic resupply, if vehicle is in earth orbit, thus lowering on-board spares weight.
- o Flexibility in level of replacement.
- o Versatility in flight profile.
- o Simplicity of design since redundant parts do not have to be designed into the operating system.

APPLICATION

Grumman has conducted studies covering earth orbiting vehicles for missions of up to five years. A typical vehicle is equipped for stellar and solar astronomy, meteorology, biology and earth resources investigation. It provides living accommodations for the experimentors, and a stable platform for the experiments. This space station would have an upper limit of approximately 220,000 lbs. (due to Saturn V launch vehicle constraints) and house between nine and twelve men. To provide these basic functions, the vehicle consists of the following subsystems.

Power Generation and Distribution
Environmental Control/Life Support
Navigation and Guidance
Stabilization and Control
Propulsion and Reaction Control
Data Handling and Communication
Instrumentation
Controls and Displays

Incorporated in subsystem design are innovations to minimize consumable requirements such as:

a) Oxygen recovery from Carbon Dioxide
b) Water reclamation from humidity condensate and urine
c) Control moment gyros for removal of cyclic disturbances
d) Magnetic unloading for removal of gravity gradient bias torques
e) Use of waste products as fuel in a resistojet propulsion system

Using the above defined vehicle as a typical long duration manned spacecraft, an analysis was conducted to determine the crossover time. The parameter values, representative of this vehicle, required to apply the analytical technique outlined in the Appendix were generated and are listed in tables I and II. Upper and lower bounds were also generated so that a sensitivity analysis could be conducted on the assumed likeliest values.

TABLE I
SYSTEM AND NON - MAINTAINABLE PARAMETERS

System	LOW	LIKELIEST	HIGH
Rel Goal	.9	.95	.99
Total Components or Functions	1000	2000	4000
Average Component Failure Rate $(\frac{\text{Fail}}{\text{Hr.}})$	5×10^{-6}	10^{-5}	2×10^{-5}
Average Component Weight (lbs)	10	10	10
Non Maintainable Penalties			
Switch Weight (lbs)	1	2	3
Switch Failure Rate $(\frac{\text{Fail}}{\text{Hr.}})$	10^{-7}	10^{-6}	10^{-5}

TABLE II
MAINTENANCE PARAMETERS

	LOW	LIKELIEST	HIGH
Connector Weight (lbs)	$\frac{1}{2}$	1	2
Connector Failure Rate $(\frac{\text{Fail}}{\text{Hr.}})$	2.5×10^{-7}	10^{-6}	4×10^{-6}
Tools and Tech Data (lbs)	200	1000	4000
Isolation Valves	100	200	400
Isolation Valves Weight (lbs)	1	1	1
Induced Failures	10% of Expected Number of Failures		
Isolation Valve Failure Rate $(\frac{\text{Fail}}{\text{Hr.}})$	10^{-6}	10^{-6}	10^{-6}

The crossover times established by varying maintainable penalties non-maintainable penalties and system complexity using the technique described in the Appendix are shown in Table 3.

For example, suppose we had a system with "low" parameter values:

Reliability goal	0.9
Components	1000
Avg Fail Rate	5×10^{-6}
Avg Comp Wt.	10

And if we assumed "low" non-maintainable penalties:

Switch wt.	1
Switch Fail rate	10^{-7}

And if we assumed "likeliest" maintainable system penalties:

Connector wt.	1
Connector fail rate	10^{-6}
Tools and tech data	1000
Isolation valves	200
Isolation valve wt	1
Isolation valve fail rate	10^{-6}
Induced failures	10%

We would enter Table 3 at the LOW system requirements, and the 1 lb Switch weight (third row down), and at the LIKELIEST maintainable penalties (second column), and find that the crossover time is 330 hrs. That is, if the mission time was scheduled to be more than 330 hrs., a maintainable system would be more desireable.

TABLE 3 CROSSOVER TIME - HOURS

System Req	Sw Fail Rate	Switch Wt	Maintainable Parameters: Low	Likeliest	High
Low	10^{-7}	3	50	180	2500
		2	70	285	5000
		1	90	330	48,000
Likeliest	10^{-6}	3	40	100	550
		2	50	130	1000
		1	75	200	5000
High	10^{-5}	3	~ 0	10	75
		2	~ 0	20	110
		1	~ 0	30	700

insignificant weight. Thus, the trade-off was an automated standby system with switches versus a man maintained system with penalties for tools, connectors, isolation values and relays.

Non Maintainable Automated System

The automated system is illustrated in Figure 9 where each equipment and spare is protected by a switch which consists of a package of electrical relays or hydraulic valves, depending upon the subsystem.

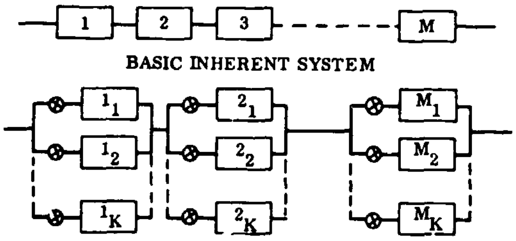

FIGURE 9 STANDBY REDUNDANT SYSTEM-MAXIMUM SWITCHES

The number and weight of switches increases as the level of implementation decreases. On the other hand, the spares and weight decrease at a lower level of redundancy, due to the standby units lower failure rate. An analysis was, therefore, conducted to determine the optimal level of switching to give minimum system and spares weight to achieve the system reliability goal. The minimal number of switches is shown in Figure 10 which illustrates the case of a completely redundant system. If any one equipment fails all the units on line in that system are useless. The maximum switch case was illustrated in Figure 9 above where a failure of one equipment does not affect any others.

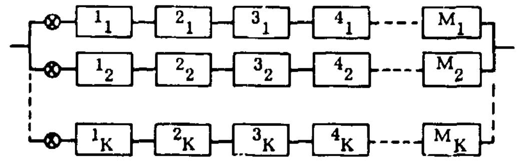

FIGURE 10 STANDBY REDUNDANT SYSTEM-MINIMUM SWITCHES

The reliability expression, assuming that all switches must operate and that at least one of each standby unit is in operation, is as follows:[3]

$$\text{Rel} = (\text{Rel Switch})^{N+\sum_{i=1}^{N} k_i} \prod_{i=1}^{N} \sum_{x=0}^{k_i} e^{-\lambda_i t} \frac{(\lambda_i t)^x}{x!} \qquad (1)$$

k_i = number of spare standby units for i^{th} serial unit

λ_i = failure rate of i^{th} serial unit

t = time of operation

N = number of serial standby units in basic system and can vary from 1 as in Figure 10 to M as in Figure 9.

To reduce the enormous number of combinations possible to achieve the same reliability, the goal was apportioned equally between the switches and spares. Therefore,

$$(\text{Rel})^{\frac{1}{2}} = (\text{Rel Switch})^{N+\sum_{i=1}^{N} k_i} = \prod_{i=1}^{N} \sum_{x=0}^{k_i} e^{-\lambda_i t} \frac{(\lambda_i t)^x}{x!} \qquad (2)$$

An estimate ($\bar{\lambda}$) of the average equipment failure rate is generally obtainable early in the preliminary design and believed more accurate than the individual equipment estimates, thus, a simplification was made in calculation of the number of spares required to achieve the reliability goal.

The number of spares must be an integer for an individual unit, but the system average or expected value need not be. If it is assumed that spares are a continuous and linear function of λt, which is a good approximation in the high reliability range for λt, greater than 0.2, see Figure 11, then the spares required can be calculated as follows:

$$\text{Number of spares}_i = A_i(\lambda_i t) + B_i \quad (3)$$

$$\text{Total system spares} = \sum_{i=1}^{N} \text{spares}_i$$

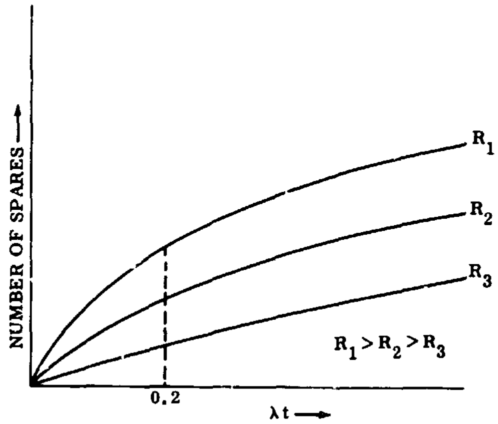

FIGURE 11 SPARES VS λt

If many of the standby units failure rate multiplied by time (λt), are less than .2 a linear regression line can be established. Equation (3) with A and B constant for all i, is used in calculating spares for all units, if all units have the same reliability goal. Thus,

$$\text{Total spares} = \sum_{i=1}^{N} A_i(\lambda_i t) + B_i$$

$$= A_i \sum_{i=1}^{N} \lambda_i t + NB$$

And if t = constant for all equipments then:

$$\text{Total spares} = At\sum_{i=1}^{N} \lambda_i + NB$$

$$\text{But } \bar{\lambda} = \sum_{i=1}^{N} \lambda_i / N$$

Therefore:

$$\text{Total spares} = N(A\bar{\lambda}T + B)$$

Where N represents the number of switches in the basic inherent system and ($\bar{\lambda}$) is the average failure rate of a piece of equipment or group of serial equipments. For a specified reliability goal **A** and B are determined and total spares weight, assuming all equipments approximately equal in weight, is equal to

$$(\text{Inh. Sys. Wt.}) \times (A\bar{\lambda}t + B)$$

The total number of switches required is equal to the total number of spares plus N.

$$\text{Number of switches} = N + N(A\bar{\lambda}t + B)$$

The required reliability of each switch is determined from eq. (2)

Req'd switch rel. =

$$(\text{Rel. Goal})^{\frac{1}{2(N+N(A\bar{\lambda}t+B))}} \quad (4)$$

The switch is assumed to consist of a series of values or relays, each with a single mode of failure. If a switch element has an exponential failure density function, and a switch is composed of a group of elements in parallel operating redundancy (1 out of 3 must operate), then the following expression determines the number of elements per switch to achieve the reliability goal of the switch.

$$\text{Rel switch} = 1-(1-e^{-\lambda_{sw} t})^S$$

$$\text{or } S = \frac{\log(1\text{-Rel. Sw.})}{\log(1-e^{-\lambda_{sw} t})}$$

Where

λ_{sw} = Failure rate of valve or relay

t = Time of mission

Rel. Sw. determined from eq.(4)

The total system weight (switches + spares + basic system) for a particular time, reliability goal, average component failure rate, and N switches in inherent basic system is now determined.

$$\text{Total Wt.} = S(N+N(A\bar{\lambda}t+B))\,(\text{Sw.El.Wt.}) + (N+N(A\,\bar{\lambda}t+B))(\text{Sys.Wt.})$$

By varying the number of switches, N, in the basic inherent system, a curve was developed for a specific mission time from which the minimal system weight could be determined. Figure 12 is a plot of system weight versus number of switches with W_1, W_2, W_3, corresponding to the minimum weights for time t_1, t_2, t_3 respectively. As time increased the number of switches at which the system weight was minimum also increased.

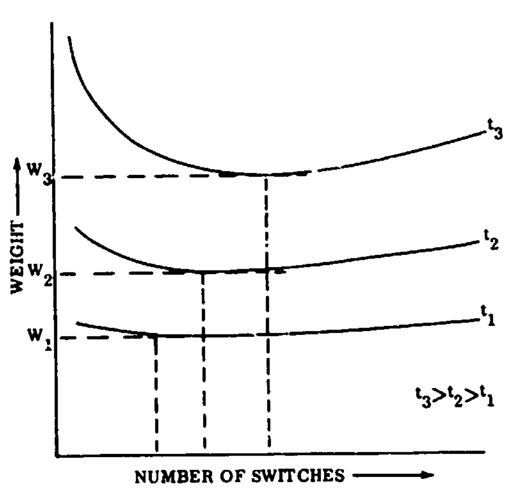

FIGURE 12 WEIGHT VS NUMBER OF SWITCHES

The minimum weights, W_1, W_2, W_3 were then plotted against mission time t_1, t_2, t_3 to establish the non-maintainable system curve as shown in Figure 6 in the paper. The two curves composing the total curve are shown with broken lines. The switch weight grows with time, similar to parallel operating redundancy shown previously, whereas the spares weight has a continuously decreasing slope similar to standby redundancy.

Man-Maintained System

The man-maintained system weight and parameters, assuming man on board as part of the mission experiments, were defined as follows:

a) The total replaceable equipments were identified.

b) The number of isolation valves required to allow for on line repair and replacement.

c) The total replaceable units were then equal to the sum of a) and b).

d) The weight of inherent system was calculated and increased by a total connector weight. Each on line and spare unit was assumed to require a connector, to allow for the maintenance action.

e) System failure rate was increased to account for the added connectors and isolation valves.

f) A fixed (independent of mission time) tools and manuals weight was generated.

g) A percentage of the expected number of repair actions were considered to be bad.

The system schematic is shown in Figure 13.

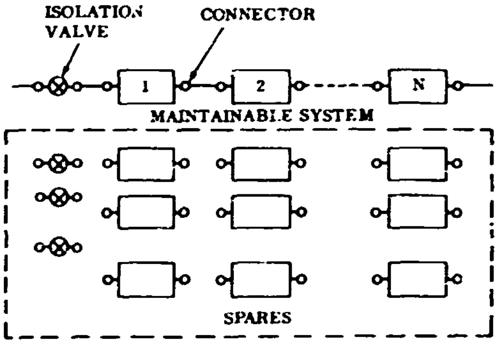

FIGURE 13 MAINTAINABLE SYSTEM WITH SPARES

On-line standby equipment may be required for some critical systems, where the probability of completing the repair in the allowable down time is low. But this would merely change the spare unit's location and would not effect the weight significantly. In fact, most systems have relatively long allowable down times in which man must accomplish the repair action.[4] Therefore, to achieve the mission goal sufficient spares must be allocated. The probability of not running out of spares for a piece of equipment with exponential failure density function is calculated by use of the cumulative Poisson expression with parameters defined previously.

Prob. of equipment operating =

$$\sum_{x=0}^{k} e^{-\lambda t} \frac{(\lambda t)^x}{x!}$$

As in calculation of the spares required for the non-maintainable case, the linear approximation was made with

$$\bar{\lambda} = \sum_{i=1}^{B} \lambda_i / B$$

Where $\sum_{i=1}^{B} \lambda_i = \sum_{i=1}^{N} \lambda_i + N \lambda_{connector}$

$$+ (B-N)\lambda_{isolation\ valve}$$

N = Number of replaceable equipments

B-N = Number of isolation valves

If K is the number of spares per on line equipment calculated by use of $\bar{\lambda}$ then

Total weight = (K + 1) (System Weight) + (Manuals and Tools wt) + % of expected number of repairs ($B\bar{\lambda}t$) with system weight defined in (d) above.

The general shape of the resulting curve is shown in Figure 7 with a breakdown of the composite. The crossover time is determined from the intersection of the resultant graphs of Figures 6 and 7 and typically shown in Figure 5.

This technique may be applied not only to spacecraft, but any vehicle or equipment requiring a tradeoff between manned maintenance and an automated redundant system. The crossover time generated is not exact and thus should only be used as a guide. If the mission time is near the crossover time further detailed analysis may be necessary to arrive at the optimal system.

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SPACE SYSTEMS MALFUNCTION ISOLATION

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SUMMARY: One of the pacing factors associated with the in-space maintenance (ISM) problem is the need to know when systems are operating correctly, and how to provide diagnostic facilities when the systems are malfunctioning. This paper illustrates how to determine what system parameters to monitor without exceeding the bounds established by reasonable expectancy.

INTRODUCTION

All manned space flights, so far, have been conducted safely without resorting to maintenance in flight. The unknowns surrounding man's ability to perform maintenance in space have been a strong motivating factor in taking other approaches. The problem has been proliferated by the desire to stretch out these missions in deep space and in the time domain. Consequently, it has become more and more obvious that some form of In-Space Maintenance (ISM) must be planned.[1,2]

To implement such a program, it is necessary to provide adequate on-board information on systems performance and follow specific guidelines. Correct operation must be confirmed by a performance monitor and diagnostic data must be provided whenever a malfunction occurs. The information must be timely so that restoration can be safely accomplished within time constraints, but must not unbalance the spacecraft or crew with superfluous weight or time-consuming activities.

Since the average spacecraft has literally millions of parts, it follows that each one, theoretically, contains the possibility of failure. However, experience has shown that we must only be prepared for those malfunctions that actually occur; yet we tend to want to bring the whole store with us to meet any eventuality. Experience has demonstrated that there are practical limits, established by effectiveness criteria and/or weight and storage space limits. How then can the space-mission planner determine what to evaluate and how many things to monitor without exceeding the bounds established by reasonable expectancy?

The usual approach to test system design is to poll all of the subsystem engineers and ask what they think should be monitored and where the test points should be placed. The tendency is to look at everything and the result is an in-flight test system which approaches the size and weight of the rest of the mission systems. A technical decision must be made to reduce, to a manageable level, the number of parameters monitored. A random approach can present a frustrating dilemma while a logical process can save much weight, time, and program funds without impairing the probability of mission accomplishment. Therefore, one alternate solution is to eliminate all ISM potential from the mission and make the functions sufficiently redundant and reliable so as to cancel out the need for diagnostic equipment. The space programs to date have essentially operated on this principle. However, the time has come with the extended mission studies when this can no longer be used; the ISM must be properly planned to assure reasonable probabilities of mission success.

PREDICTING THE ISM

A logical way to resolve this problem was identified during a recent study of space mission duration extension problems. The basis for the approach was the "unreliability index," the "failure manifestation," and the "failure mode" data. By using these data the "expected" maintenance action is identified specifically and the functions with acceptable risks are eliminated from further maintenance considerations.

Consequently, the performance monitoring and diagnostic equipment can be designed to deal only with those "expected" failures and the primary failure manifestations. When this is achieved, an ISM plan can be evolved and implemented, based on probability and logic, rather than an incoherent, random process.

The procedure involves use of reliability data, not in the absolute realm, but as a relative figure of merit. This ISM identification technique is illustrated in Figure 1.

The analysis starts with a reliability assessment of the systems. First, each part or assembly is listed in order of unreliability. Then the failure modes and manifestations are determined. After that a complete ISM plan can be devised by using these factors as a guide.

The failure manifestation helps select the performance-monitoring requirements, the diagnostic equipment, and subsequently the safety and training requirements. This results from our ability to specifically identify the potential failure, its modes and manifestations. And this makes it possible to preselect the best or most logical manifestation to use.

The failure mode analysis already has been shown as a useful tool, first for selecting the best level to spare, the required spares complement, the tools required, as well as the supporting design requirements and constraints.[3]

The unreliability index provides the necessary visibility into the LSM requirements. By ordering the components, as indicated by

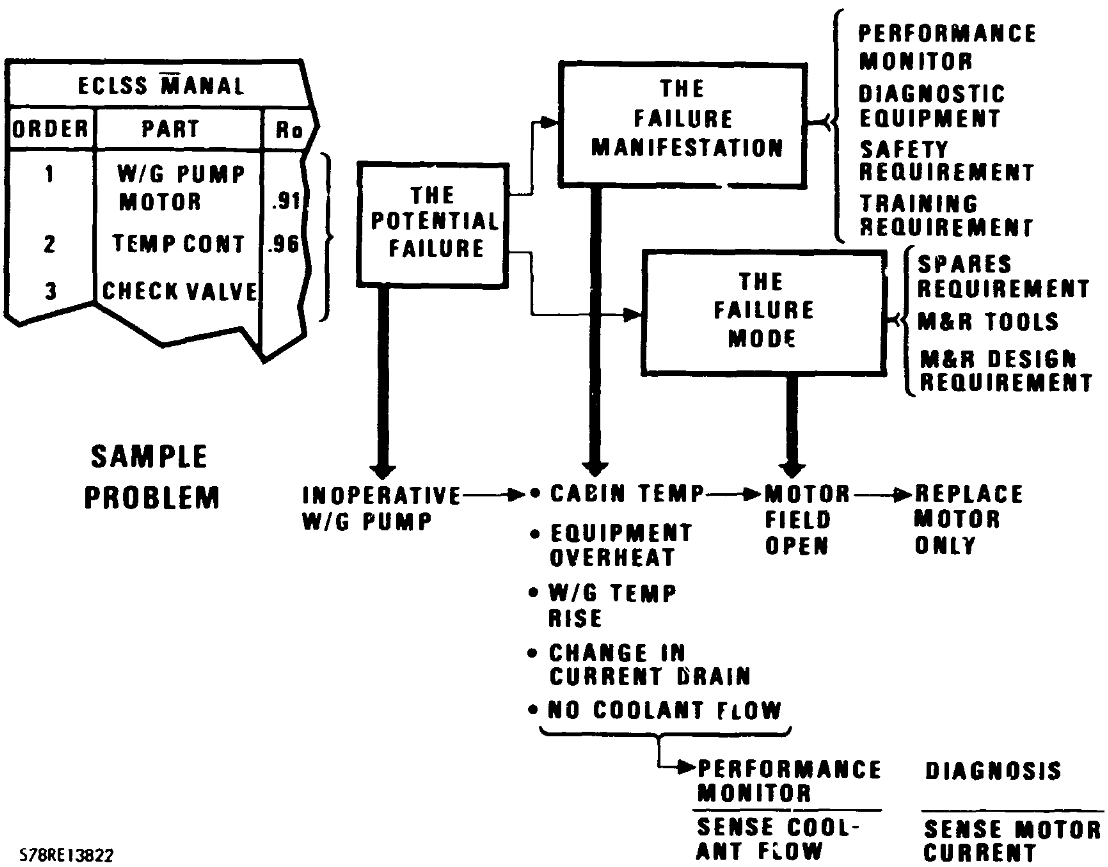

Figure 1. LOGIC FOR MEETING MAINTENANCE PROBLEM

the example of Figure 2, the weakest links, or the most failure-prone components appear at the top and the least likely appear at the bottom. Then an acceptable risk level is established, not necessarily for the entire mission or system, but rather for the individual functions of each mission system.

The level of assembly at which the risk should be equalized depends on the specific mission/system. For the example chosen, the black box level is used; that is, each item is a separate and identifiable box, and its reliability is expressed as a bar to the nearest order of magnitude. Each additional segment of the bar represents the effects of a spare stored under optimum conditions. Providing for ISM's tends to neutralize some of the effects of uncertainty in the actual reliability and the resulting mission is less sensitive to these uncertainties.

Figure 2 lists the data storage unit as the weakest link in the communications and data system, and by drawing a line at the acceptable risk level, items requiring maintenance are identified specifically. All others are ruled out by definition; that is, we have inferred that they are within the acceptable risk. Therefore, for this example, the ISM requirements to achieve or exceed 0.999 per component are as indicated in Table 1.

These data indicate that the data storage system may fail up to three

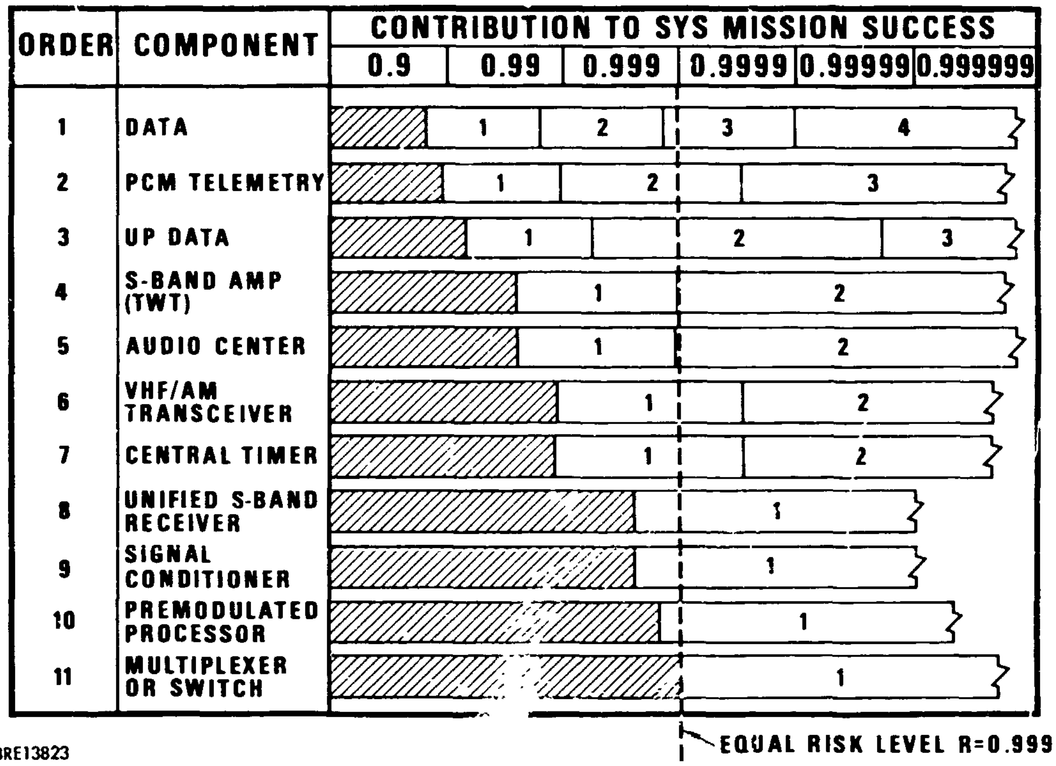

Figure 2. FUNCTION UNRELIABILITY AND THE SUPPTG ISM REQMTS COMM AND DATA MGMT - BOX LEVEL

times while the central timer would fail only once, and that the multiplexer (MUX) "will not" fail. We now "know" what will fail and how often. These are the basic constituents to a logical approach to malfunction isolation and correction plan.

PLANNING FOR THE ISM

Two things are evident from the latter analysis:

1. What will fail—the scope of the maintenance program;

2. How often they will fail, or what is most likely to fail.

These then are the keys to solving the problem. The next activity required is the identification of how they will fail, the failure mode, how we know it failed, and the failure manifestation.

Table 1. ISM REQMNTS FOR COMMUNICATION AND MGMT SYS

ORDER OF FAILURE	SYSTEM COMPONENT	EXPECTED NUMBER OF ISM ACTIONS
1.	DATA STORAGE UNIT	>2 <3
2.	PCM TELEMETRY	<2
3.	UP-DATA LINK	<2
4.	S-BAND TWT	<1
5.	AUDIO CENTER	<1
6.	VHF/AM TRANSCEIVER	<1
7	CENTRAL TIMER	<1
8.	UNIFIED S-BAND RECEIVER	<<1
9.	SIGNAL CONDITIONER	<<1
10.	MULTIPLEXER	0
	TOTAL NUMBER EXPECTED	<13

The Failure Mode Analysis examines each "unreliable" component in much the same way the component was identified as potentially unreliable in the first place, i.e., the analysis is to determine what part of the assembly fails first and by what probability ratio. In the example of Figure 1, the glycol pump was identified as a potential ISM candidate. The failure mode analysis indicated that the motor would fail first by a ratio of 100 to 1; and further, the motor field was most likely to fail open.

One problem to face, not a part of this paper, is the decision as to what assembly level to make the maintenance action. It could be the glycol pump assembly, the motor, or the motor field.[4] In this case, the motor was the most practical level at which to make the repair.

We now know:

1. What will fail

2. How often they will fail

3. How they will fail

Since the subject of this paper is how to isolate the malfunction in a dynamic situation such as encountered in manned space flight, we must next identify the method of flagging the "look points" or determining where the tests points would be most effective.

The Failure Manifestation is the tool which will permit logical isolation of the malfunctioning component.

Again, from the example of Figure 1, the pump motor is known to be the most unreliable component. Table 2 presents some of the expected manifestations of such a failure, by class.

Table 2. FAILURE MANIFEST, GLYCOL PUMP MOTOR FAILURE

CLASS	MANIFESTATION
PRIMARY	1. NO COOLANT FLOW 2. CHANGE IN CURRENT DRAIN 3. NO VIBRATIONS OR ACOUSTIC OUTPUT
SECONDARY	4. GLYCOL TEMPERATURE RISE
TERTIARY	5. CABIN TEMPERATURE RISE 6. EQUIPMENT TEMPERATURE RISE 7. OTHER REASONS

Failure manifestations can be separated into classes as indicated in the table; any one of these could indicate a pump failure. To avoid unnecessary diagnostic activity, it is necessary to single out those manifestations which would be peculiar to pump motor failure only.

Selecting the appropriate manifestation(s) to use as an index of failure is a critical decision, as shown in Figure 3. Given the failure modes and manifestations of all probable events, an analysis of these data is made to ferret out common traits of manifestations. For example, in Table 3, part of a thermal control loop is illustrated in simplified form. In three of the four potential situations, failure will result in complete or partial loss of coolant flow. Therefore, this could not be used only for isolating the failure to the pump. However, it may be desirable to monitor flow since this

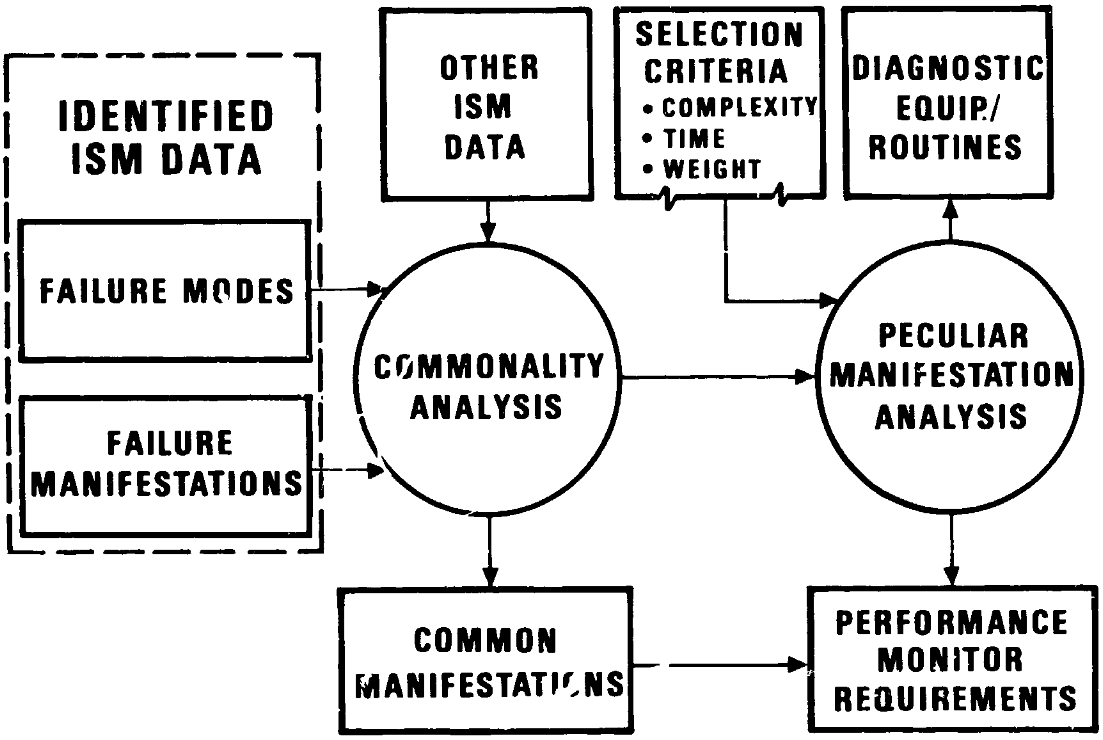

Figure 3. SELECTING FAILURE ISOLATION PROCEDURE AND SUPPORT EQUIP

Table 3. SAMPLE MANIFEST ANALYSIS, THERMAL CONTROL LOOP, PART OF ECS

ASSEMBLY/ FAILURE MODE	PRIMARY MANIFESTATIONS	PECULIAR MANIFESTATIONS	POINTS DIAGNOSTIC
1. GLYCOL PUMP (MOTOR FIELD OPEN)	1. NO COOLANT FLOW 2. CHANGE IN CURRENT DRAIN 3. NO VIB OR ACOUSTIC OUTPUT	1. CHANGE IN CURRENT DRAIN 2. NO VIB OR ACOUSTIC OUTPUT	CHECK CURRENT
2. TEMP CONTROL VALVE (FAILED CLOSED)	1. NO COOLANT FLOW 2. NO CURRENT FLOW 3. NO MANUAL RESPONSE	1. NO CURRENT FLOW 2. NO MANUAL RESPONSE	CHECK CURRENT DRAIN
3. RAD ISOLATION VALVE (FAILED CLOSED)	1. NO COOLANT FLOW THROUGH THAT SECTION 2. NO CURRENT FLOW	1. NO CURRENT FLOW	CHECK CURRENT
4. GLYCOL TEMP SENSOR (NON-RESPONSIVE)	1. IMPROPER TEMP INDICATION 2. LOWER CURRENT DRAIN 3. GLYCOL TEMP EXCURSIONS	1. IMPROPER TEMP INDICATION ON ONE OF THE SENSORS ONLY 2. LOWER CURRENT DRAIN	CHECK REDUNDANT SENSOR

will be a single manifestation which can be used to indicate a failure has occurred. This would then initiate the need for further diagnosis.

Selecting the Performance Monitoring Points involves a set of criteria different from the diagnostic routine. The performance monitor must be simple, yet include all of the "expected" malfunctions. Its job is to sound the alarm indicating that a malfunction has occurred. To simplify the system, reduce weight, time, power consumption, etc., it is desirable to monitor those functional manifestations which are common to the most "expected" malfunctions. Therefore, in the selected example, it would be desirable to sense the coolant flow since stoppage is common to many of the identified failure modes.

Selecting the Diagnostic Points and associated supporting equipment requires consideration of somewhat different criteria. Complexity, time, training requirements, weight of equipment and, most important, the peculiarity of the manifestation are the influencing criteria. As indicated in the example of Table 3, the current drain associated with the specific item for cases 1, 2, and 3 seems to be the best or only peculiar manifestation. As a case in point, consider the glycol pump, where the most probable failure is an open field in the motor. This is manifested by zero current drain and a positive indication of the open field. The same is true of the radiator isolation valve since the open coil of the solenoid would draw no current.

Selecting the Diagnostic Routine involves the establishment of a logical approach to isolating a given malfunction. The proposed approach is based on the premise that the performance monitor (PM) already has indicated that failure has occurred somewhere within a system and/or function, and checking the PM is the first step. The diagnostic routine should indicate the subsequent path to follow in isolating the malfunctioning component to the level selected for the repair or replacement action.

To be most effective, the diagnostic routine should start with the component that is most likely to be the cause and then proceed toward the least likely. As a case in point, consider that the relative failure potential of the sample function is as given in Table 4; the diagnostic routine for this example would then be as reflected in Figure 4. Once it has been established that the glycol is not flowing, the pump motor would normally be checked first, because it is most likely to fail. The next three components are checked in order of significance until the potential malfunctions have been exhausted. Any other components in the function need not be checked since, by definition, the failure must be one of the selected components (within the acceptable risk level).

Sensitivity to Inaccuracies

Perhaps the most pertinent question relating to the use of reliability data, and therefore the acceptance of this approach, relates in turn to the known inaccuracies associated with the input and data.

Table 4. RELATIVE FAILURE POTENTIAL, GLYCOL LOOP COMPONENTS (EXAMPLE)

ORDER	COMPONENT	HAZARD RATIO
1.	GLYCOL PUMP	1000
2.	TEMPERATURE CONTROL VALVE	100
3.	RADIATOR ISOLATION VALVE	10
4.	GLYCOL TEMPERATURE SENSOR	1
	OTHERS	INSIGNIFICANT

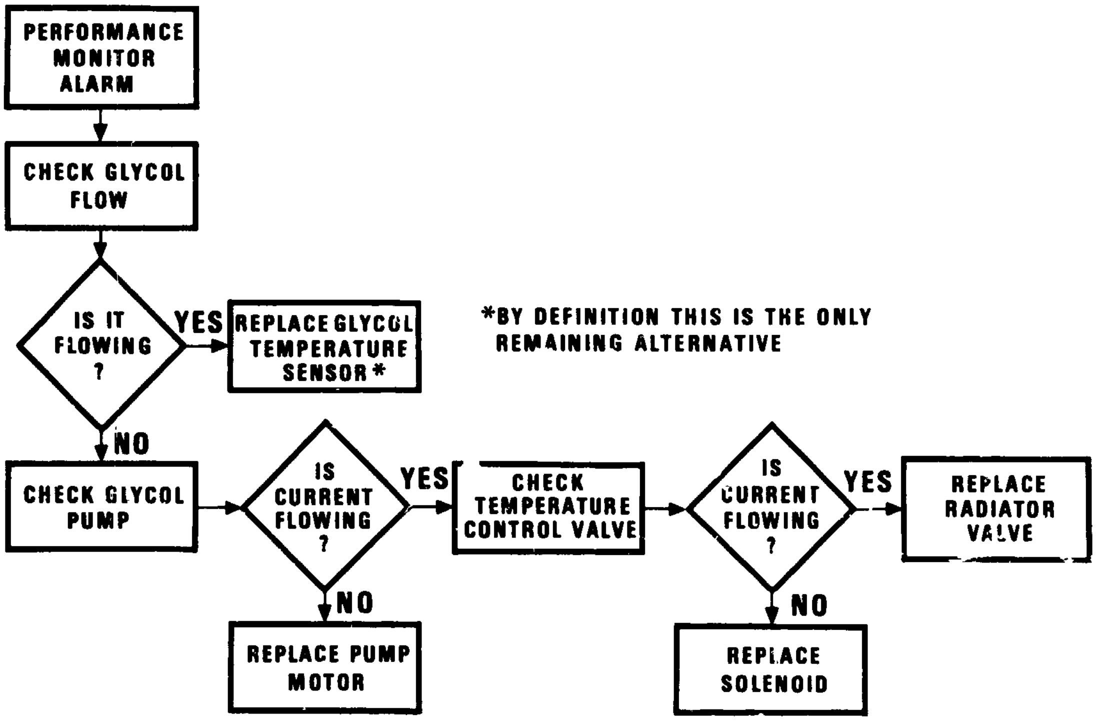

Figure 4. DIAGNOSTIC ROUTINE, GLYCOL FLOW, PART OF ECS

Reliability data is known to be suspect in the absolute realm and, at best, good to about one place, even in the relative realm. However, its value in this particular application is more in the relative sense than the absolute, and, further, one-place accuracy provides the missing intelligence. This becomes obvious by referring back to Figure 2 where variations in the reliability between Items 1 and 3 range from 0.6 to 0.92, and yet two ISM's are scheduled for each. Further, it takes an inaccuracy of over an order-of-magnitude to impose a requirement to schedule an additional component for ISM.

It becomes equally evident for larger inaccuracies provided the expected reliability is better than the selected equal risk level for the function.

Where honest doubt exists as to the component level at which to equalize reliability and the optimum risk, the effects of selecting a lower risk (higher reliability) seems to have little effect on the spares level or the ultimate spacecraft weight. It simply adds a few components to the bottom of the list to spare and perhaps an extra spare for one already on the list.

CONCLUSIONS AND CONFIDENCE ASSESSMENT

While the methodology presented herein seems logical, someone is sure to bring up Murphy's Law. It is usually prefaced by: "But what if . . ." If we accounted for all the "what if's" the program would never come to fruition and the vehicle would never get off the ground. Practicality must come into play, and practicality will be implemented in a technical manner by establishing an acceptable risk level (perhaps associated with the weight limits of the booster) and then designing to meet these objectives.

Two factors influence our confidence in a given design: the length of time the components have been considered state-of-the-art, qualified, or space rated, and the reliability assessment associated with the individual components. These are, of course, closely associated, or at least should be. Given that an acceptable risk level has been set for the mission, and the components used in the design are known to be space-rated or qualified, it has been shown that small inaccuracies in the reliability estimates have little effect on the ultimate mission reliability of a maintained system.

The gains to be derived from a maintainable manned space mission concept are evident. Equally evident are the gains to be derived from a logical design of the ISM concept, particularly where performance monitoring and diagnostic elements are to be planned. Logic takes the place of unassessed chance when the logic is reasonable, understandable, and implementable. Such is the case for isolating the elusive malfunction when the plan presented herewith is given serious application.

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SPACE MAINTAINABILITY ANALYSIS OF THE APOLLO TELESCOPE MOUNT

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SUMMARY: The objectives of this study were through an analysis of an operational program (1) to verify that the concept of maintainability in space is feasible, (2) to determine what repair tasks could be performed, and (3) to evaluate the increase in program reliability attainable by performing in space maintenance and repair.

INTRODUCTION

Long duration manned missions require the use of many complex systems which must operate properly from one to several years. Reliability, therefore, is extremely important. It is too costly to attempt to obtain all of the necessary reliability through use of product improvement methods. Correspondingly, the use of redundancy to increase reliability is limited by consideration of weight, volume, and system complexity. Failures in components and systems will undoubtedly occur during long missions; as system operating time increases, even more failures can be expected. There is a practical means to attain a maximum probability of mission success, as a supplement to using highly reliable hardware and imposing tight quality control during manufacture. This is to provide a capability for performing maintenance and repair in space.

Space maintainability, as we shall call these techniques, should be established as a basic discipline upon which designers can and must rely to attain maximum program reliability. It is desirable to gain acceptance and use of space maintenance as quickly as possible. The best approach to gain this acceptance is to verify the space maintenance techniques on an operational program.

PROGRAM SELECTION

One of the earliest opportunities to implement space maintainability is in the Apollo Applications Program (AAP), a series of primarily scientific missions, a follow-on to the Apollo man lunar landing program which will make the most of Apollo hardware. One AAP payload is the Apollo Telescope Mount (ATM), a system of precision telescopes for making solar observations. The objectives of the ATM are to acquire precision scientific data on the structure and behavior of the sun for 56 days and to evaluate man's ability to perform for an extended time in a space environment the complex experiments necessary to acquire this data.

This mission will require two Saturn IB launches: one to launch the ATM, the other to launch the crew to operate the experiments.

The NASA-Marshall Space Flight Center (MSFC) is responsible for designing the structure and the support systems of the ATM; documentation was available early. The ATM was, therefore, chosen for analyses to determine whether an in-space repair capability was feasible and, if so, what repair capability should be planned.

SYSTEMS DESCRIPTION

The ATM includes six experiments and the necessary support systems to operate them and to record and transmit the data. The ATM structure is a rack on which the support systems are mounted with a barrel or canister attached inside the rack in which the experiments are housed. An Apollo Lunar Module (LM) ascent stage is permanently attached to the ATM and provides a shirt sleeve environment for the astronaut operating the experiments. (Figure #1, Artist's concept of the ATM)

Support Systems on the ATM are:

1. Electrical Power - provides power to support systems, experiments, and LM

2. Pointing Control - provides the orientation and stability necessary for successful ATM operation

3. Display - provides monitoring and control capability of the housekeeping systems and experiments

4. Telemetry - provides data transmission from the support systems and experiments

5. Command and Communication - provides RF and voice communication between astronaut and ground stations

6. Thermal - provides necessary cooling and/or heating for support systems and experiments

7. Instrumentation - monitors and measures parameters of interest of the housekeeping systems and experiments

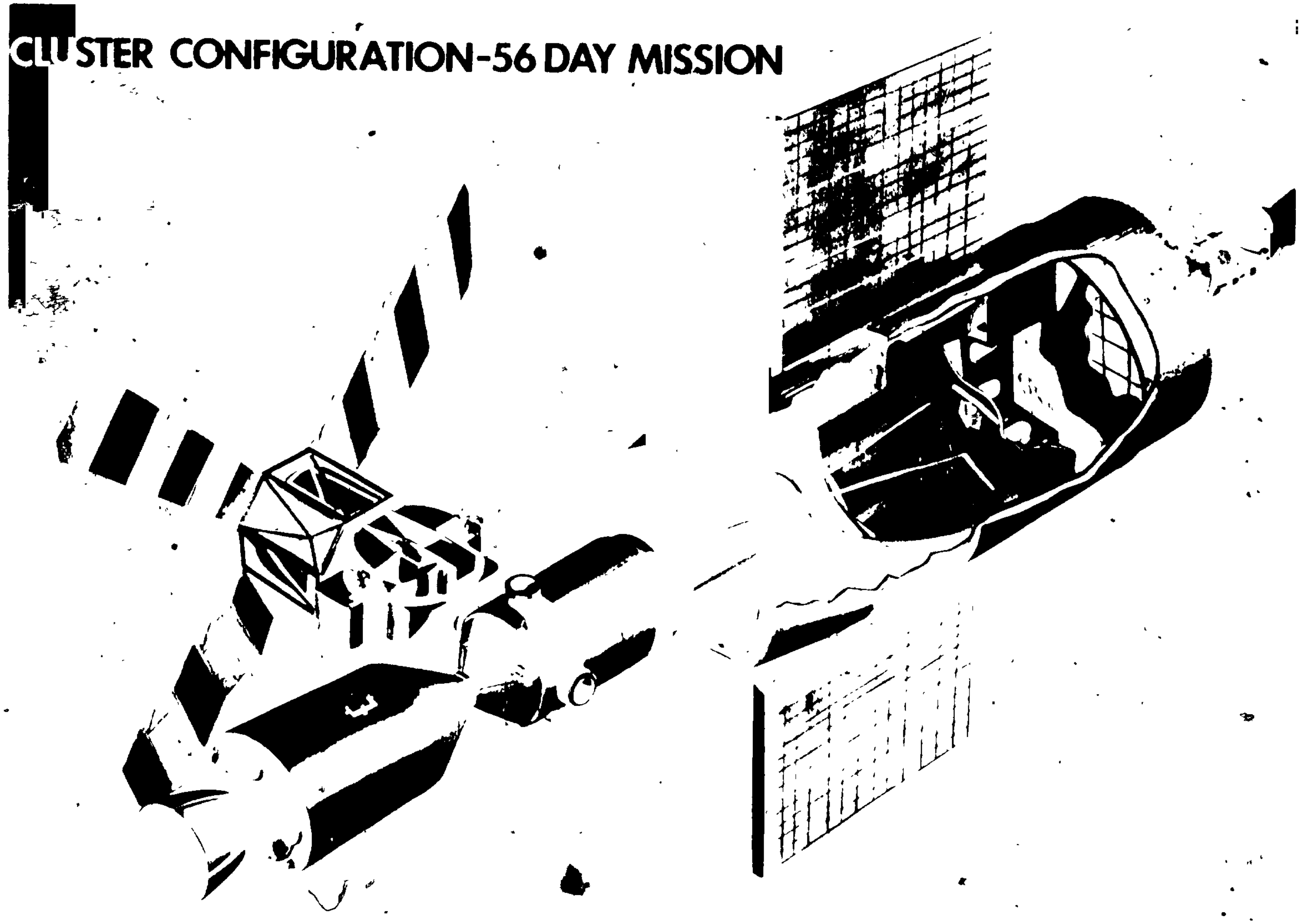
Selected Experiments are:

1. SO-52 White Light Coronagraph

2. SO-82 A & B Ultra Violet Corona Spectrograph

3. SO-54 X-ray Spectrograph Telescope

4. SO-83 Harvard College Observatory



Artist's Conception of ATM

Figure 1

5. SO-56 X-ray Telescope

6. H-Alpha Telescope 1

7. H-Alpha Telescope 2

The ATM presents three sizable limitations to the implementation of an in-space repair capability. First, all ATM systems are exposed to the space environment; therefore, any repair must be an extravehicular activity, requiring the astronaut to perform all maintenance in open space, hampered and protected only by his space suit. Second, the conceptual design was complete; preliminary design was far advanced; in some cases, hardware procurement contracts had been let. Third, the ATM had been designed to obtain the required program reliability through redundancy, not through maintainability in space.

Besides these inherent limitations in the ATM, an additional limitation was placed on the selection of repair tasks. Not only must the task verify that repair in space is feasible, but the task must also significantly increase the probability of ATM program success.

ANALYSIS

Under these limitations, a space maintainability analysis of the ATM was begun. The experiments were not considered in the analysis because the location of the experiments inside the canister made the experiments' accessibility to a pressure suited astronaut difficult. Only the support systems--electrical power, pointing control, display, telemetry, command communications, thermal and instrumentation--were considered.

As an actual operational payload was being studied, it was desirable that the analysis be completed prior to finalization of design and beginning of hardware fabrication. It was also desirable that the analysis concentrate on tasks which could most feasibly be performed and on repair which would contribute most significantly to the success of the mission.

Initially, therefore, each system was evaluated by four criteria to decide whether a more detailed analysis should be performed. These are (1) Mission Criticality, (2) Astronaut Safety, (3) Minimum Design Modification, and (4) Supporting Development.

Mission Criticality: Criticality is determined by the impact a malfunction has on the mission. If a malfunction of an item will result in loss of life or personal injury, the item is placed in criticality category 1. If a malfunction of an item will result in an abort or major degradation of mission, the item is placed in criticality category 2. All other malfunctions will result in the item's being placed in criticality category 3. No items on the ATM are rated in criticality category 1. Only items rated in criticality category 2 were considered in the analysis. As the components of the display and instrumentation system were classified in criticality category 3, these systems were not considered for repair.

Astronaut Safety: Maintenance and repair tasks which would subject the astronaut to significantly greater hazards than the inherent danger involved in performing the original ATM mission were excluded. This criteria caused two subsystems in the electrical power system to be excluded.

Design Modifications: Design for repair must not require changes that would affect the operation or function of a component. Only modifications such as a change in fasteners and connectors, installation of handles, or rounding sharp corners would be allowed.

Supporting Development: Development is required techniques or hardware which have not evolved or are still in the concept stage. Any development must be accomplished at a minimum cost, as an in-house activity, and without impacting the schedule.

As a result of applying these criteria, the electrical, pointing control, and telemetry systems were chosen for detailed consideration. Loss of the electrical power system and/or pointing control system will abort the mission. Loss of the telemetry system will result only in loss of data. The level of repair to be considered was black box replacement; to perform lower level repair as an extravehicular activity is impractical.

The matrix, figure 2, was used to collect the data for each system to the black box or component level. In some cases the design had not progressed to the point that the data was available.

After the matrix had been filled in, the data was analyzed individually and collectively by a group of engineers familiar with the systems and knowledgeable of the environment and conditions under which the repair would be performed. This analysis determined (1) qualitatively, whether the probability that a malfunction resulting in an abort or a major degradation of the mission would occur was sufficient to justify planning for in-space repair, and (2) whether the needed repair could be feasibly performed by an astronaut. As an aid in determining repair feasibility, a detailed procedure was formulated enumerating step by step each action of the astronaut for each task under consideration.

As a result of this analysis, a preliminary selection of repair tasks was made. Table 1 lists the tasks, the modifications needed on the components and on the ATM structure, the number of disconnects to remove the item, the estimated additional weight, and the required development.

Further evaluation of the repair tasks listed in table 1 narrowed the selection. Tasks listed in table 1 which were eliminated are replacement of the distributors, manual switching to the redundant control computer, and replacement of the digital computer. The replacement

ATM SYSTEM ________ SUB-SYSTEM ________

ITEM, BLACK BOX, COMPONENT	EFFECT OF FAILURE	PROB. OF FAILURE	IS ITEM REDUNDANT? DOES REDUNDANCY REQ ASTRONAUT ACTION? IF SO WHAT?	IS ANOTHER ITEM AVAILABLE TO PERFORM FUNCTION? HOW PLACED IN SERVICE?	LOCATION	DESCRIBE ACCESSIBILITY	HOW IS ITEM OR CONNECTED ELECTRICALLY MECH

TOOLS & EQUIPMENT REQUIRED FOR, ACCESS, ISOLATION, REPAIR, AND CHECKOUT.			SPARES REQUIRED TO PERFORM REPAIR			TIME REQUIRED TO PERFORM TASK	SKILLS REQUIRED TO PERFORM TASK.	HAZARDOUS CONDITIONS	OTHER
	WT.	SIZE		WT.	SIZE				

Space Maintainability Analysis Matrix

Figure 2

REPAIR	INCREMENTAL MODIFICATION TO COMPONENT	INCREMENTAL MODIFICATION TO A/TM	ADDITIONAL ITEMS REQUIRED	LOCATION	NO. OF CONNECTORS	NO. OF FASTENERS	ADDITIONAL COMPONENT WEIGHT	ADDITIONAL ATM WT	ADDITIONAL CSM/MDA WEIGHT	DEVELOPMENT
REPLACE CBR	CONNECTORS FASTENERS DE-ENERGIZE POWER HANDLE SPACE TO CONNECT TEST PLUG IDENTIFICATION FAULT ISOLATION INFORMATION	CONNECTORS SWITCH (IN CONNECTORS) FASTENERS DE-ENERGIZE POWER	REPAIR KIT TOOL FOR FASTENER OR CONNECTOR (IF REQUIRED) SPARES (2) CABLES (4)	17, 19, 21, 23	4/C BK	2 8/BK	8 LB/CBR 144	8 LB/CBR 144	100 LB	REPAIR KIT FASTENER CONNECTOR SWITCH
JUMPER SOLAR PANEL TO CBR	ELECTRICAL CONNECTOR CBR		BRACKET FOR CABLE CONN COVER CONN ADAPTOR	17, 19, 21, 23	2		1 LB-18	1 LB-18	40 LB	
JUMPER BETWEEN CBR UNIT	(1) ELECTRICAL CONNECTORS/CBR ISOLATE BATTERY DE-ENERGIZE POWER SUPPLY POWER TO CHARGER AND REGULATOR FROM NEW BATTERY	ISOLATE BATTERY DE-ENERGIZE POWER	CABLES (4) CONNECTOR COVERS		2		2 LB/CBR-36	36	40 LB	
REPLACE FUSES/ DIODES IN DISTRIBUTOR	ACCESS TO FUSES/DIODES FUSES/DIODES REPLACE-ABLE HANDLE ON COVER POWER DE-ENERGIZE PRIOR TO DISCONNECT	FASTENERS	SPARE FUSES/ DIODES FIXTURE	14		4	4	4	22 LB	REPLACEABLE FUSES & DIODES
REPLACE ASAP RECORDERS	FASTENERS CONNECTORS HANDLE TEST CONNECTOR FAULT ISOLATION INFORMATION AT TEST CONNECTOR IDENTIFICATION POWER DE-ENERGIZE	FASTENER CONNECTOR SWITCH IN CONNECTOR	SPARE RECORDER	16	2	4/ RECORDER		2 LB	12 LB	
REPLACE DC-DC (ASAP)	FASTENER CONNECTOR HANDLE TEST CONNECTOR IDENTIFICATION FAULT INFORMATION AT TEST CONNECTOR POWER DE-ENERGIZE	FASTENER CONNECTOR SWITCH IN	SPARE DC-DC	16		4	3 LB	3 LB	13 LB	
OTHER ASAP UNIT CIO MM AIO	FASTENER CONNECTOR HANDLE TEST CONNECTOR IDENTIFICATION FAULT ISOLATION INFORMATION AT TEST CONNECTOR POWER DE-ENERGIZE	FASTENER CONNECTOR SWITCH IN CONNECTOR	SPARE UNIT	16	5 (1?)		7 LB	7 LB		
CONTR COMPUTER MANUAL SWITCH CABLE CHANGE	FASTENER CONNECTOR	MANUAL SWITCH & CONNECTORS	SWITCH COVER & FOOL PROOFER	11	15	2		10 LB		
DIGITAL COMPUTER REPLACE	FASTENER CONNECTORS HANDLE IDENTIFICATION POWER DE-ENERGIZE	FASTENER CONNECTOR SWITCH IN CONNECTOR		10	2	4	3 LB	3 LB	20 LB	

Preliminary Selection of Repair tasks

Table 1

of the distributor was eliminated because of the difficulty in determining why a protective device failed. Manual switching to the redundant control computer was dropped when a design decision was made to perform this function automatically. Replacement of the digital computer was eliminated because the function of the digital computer could be performed in a degraded manner by the astronaut and because the probability of failure of the computer was low.

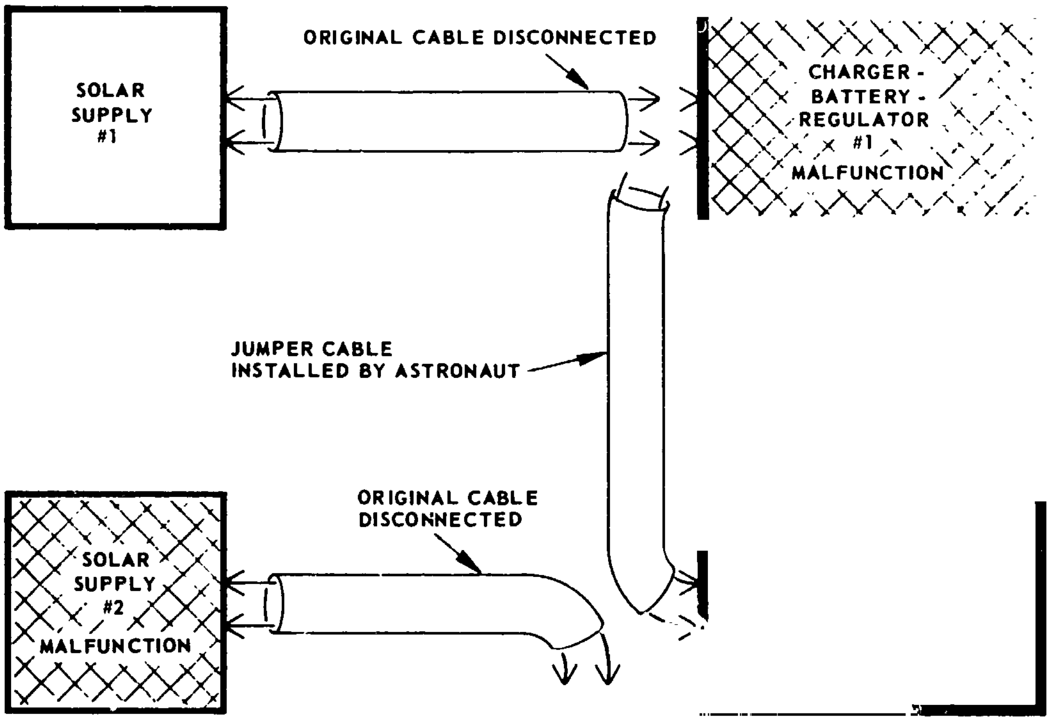
RESULTS

Four repair tasks considered feasible and worth implementing are:

1. Replacement of the charger-battery-regulators (CBR)

2. Capability to jumper between a solar panel and a CBR (figure 3)

3. Capability to jumper between



A repair of two power supply circuits that have different malfunctioning units is shown. Result of repair is to place back into operation an equivalent of one power supply circuit. The example depicts the situation of a malfunctioning CBR and a malfunctioning solar supply. Combining the good units provides one operational power supply circuit.

Jumper between Solar Panel and CBR

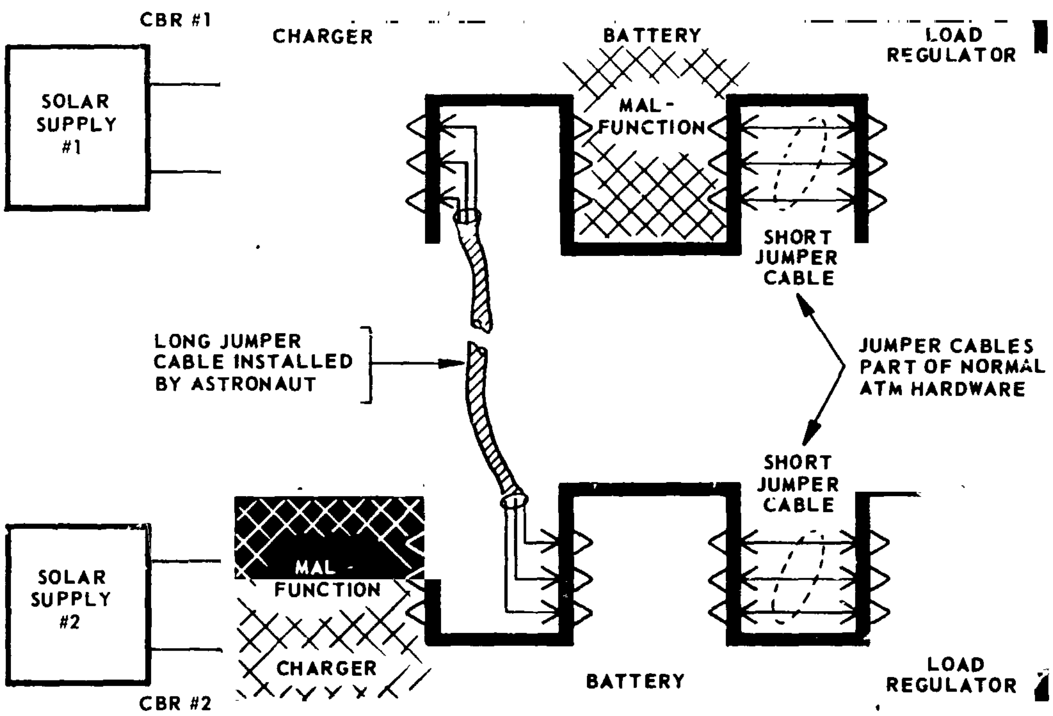
Figure 3

two major units of the CBR. (This task was later eliminated for consideration because necessary design changes were more extensive than anticipated.) (figure 4)

4. Replacement of the tape recorders. The recorders are an item of the automatic storage and playback unit, a telemetry sub-system.

Each CBR is approximately 21 x 17 x 4 inches, weighs approximately 87 pounds and consists of a battery and the electronics to recharge the battery from the solar source, to provide a constant power output to the load, and to monitor the CBR operation.

The CBR's are the heart of the power system. The original design consisted of 24 CBR's of which only 18 were required for successful



A repair of two CBRs that have different malfunctioning circuits is shown. Result of repair is to place back into operation an equivalent of one CBR. The example shown depicts the situation of one CBR having a malfunctioning charger circuit which places entire CBR into an inoperative state and a second CBR that has a failure in the battery. Combining the good parts of each CBR results in providing power to supply ATM load.

Jumper between Major Units of CBR

Figure 4

operation. The additional CBR's were designed-in redundancy in case one or more malfunctioned.

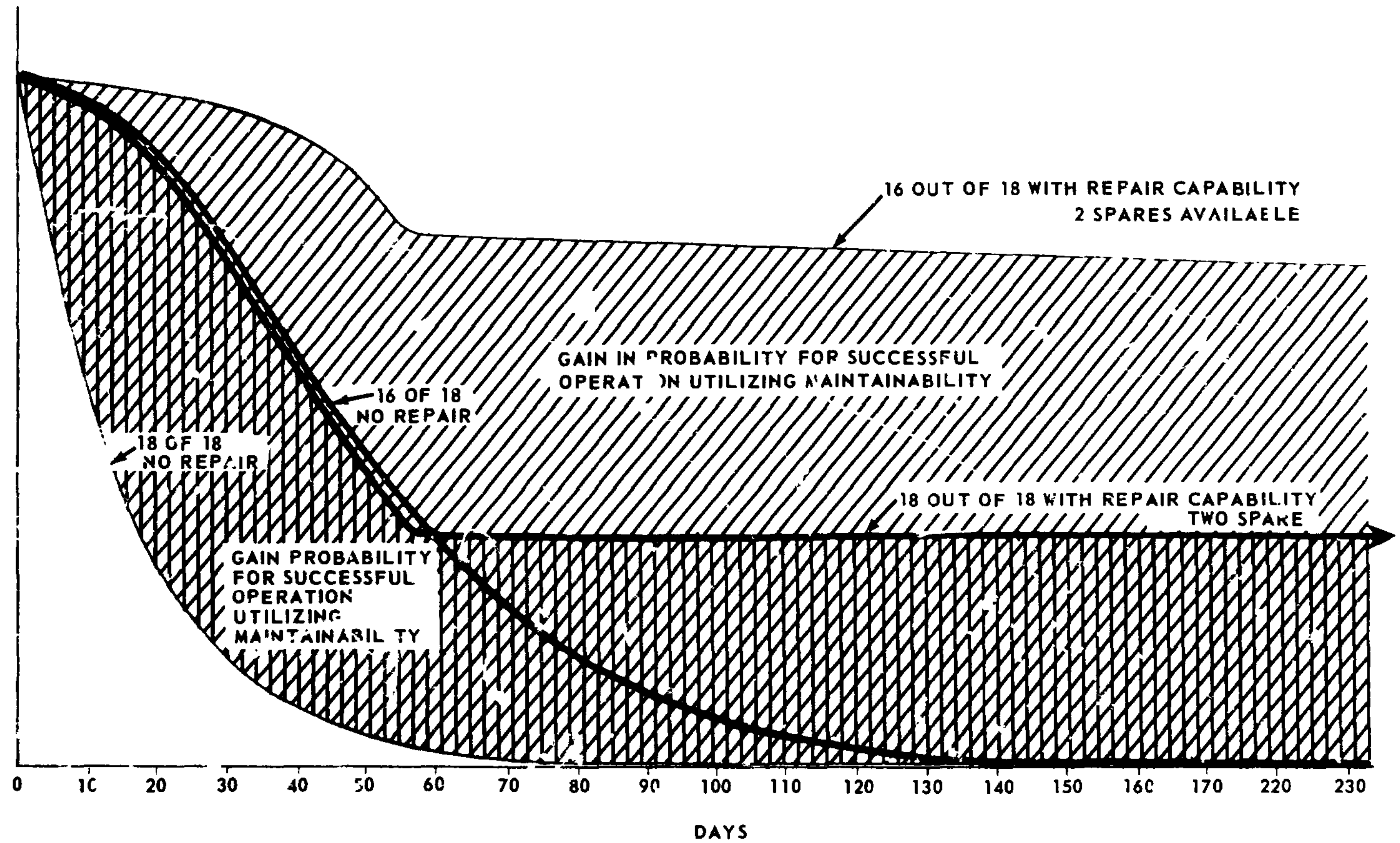
Because the weight of the ATM had to be reduced, and space was not available on the rack to mount all the required components, the number of CBRs was reduced from 24 to 18. Therefore, the original designed-in redundancy no longer exists. At present, at least 16 CBRs must operate. If the power requirements increase, all 18 CBRs (no redundancy) will be required for the ATM to operate successfully. The probability that all 18 CBRs will operate for the 56-day mission is approximately three percent. The power system is mission critical--the successful operation of all other systems depend on the proper operation of the power system.

The tape recorders are not a mission critical item. Their loss will not result in the loss of the mission, but will result in the loss of support systems and experiment data. Past history indicates that tape recorders have a high probability of failure. Because the tape recorders are not mission critical, replacement should be considered only if the capability to replace the CBRs is to be provided. The development of the tools, equipment and techniques necessary to perform the repair cannot be justified to replace only the tape recorder; but, if the CBR's are to be replaced, then the only additional requirements to replace the tape recorders are (1) to mount the recorders to be easily removable by an astronaut, and (2) to provide on-board spares.

VALUE OF SPACE MAINTAINABILITY

What is the value of the recommended repair capability in terms of increased probability of program success? To determine this value, a reliability evaluation was performed. Figures 5 and 6 and table 2 depict the benefits of space maintainability.

Figure 5 shows the probability of both 18 CBRs and 16 out of 18 CBRs operating for 224 days with and without the capability of replacement during the mission. The level of probability depicted in figure 5--that all 18 CBRs or 16 out of 18 CBRs can be maintained in operating condition by replacing a malfunctioning unit is based on two spares being available. If more than two spares are available, the level of probability car be raised. A third spare almost doubles the level of probability that 18 CBRs can be maintained in operation for 224 days. The checked area indicates the increased probability of all 18 CBRs operating if replacement is provided, whereas the striped area indicates the increased probability of 16 out of 18 CBRs operating if replacement is provided. If resupply of spares is available on a periodic basis, a desired level of probability can be sustained indefinitely. If every 60 days the supply of spares can be replenished, the probability that all 18 CBRs will operate successfully for 224 days is the same as for 56 days. Without the repair capability, the probability that all CBRs will



Probability for Successful Operation of CBRs without and with repair capability

Figure 5

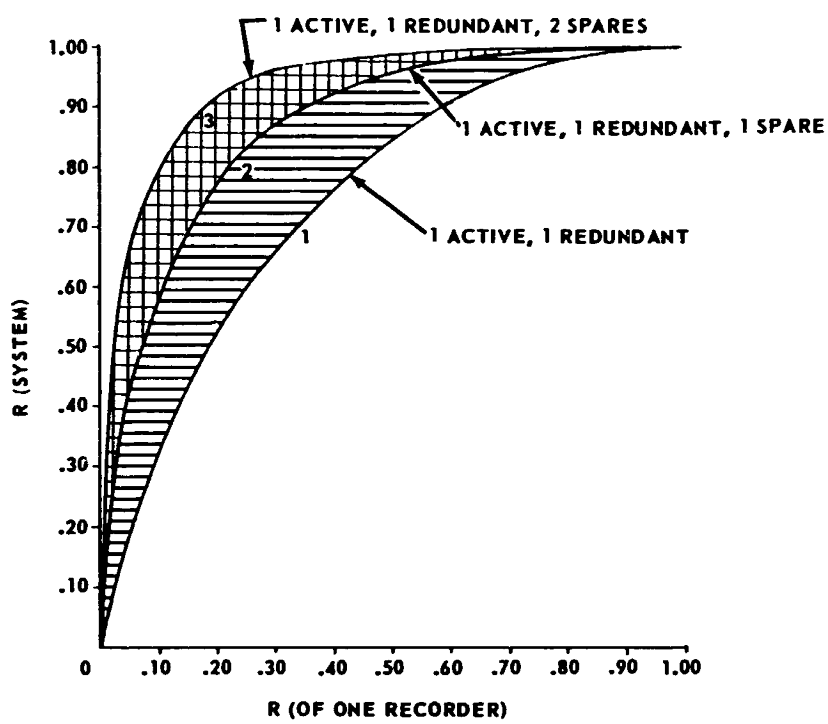
operate for even an additional 60 day is nil.

In figure 6, the striped and checked areas show the gain in reliability of recorder operation utilizing the repair technique with two and three spares available, respectively. As the reliability of the recorder to be used on the ATM was not available, table 2 was prepared giving the reliability for 56, 112, 168, and 224 days for a given mean time between failure (MTBF).

Although repair may significantly increase the probability of program success, the capability to perform inspace repair requires additional weight. For the three repair tasks considered feasible on the ATM, the necessary weight is estimated at 250 pounds on the ATM and 400 pounds on some other element, probably the Multiple Docking Adapter. This means a trade off decision between the gain in reliability through in-space repair and the extra weight which must be placed in orbit to implement the repair.

REPAIR SIMULATION

Mock-up of an ATM panel and of a mobile work station were constructed so that removal and replacement of a CBR can be simulated with a five-degree-of-freedom simulation under shirt sleeve and space suit conditions. The objectives of this simulation are (1) to check clear-



Probability of Successful Recorder Operation with repair capability

Figure 6

RELIABILITY VS. MTBF				
MTBF (HOURS)	DAYS	DAYS	DAYS	DAYS
	R(56)	R(112)	R(168)	R(224)
100	1.45×10^{-6}	2.10×10^{-12}	3.04×10^{-18}	4.42×10^{-24}
200	1.21×10^{-3}	1.45×10^{-6}	1.76×10^{-9}	2.10×10^{-12}
300	1.11×10^{-2}	1.24×10^{-4}	1.45×10^{-6}	1.54×10^{-8}
400	3.47×10^{-2}	1.21×10^{-3}	4.19×10^{-6}	$1.45 \times 10^{-}$[illegible]
500	.068563	.004701	.000322	.000022
600	.106459	.011334	.001206	.000128
700	.146607	.021493	.003151	.000462
800	.186374	.034735	.006474	.001207
900	.224625	.050456	.011334	.002546
1000	.261846	.068563	.017953	.004701
2000	.511709	.261846	.133989	.068563
3000	.638910	.408205	.261846	.166632
4000	.714632	.511709	.364961	.261846
5000	.764300	.584154	.446469	.341236

Reliability vs. MTBF
Tabel 2

ances for removal of CBRs, (2) determine optimum work position of the astronaut at each location, (3) to determine the best position of the mobile work station at each location, (4) to assess the difficulty of removing the bolts under zero G in a space suit, and (5) to determine the usefulness of a specially developed tool in removing bolts. These tests have not yet been performed.

CONCLUSION

Space maintainability can greatly augment the probability of program success. Therefore, space maintainability is a technique on which design must and should rely. But if design is to rely on this technique, then a means must be developed to select the most meaningful repair tasks and to assure that the capability to perform these tasks will exist during mission operation. To accomplish this, space maintainability should be established as a program requirement from the design concept phase through mission completion. Documentation should be developed setting forth space maintainability program provisions necessary to carry out this requirement. A document, "Space Maintainability Program Provisions," is under development to establish common requirements for applying this technique, for selection of repair tasks, and for implementing the selected maintenance and repair.

This study of an operational

program has provided insight and knowledge as to the problems, techniques, and procedures for implementing space maintainability into future programs. However, the most important contribution is a verification that program reliability can be significantly increased, from 10 to 30 times or more, through the use of in-space repair and maintenance.

SESSION III

RELATED MAN-MACHINE INTERFACE PROBLEMS

Session Chairman: Dr. E. B. Konecci
University of Texas

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HUMAN WORK PERFORMANCE IN SPACE

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SUMMARY: Methods of manual performance of extravehicular and intravehicular work in space are being investigated by means of simulation techniques in order to develop locomotion and restraint aids, equipment design data, and manual task procedures. Simulation and flight experiences to date have provided insight into problem areas for application to maintenance, repair, and assembly tasks during space-flight operations.

INTRODUCTION

How does man accomplish useful work in space, what are his capabilities, and how do we design spacecraft systems for the efficient performance of manual operations under weightless conditions? In order to answer these questions, NASA has been conducting advanced research and development on man's EVA and IVA capabilities. Conceptual studies of future space vehicles have also indicated that the economical accomplishment of extended flights requires a thorough understanding of human work performance as well as engineering data on man's biomechanical capabilities, physiological limitations, energy expenditure, and life support requirements. Therefore, in support of future missions, human factors research has been in progress for several years.

Data are being developed by the use of simulation techniques supported by the actual EVA flight experience of the Gemini program. Simulation techniques, especially neutral buoyancy, have given us a powerful tool for the detailed examination of many EVA and IVA operations and have furnished valuable insight into human work performance in space.

ANALYSIS AND DATA

Two major factors degrade the astronaut's ability to work in space: first, the lack of traction due to gravity and, second, the reduced mobility, dexterity, and tactility caused by the pressurized suit. Considerable progress has been made in providing traction and developing new operational procedures through the use of simulation techniques. Progress in developing more suitable pressure suits for working in zero gravity has been slow.

Traversal to the worksite is necessary at the beginning of any EVA task. Figure 1 shows a pressure-suited subject making egress through a transparent model of a space-vehicle air lock during neutral-buoyancy tests. Ingress-egress operations through air locks with various length-diameter ratios have been investigated to determine performance envelopes and modes of operation. Locomotion and turnaround in the air lock are accomplished by bracing between surfaces. Figure 2 shows the experimentally derived length-diameter performance envelopes between possible and impossible operation with the suit configuration shown in figure 1.

Means of locomotion about the exterior of the spacecraft have been investigated, including magnetic shoes, Velcro foot pads, ladders, handholds, and single and double handrails. Traversal by means of handrails has proven more practical than the other methods investigated but is not without difficulties. Figure 3 shows the astronaut traversing along a handrail under water in preparation for the Gemini XII mission. Operations of the subject with his body parallel to the handrail were complicated by a tendency to rotate about the handrail and by difficulty in maintaining body attitude parallel to it. In this case the procedure developed was to operate with the body perpendicular to the handrail. Body attitude in yaw and roll are then controlled by simultaneous application of force with both hands. Better control of rotation about the handrail can be achieved if its cross section is oval or rectangular rather than circular. Locomotion procedures which proved successful during the neutral-buoyancy simulation training were equally successful in space. Where handrails can be provided for traversal about the exterior of the spacecraft, they are superior to reaction propulsion devices, which are less efficient and require storage and preparation outside the spacecraft.

Two approaches are available for the performance of work tasks. One is to place the astronaut in a coplaner orbit, provide him with a reaction propulsion unit for maneuvering, traversing, and materials transfer, and provide him with torqueless tools for his operations. The second approach is to provide him with adequate traction to accomplish the task in place.

For certain tasks, only a small amount of traction is necessary. Figure 4 shows a test subject assembling components of an antenna while operating from a semi-free-floating mode. Light tasks not requiring sustained application of force can be performed effectively in this manner, since the subject can intermittently correct body attitude and traverse by grasping elements of the structure. This method has the advantage of requiring no other means of traction, but a safety line is necessary to prevent the subject from drifting from his worksite. By slow and deliberate operations, the astronaut can minimize the loads imparted to the structure. Thus, it may be possible for him to assemble lightweight structures in space which would not be able to support their own weight on the ground.

Where greater traction is required for the application of moderate forces or for the intermittent application of large forces, the astronaut's body must be restrained in some manner. Figure 5 shows the Gemini XII astronaut using a ratchet wrench on a bolt during neutral-buoyancy tests while restrained with a double waist tether similar to a window washer's belt. While using this restraint, he corrects body position by grasping a fixed object or handhold and by contacting the surface of the spacecraft with his hands and feet. This restraint system is not practical unless the astronaut can contact the spacecraft wall with his hands to control the pitch attitude of his body, and is not suitable for tasks requiring the use of two hands on a continuous basis. Excellent correlation was obtained with the neutral-buoyancy simulation of the task and the Gemini XII flight, since movements are sufficiently slow to be influenced very little by the damping effects of the water. A number of variations of the waist tether have been tried. Rigid waist tethers are generally less advantageous than the flexible straps since they restrict movement about the worksite and are bulkier to carry.

Neither of the working modes previously described are satisfactory if it is necessary to apply large sustained forces. Stronger, more rigid restraints are then necessary. One solution is to restrain the feet in some manner to approximate the standup working mode at earth gravity. A number of different restraints have been investigated, including Velcro foot pads, foot stirrups, toe traps, ski footholds, and rigid foot restraints. Figure 6 shows the use of the familiar "Dutch shoes" on the final Gemini mission. This type of restraint allows the astronaut to apply large sustained forces or work freely with both hands on more intricate tasks. He is able to control his movement backward for 90° and to either side for 45°. However, he is restricted to a fixed worksite. Weightless-simulation experiments have indicated that the traction provided by this restraint system enables a man to manipulate masses up to several hundred pounds. This restraint system, however, has the disadvantage that it must be moved each time the worksite is changed or else an additional set of restraints must be provided.

The concept of a stationary worksite is illustrated in figure 7. The astronaut assembles the components of a major structure from a fixed work platform. The structure is manipulated to new positions and reattached to the worksite as necessary to continue assembly.

For even larger assemblies in space, components could be managed from an articulated cage similar to a "cherry picker." Figure 8 shows experiments with a cage type of system from which the subject was able to work effectively under simulated weightless conditions. He braces himself within the cage and is restrained by a waist tether which prevents him from floating out.

Experiments in weightless simulations and during the Gemini flights indicate that most common handtools can be effectively used if they can be manipulated with the pressurized glove and if suitable restraints are used to provide traction. Means of retaining both the tools and fastening devices are required, and therefore the number of tools and parts should be kept to a minimum. Experiments with captured quick fasteners and self-alining devices indicate that the assembly time can be reduced by a factor of up to 10.

The transport of large and bulky masses has been explored by simulation techniques. Manual transport of masses while traversing on a handrail is difficult because both hands are required for locomotion. Figure 9 shows the test subject moving a 150-pound mass by this method. A better, but not entirely satisfactory, technique is for the subject to carry the mass on his back. Figure 10 shows a summary of energy expenditure measurements during typical transport tasks, derived from portable metabolic measuring equipment carried in the backpack. Similar measurements are being made on various EVA tasks to determine the workloads.

Since all but a small percentage of the astronaut's time will be spent inside the spacecraft, research is being conducted on the various IVA work tasks. During IVA the astronaut's capability and dexterity will be greatly improved if he is not required to wear a pressurized suit. Floating away from his worksite will be less of a problem if the walls and equipment are close enough to provide traction.

The principles for manual performance of IVA tasks are similar to those for EVA work. Figure 11 shows a neutrally buoyant test subject free-floating while working in the interior of a mockup. Many light short-duration tasks can be performed by this mode of operation. Conveniently located handrails and closely spaced walls enhance the astronaut's ability to locomote, maneuver, and maintain his work position. The free-floating work mode is not suitable for tasks requiring sustained forces or for intricate tasks requiring the use of two hands simultaneously.

As illustrated in figure 12, the use of the waist tether for IVA did not have any advantage over the free-floating mode when traction could be attained by contacting or bracing against interior walls. Small rooms or compartments are advantageous in spacecraft since they aid in working, traversing, and maneuvering. Utilization of cabin space can be more efficient than under gravity conditions since the astronaut is not restricted to an upright position.

For lengthy IVA tasks at a fixed worksite or for intricate tasks requiring the simultaneous use of both hands, foot restraints have proven advantageous. Figure 13 shows a test subject using a set of flexible rubber toe traps while working at a bench. This mode allowed him to work on intricate tasks with a dexterity similar to that at earth gravity. Velcro foot pads were not satisfactory because they tore loose when the subject leaned backward and tried to right himself.

Figure 14 shows the subject working at a control console while seated in a chair with a safety belt during simulated weightlessness. This mode was satisfactory for working at a control console for long periods of time provided the controls were within his reach. In addition, intricate assembly-disassembly operations on small equipment can be accomplished in this mode.

Figure 15 shows the IVA test subject going through an exercise routine while suspended by a waist tether stretched between two walls. This method of exercising in a weightless state has the advantage that only minor disturbances are imparted to the spacecraft, even when the exercises are vigorous.

The use of tools in IVA tasks is similar to that in EVA except that they are easier to manipulate without the pressurized gloves. Tools and objects within the cabin need to be retained when not in use but need not be tethered as in EVA, since they can be recovered.

CONCLUDING REMARKS

Since each man-hour in space will cost thousands of dollars, it is necessary that human factors data be made available to the designers in handbook form, task procedures thoroughly developed and rehearsed, support hardware tested, and time lines developed by simulation techniques well in advance of the space flight. In addition, the astronauts should be trained through the use of simulation techniques to complete each assigned EVA and IVA operation.

A 4-minute movie will be shown to illustrate some of the tasks that have been discussed in this paper.

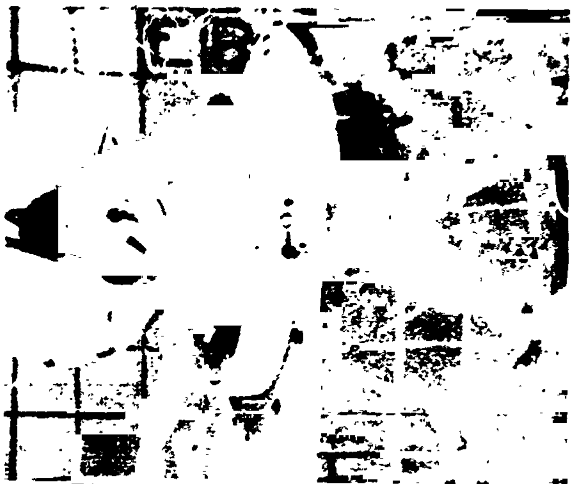

Figure 1.- Ingress - egress.

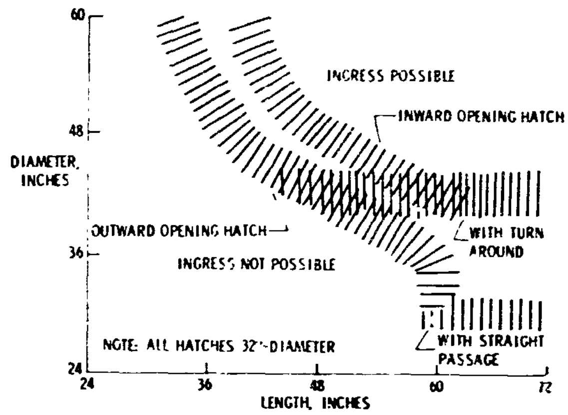

Figure 2.- Airlock dimensional limitations.

Figure 3.- Traversal on handrail.

Figure 4.- Antenna assembly.

Figure 5.- Waist restraint.

Figure 6.- Foot restraints.

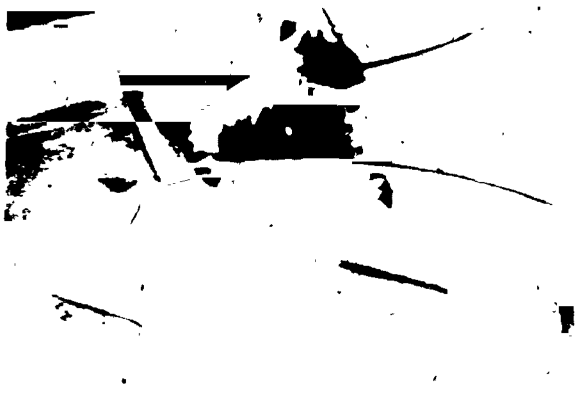

Figure 7.- Stationary worksite.

Figure 8.- Cage restraint.

Figure 9.- Package transfer.

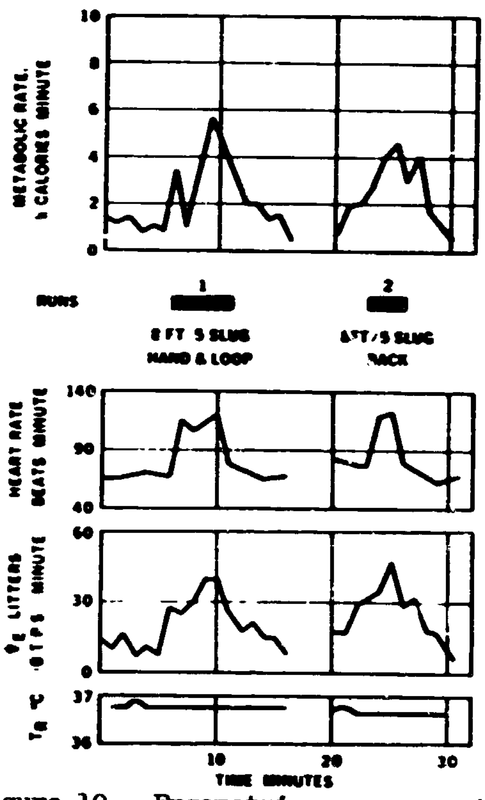

Figure 10.- Ergometric measurements.

Figure 11.- IVA free-floating mode.

Figure 12.- IVA waist restraint.

Figure 13.- IVA foot restraints.

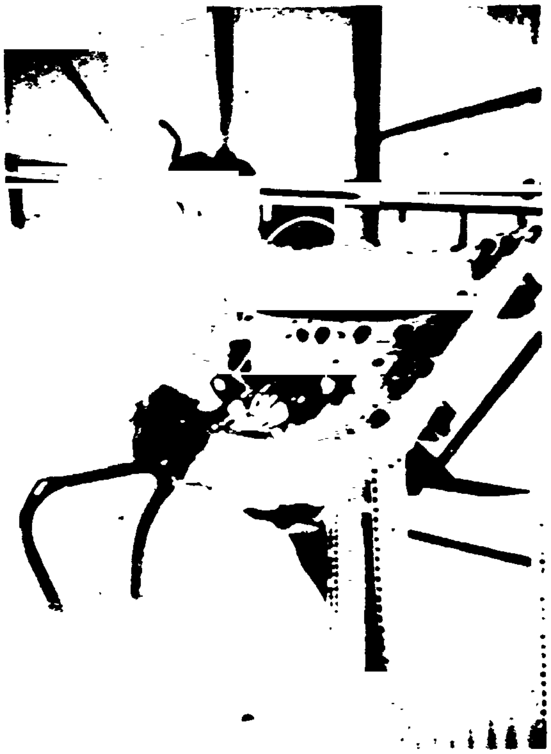

Figure 14.- Chair and safety belt.

Figure 15.- Exercise mode.

THE MECHANICS OF WORK IN REDUCED-GRAVITY ENVIRONMENTS

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SUMMARY: Research data on weightless and lunar gravity performance are considered on a single continuum of reduced traction. Apparent contradictions in the effects of reduced gravity on upper torso work and locomotion are resolved. It is concluded that reduction in traction systematically reduces the efficiency of work for all reduced-gravity conditions.

INTRODUCTION

The thesis of this paper is that work at reduced gravity is a continuum of effects that are consistent from earth gravity through lunar gravity to weightlessness. Consequently, the problems of work to be expected at lunar gravity or weightlessness can be elucidated by tests at other levels of reduced gravity, as well as by simulation of the anticipated gravity conditions. Since neither lunar gravity nor weightlessness can be simulated with perfect fidelity, the finding of systematic effects with reduction in gravity adds confidence to the general effects that might be predicted.

The space walks have demonstrated that human performance capability in space is dramatically different from that on earth. In general, the astronauts have observed that tasks in weightlessness are significantly more difficult than they are on earth. Terrestrial studies using a variety of simulation techniques have yielded similar findings. The effects of reduced gravity and reduced friction environments on human performance are generally to reduce work capabilities and increase task times.[1-10] Tests to determine the effects of these environments on energy expenditures indicate a corresponding increase in the metabolic rate for tasks performed in simulated weightlessness and other levels of simulated reduced gravity.[4,10,11] On the other hand, results of research on energy expenditures at various levels of reduced gravity [12-15] conversely indicate a reduction in metabolic rate for walking at simulated lunar and other subgravity levels. Walking, loping, and running tests during simulated lunar gravity[15-22] confirm these metabolic observations by demonstrating improvement in these activities.

The results of these experiments are summarized by Prescott and Wortz[11] and Wortz and Prescott[12] and are presented in Figure 1. As shown, energy expenditures for a given upper torso task are systematically increased as the level of simulated gravity (i.e., traction) is reduced. When no external work is involved, however, as in the case of calisthenics, reduced traction does not affect metabolic rate. On the other hand, the energy cost of locomotion is systematically reduced when traction is reduced. The systematic increase in energy expenditure for upper torso work

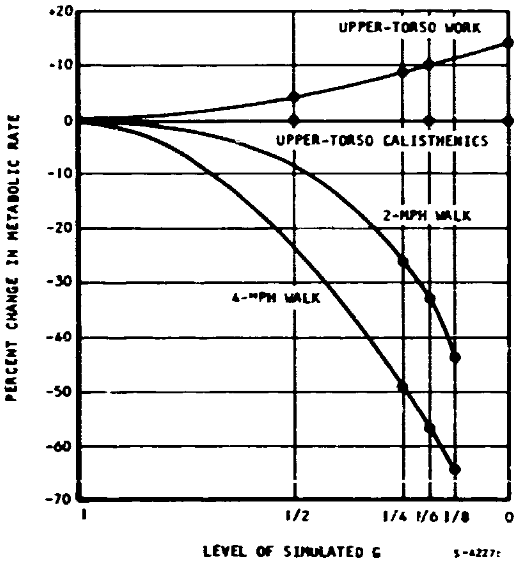

FIG. 1 METABOLIC RATES FOR VARIOUS TASKS UNDER SIMULATED REDUCED GRAVITY

as traction is reduced and the decrease in total energy expenditure for locomotion at reduced gravity are seemingly contradictory.

Discussion

A simplified view of the mechanics involved reveals that the major alteration occurring in reduced-gravity environments is a reduction in traction. Traction is partially reduced under lunar gravity conditions and completely eliminated in weightlessness. The functional utility of traction in performing work is as the primary source of the counterforce (counteractive or reactive) during work. If the counterforce is reduced, then, according to Newton's third law, the amount of work that can be accomplished must also be reduced. If the task is constant, however, and the tractive environment is altered to a point such that the normal counteractive force supplied by traction is less than the work to be done, then either the task cannot be accomplished or a supplemental source must be found for the counterforce. In reduced-gravity environments, a supplemental counterforce is supplied by several means. The most common technique for both 1-g and weightless situations is to use one arm to accomplish the task and the second arm to transmit the reactive force to a load-sustaining object, e.g., the spacecraft. Other means of accomplishing this load transmission, as reported by Wortz, et al,[10] are by using various tethering systems, wedging the body into an opening, and using the skeletal structure in combination with a tether in the "lineman's position."

The energy balance for upper torso work under all tractive conditions may be expressed by the equations below.

$$\Delta Q_m(E) = Q_w$$

or

$$\Delta Q_m = Q_w + Q_{wc} + Q_{wr} + Q_s + Q_n$$

where ΔQ_m = metabolic cost of work

Q_w = amount of energy utilized in performing useful work

Q_{wc} = energy spent in supplying the counteractive force

Q_{wr} = energy required to restore the body to the prework position

Q_s = energy stored as body heat

Q_n = net heat loss

Consequently, as traction is reduced for a given task, the muscular energy required to supply the counterforce must increase to maintain the mechanical conditions necessary to accomplish the work. In other words, the total energy required to accomplish the same task is increased as traction is reduced. Since the efficiency of work is $E = Q_w/\Delta Q_m$, the efficiency of work is reduced as traction is reduced.

In the investigation of problems of working in weightlessness, an experimental advantage occurs in that it is relatively easy to measure the action and reaction energy involved under these conditions as compared to that under earth gravity conditions. In a current program for NASA under Contract NAS 1-7571, AiResearch has instrumented task equipment to measure the energy level involved in the action and instrumented various restraint systems to measure the energy seen in reaction. If we divide the action energy, $\bar{M}_A$, into two portions--that involved in useful work, $\bar{M}_{A_{task}}$, and that consumed otherwise, $\bar{M}_{A_{other}}$--and compare $\bar{M}_{A_{task}}$ to the reactive load $\bar{M}_R$, we have a method of evaluating tasks and restraint systems. If $\bar{M}_{A_{task}}/\bar{M}_R = 1.00$ for a given task in a restraint system, the work is being efficiently performed if $\bar{M}_{A_{task}}/\bar{M}_R < 1.00$; there is work being performed that is not part of the task, and the task or restraint is inefficient. Table I illustrates some of the data obtained under this program.

TABLE I

TYPICAL MEAN VALUES OF $\bar{M}_{A_{task}}/\bar{M}_R$

Task	Type of Restraint		
	Gemini Foot Restraint with Two Tethers	Gemini Foot Restraint with Single Strap	Cage Restraint with Single Strap
Lever, push/pull	0.43	0.65	1.00
Lever, push/pull	0.55	0.59	0.89
Wheel, torque	0.36	0.81	0.85
Bolt torque	0.67	1.00	0.76
Bolt, torque	0.49	0.42	0.68

The alteration in metabolic rate for the accomplishment of a given task in weightlessness should reflect the additional energy required to supply the reactive force by means of the musculoskeletal system over and above the energy required for the task itself. The appropriate design of tethers and restraint aids to transmit the required reactive force, however, should result in metabolic levels that are no higher than those for earth gravity conditions. When aids are not available for the transmission of the necessary counterforce, the work cannot be done. For example, Wortz[23] reported that subjects at simulated lunar gravity conditions could not exert a lateral force of 15 ft-lb while pulling a cable to lift a weight.

With respect to locomotion in reduced gravity environment, several parameters such as gait, surface characteristics, and limb segment velocity are relevant. Figure 2 presents data on energy

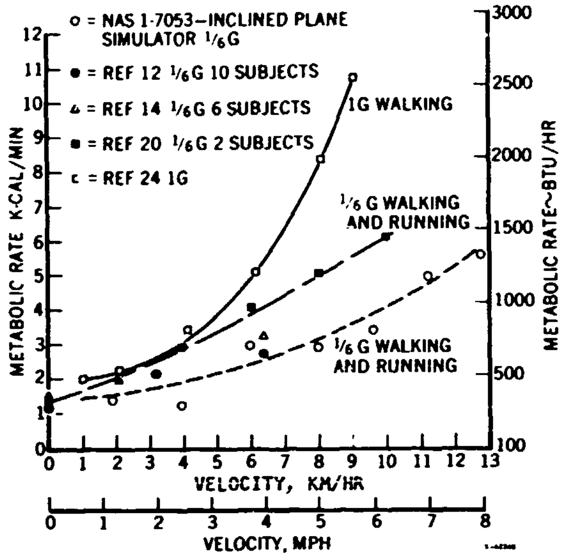

FIG. 2 ENERGY COST OF WALKING AND RUNNING IN MUFTI ON THE HORIZONTAL AT 1 AND 1/6 G

expenditures for locomotion while wearing street clothing at 1 g and at simulated lunar gravity (approximately 1/6 g) in various simulators. It is apparent that the energy expenditures for lunar gravity are substantially lower than 1-g walking data indicate. Figure 3 presents a summary of the data taken from studies of walking in pressure suits at 1 g. If these data are compared with the 1-g shirtsleeve walking data, the enormous penalties of the pressure suit in 1 g become readily apparent. For example, at 3 km/hr, the metabolic cost of walking is approximately 2.5 kcal/min. With pressure suits, on the other hand, the cost varies from 5.5 kcal/min to 10.5 kcal/min, depending upon the suit worn. At conditions of simulated gravity, Figure 4 however, the metabolic cost for locomotion in pressure suits at 3 km/hr is reduced to a range of 2.8 to 4.6 kcal/min, depending upon the suit worn. It is interesting to note that the RX-2 suit, which had the highest metabolic cost at 1 g due to its substantial weight, has one of the lowest energy requirements at 1/6 g. This somewhat indicates the preferential tradeoff for mobility over weight at 1/6 g.

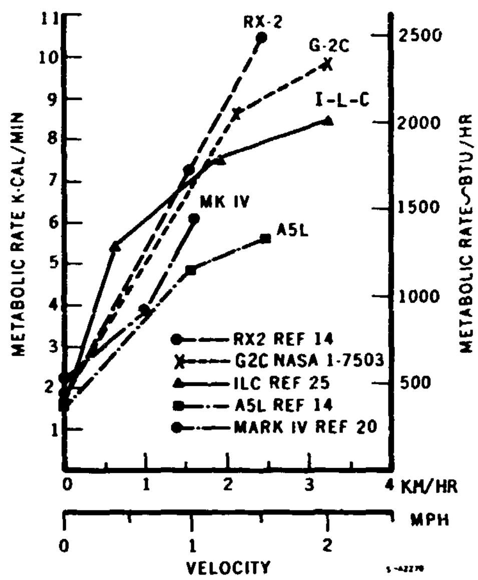

FIG. 3 ENERGY COST OF WALKING AT 1 G IN A PRESSURIZED SUIT ON THE HORIZONTAL

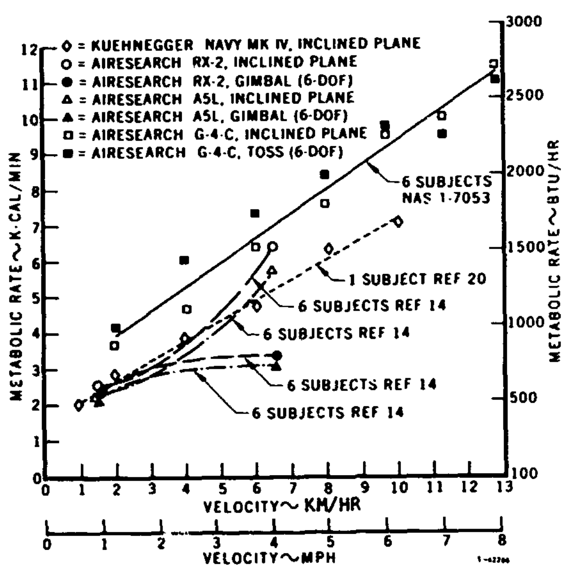

FIG. 4 ENERGY COST OF WALKING AND RUNNING AT 1/6 G IN A PRESSURIZED SUIT ON THE HORIZONTAL

Thus a simplified view of walking in reduced gravity may be to consider the task as being analogous to carrying weights while walking. As gravity is reduced, the weight carried is consequently reduced, and the energy expended for the task is similarly reduced. An effective method of testing this concept is to reduce traction, simulating reduced gravity, and to add weights to the subject to return him to 1-g weight. This was accomplished in a series of company-funded tests at Garrett/AiResearch with the six-degrees-of-freedom net suit simulator previously reported.[11,12] The independent variables were levels of simulated traction, the walking velocities, and the weights that the subjects carried. The subjects were placed in the simulator and allowed to stand on a scale, while counterweights were added until they were at the desired level of traction, as indicated by the reduced weight on the scale. Lead weights were attached to the subject for the weight-carrying tests. The mean values of metabolic rates for 10 subjects were then ascertained for 1, 1/2, 1/4, and 1/6 g; at these levels, weights were carried to return the subject to 1-g weight. The comparison with 1-g data can best be represented in terms of percentage change in metabolic rate from the 1-g condition. Figure 5 illustrates the data for the 4.0-mph walk. The figure shows that, as the simulated level of gravity is reduced, a pronounced decrease in energy expenditure occurs. When weights are added to the subjects to return them to their original (presimulation) weight, only slight increase in metabolic rate occurs, despite the substantial increments in the total weight being transported. The results shown in Figure 5 substantiate the concept that weight reduction is a primary mechanism in

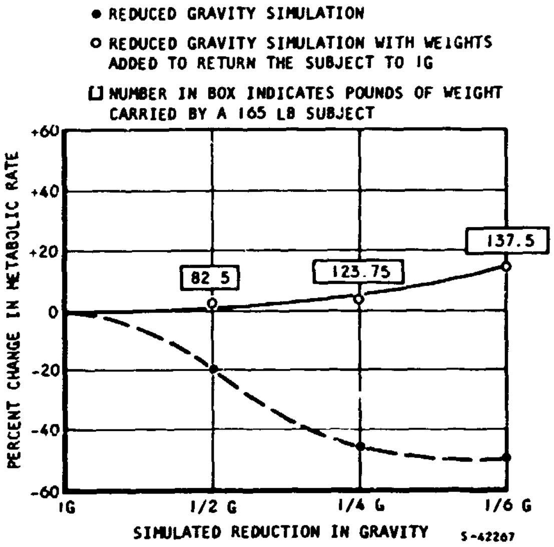

FIG. 5 METABOLIC RATES FOR A 4.0-MPH WALK AT SIMULATED REDUCED GRAVITY, WITH AND WITHOUT WEIGHTS

producing walking metabolic rates that are lower at reduced gravity than at 1 g.

Similar data were found in a current NASA program, "Man's Physical Capabilities on the Moon," under Contract NAS 1-7053. A portion of the tests in this program involved investigating the effects of carrying packs of different weights. The results of tests on the inclined plane simulator while carrying packs of 12.5, 40, and 66.6 lb with masses or earth weights of 75, 240, and 400 lb, respectively and wearing the G2-C suit, are shown in Figure 6. Little difference is seen between pack weights for velocities up to 10 km/hr. In all probability, the added traction provided by the packs compensated for the increased cost of carrying these weights. In addition, it should be pointed out that the high energy expenditure caused by the relatively immobile G2-C pressure suit masks many effects that might be seen in a more mobile suit or in mufti.

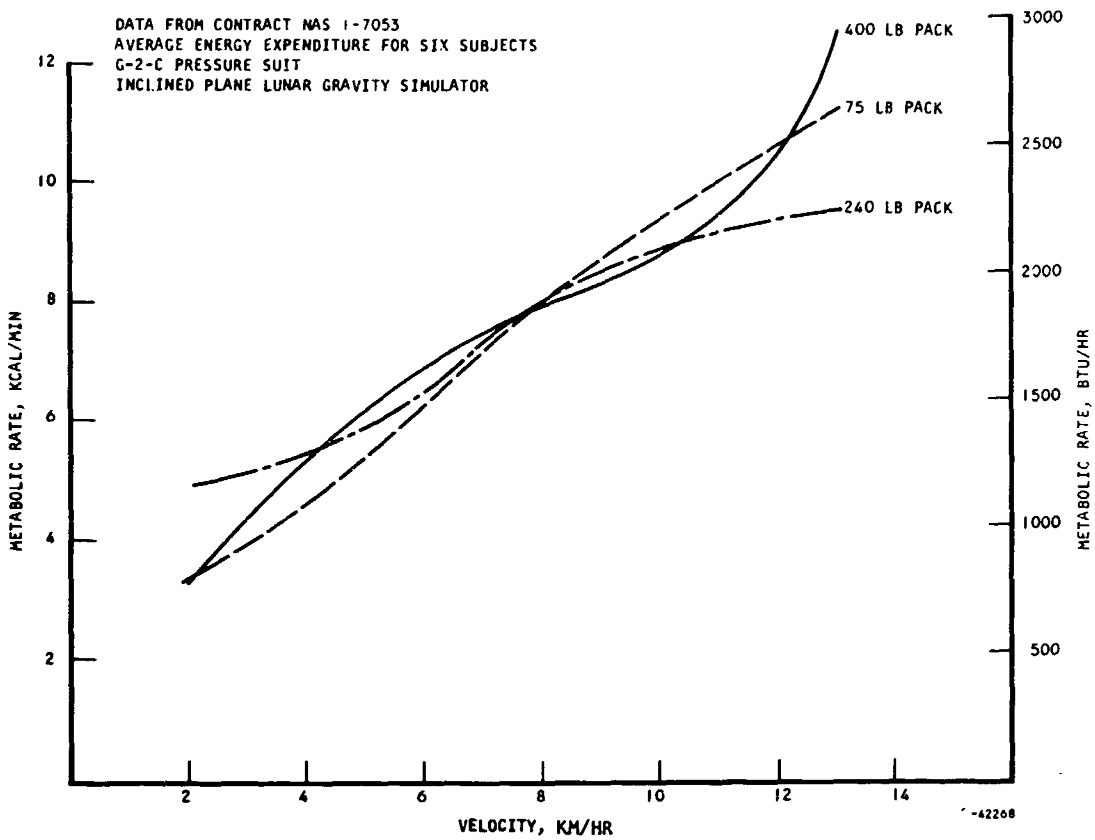

FIG. 6 EFFECTS OF CARRYING VARIOUS PACKS

The factors of traction are also significant; this is amply demonstrated by a significant decrease in the efficiency of walking, even though the total energy expenditure is dramatically reduced. This condition is cogently shown by Robertson and Wortz[14] and illustrated in Figure 7. Figure 4 had shown a dramatic reduction in metabolic rate from 1 g to 1/6 g for subjects walking while wearing the RX-2 and A5-L pressure suits. Figure 7, on the other hand, illustrates these data in terms of the lunar weight of the subjects; the metabolic rate is plotted in terms of body weight for the lunar gravity conditions. The higher cost of walking at 1/6 g (per kg on the lunar surface) indicates a substantial reduction in walking efficiency at 1/6 g. The higher cost (per kg) for the A5-L suit over the RX-2 suit indicates an increase in efficiency of work for walking with the heavier unit.

CONCLUSIONS

Nothing mechanically novel is involved in solving the problems of work in weightlessness or other reduced-gravity conditions. Because the mechanics of work at reduced gravity is straightforward, the appropriate engineering application of statics, kinematics, and dynamics should permit the design of load transmission devices that will improve the efficiency of work to allow metabolic rates equal to, or lower than, those required for the task at 1 g.

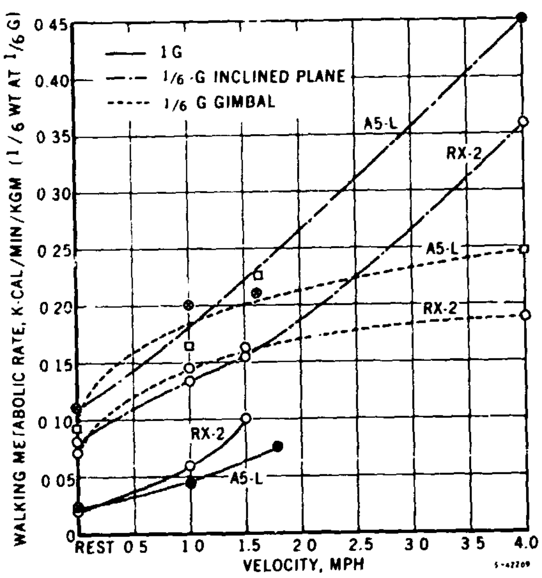

FIG. 7 METABOLIC RATES FOR WALKING IN PRESSURIZED SUITS

Other conclusions include:

a. Reduction in traction always results in reduction in the efficiency of work, whether upper torso work or locomotion.

b. The tradeoff in energetics between weight and mobility for space suits at reduced gravity is substantially in favor of mobility.

c. The metabolic cost of upper torso work increases systematically with reduction in traction.

d. The increased metabolic cost of work in weightless conditions is due to the muscular work required to provide necessary reactive forces.

e. Locomotion under lunar gravity conditions requires substantially less energy than at earth gravity.

f. The reduction in energy expenditures for locomotion at lunar gravity conditions is primarily a weight carrying phenomenon and analogous to locomotion while carrying less weight.

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MAN/SYSTEM LOCOMOTION CONTROLLER CRITERIA FOR EXTRATERRESTRIAL VEHICLE - PHASE 2

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Summary: The program objectives were to define the minimum human factors requirements for steering, navigation and caution warning systems for a small open cockpit lunar driving vehicle.

INTRODUCTION

Based on Mercury & Gemini successes, NASA is now looking ahead to post-Apollo space missions and requirements so that adequate preparation can be made for future programs. An important phase of America's space program planning is lunar inhabitation and exploration. Beyond the scientific significance of lunar investigations, the moon has often been mentioned as a possible "way station" for advanced extra-terrestrial missions. This implies permanent facilities, including scienfific laboratories, storage facilities and personal living quarters. This also implies some form of lunar transportation, for from the time the initial lunar exploration party arrives man will need some method of traversing the lunar terrain.

In anticipation of this requirement, the George C. Marshall Space Flight Center (MSFC) has been conducting design studies in support of manned lunar transportation. One proposed system is the Mobile Laboratory (MOLAB). This vehicle would be capable of traveling the lunar surface carrying two astronauts in an environmentally controlled cabin with power and provisions for a two-week scientific journey. As a more economic alternative to the MOLAB, a smaller, less refined Lunar Local Scientific Survey Module (LSSM) has been proposed. This one-to-two man, open cockpit LSSM would have a shorter range than the MOLAB but would be less complicated. It would weigh less, cost less to develop, and be available for service at an earlier date.

In anticipation of this information requirement the George C. Marshall Space Flight Center (MSFC) has been conducting preliminary design studies in support of manned lunar exploration and particularly the LSSM.

The human engineering of any lunar surface system is viewed as extremely critical. The terrain composition, visual environmental characteristics and the psychophysiological stresses to be encountered by the first humans to traverse the lunar surface are presently unknown. In addition to these factors, lunar gravity, use of pressure suits for long periods and the need to be able to respond quickly to emergencies make it important that the man-machine combination of a lunar roving vehicle system represent the most efficient design possible. In order to obtain maximum vehicle-operator performance, a highly efficient crew-station must be developed. Crew station criteria must be based on an assessment of the effects on human performance produced by conditions which will be encountered on the moon. Definition of crew-station criteria can only be achieved by real-time, high-fidelity mission simulation prior to actual lunar operations. To assure that criteria thus derived are valid, careful attention must be given to identification of appropriate variables and development of experimental design techniques and procedures, if meaningful results are to be obtained.

In order to insure relevant areas of research a comprehensive mission and systems analysis was completed to:

1. Establish baseline mission and system requirements and

2. To analyze in depth the mission-vehicle-operator-system functional interface.

The specific experimental test programs reported here were determined from the results of the mission system analysis.

1. Locomotion Controllers

No system has employed a broader range of concepts during the design development phase for lunar roving vehicles than the vehicle controller. Wheels, tillers, dual and single sticks and foot pedals have all been suggested. Right-and left-handed operations have appeared with switching functions located on the controller to operate skid steering and forward-reverse modes. They have incorporated proportional and fixed speed velocity controls; automatic and locked throttle devices; proportional closed-loop and open-loop fixed steering rates, and various integrated control, throttle and brake innovations. Neutral deadbands have varied from a few degrees to 45^o. This multitude of design concepts was initially reduced during the mission-vehicle systems analysis phase and are reported in Experiment I.

2. Navigation Aids

Although the lunar terrain is abundant in rocks, mountains and craters, it presents a degraded cue environment relative to earth stimuli. Recent lunar roving vehicle research studies have shown that without adequate navigational aids the chance for a successful mission and safe return are drastically reduced because subjects become disoriented and "lost". Additional data were obtained which show that the addition of topographical terrain maps with marked courses is still not adequate, even for distances of about 1/3 of a kilometer. Although the addition of directional gyro and odometers aided in the navigation of lunar roving vehicles, the operators still became disoriented and "lost". During an actual mission even a single "lost" condition cannot be tolerated. The conclusion reached from past experience is that it is possible to become "lost" on the lunar surface unless sufficient sophisticated navigational aids are provided.

3. Caution-Warning System

Present lunar roving vehicle caution-warning (C/W) system concepts use a transmitted tone, picked up by the Portable Life Support System (PLSS) receiver as an alerting signal, combined with visual displays. (either electro-mechanical indicators or lights). Considerable doubt still exists as to the optimum caution-warning system stimuli.

Approach The Experimental Approach consisted of static computer-based visual simulation. The visual simulator was an Air Force SMK-23 Aircraft Visual Traveling Simulator modified by the Marshall Space Flight Center, Computation Laboratory, Simulation Branch. The modified SMK-23 consisted of a television camera and model unit, and a projector-screen which presents a view of the lunar terrain to be traversed to the cockpit simulator operator. The three-dimensional terrain model is supplied on a large roller belt, 4 feet wide by 26 feet 10 inches long, with the ends attached so that it provided continuous motion. An area of lunar terrain 600 feet wide by 4000 feet long is represented in a scale of one to 150.

The television camera scans the three-dimensional belt model with an optical pickup that provides roll, pitch and yaw. To allow for the close approach of the optical(camera lens) to the terrain model, the initial element in the optical chain is only about 2 millimeters in diameters. The depth of field of the optical system permits a minimum viewing distance for terrain objects of 12 to 20 feet, and a maximum viewing distance of approximately 2000 feet (at the end of the belt).

The picture obtained from the camera and model unit is provided by a video link to an Eidaphor projector, which is a control layer-type projector that projects the image on a 12-by-16 foot screen.

The construction of the lunar terrain belt model was based on Ranger VIII photographs with a scale of 1 to 150. This scale was selected to accommodate terrain sensors and to correlate with the visual picture with the dynamics of the simulated vehicle as it reacted to the terrain surface features. Original lunar terrain area used was easily driven, presenting no real obstacles or navigation cues. Therefore, modification were made on the basic terrain map belt by adding several distinguishing terrain features in the form of hills, peaks, ranges and fields of rubble.

EXPERIMENT I
LOCOMOTION CONTROLLERS

It was previously noted that a number of candidate vehicle controller concepts have been proposed. An attempt was made, therefore, in the first experiment to select a representative sample of the various steering and throttle conditions and combinations. The various steering and throttle conditions are listed below:

Steering

1. Wheel - 240^{o} lock-to-lock power assisted.

2. Wheel - 4 turn lock-to-lock, mechanical.

3. Stick controller - Discrete Operation Mode.

4. Stick Controller - Proportional Operation Mode.

Throttle

1. Stick Controller - Discrete Operation Mode.

2. Stick Controller - Proportional Operation Mode.

A brief description of the selected controls, their mode of operation and reason for inclusion in the experimental program is given below.

1. Steering

a. Wheel - 240^{o} lock-to-lock, power assisted: Of all possible steering devices this is perhaps the most obvious one for investigation since by far the greatest number of vehicles that man operates are controlled in this manner. Thus, a maximum amount of positive transfer of training would be obtained. In addition, this concept has been proposed on several open cockpit vehicle design studies indicating popular support for this type of controller. The wheel operates in a normal fashion with a full travel of 240^{o} which corresponds to a full turn capability of the vehicle wheels. The power assist reduces the operators work load.

b. Wheel - 4 turn lock-to lock, mechanical: Inpart, the rationale for inclusion of the mechanically operated wheel in the experimental study coincides with that listed for the power assisted wheel; namely, the familiarity and previous practice possessed by an LSSM crewman with this device. Also, this concept has been suggested in previous design studies but primarily as a backup mode. This means that a mechanical wheel, with its fairly high reliability factor, could be used in case the primary steering device failed. With complete mechanical operation from steering wheel to vehicle wheel, the possibility of a steering system malfunction is greatly reduced. The fact that the system is purely mechanical relegates it to one of a backup mode however. Force requirements can be reduced by incorporating into the steering system a gear reduction device. Thus, the operating force can be brought to a reasonable level but at the sacrifice of requiring more turns of the wheel to accomplish an equivalent travel of the vehicle wheels.

c. Stick Controller-Discrete Operation Mode: The stick has long been used as a method of controlling aircraft. The Apollo spacecraft uses a stick controller. Of all control devices reviewed, the stick controller (with variations) was the one most often suggested. In addition, and in contrast to the wheel, a stick controller can be mounted to the side of the operator leaving the area in front of the driver clear for easy access to and from the LSSM vehicle. The discrete operation mode refers to the method in which the hand controller transfers the operator's inputs into vehicle wheel motions. To accomplish a turning maneuver, the hand grip must be moved to its full right or left position. Any movement less than maximum retains the control's sensors within the "deadband" region. Once the control stick reaches the full throw position, the vehicle wheels begin to turn in the appropriate direction. To terminate a turning movement the operator returns the grip to a "neutral" position. The vehicle wheels remain, however, in the turn angle occupied at the time of control neutralization and will remain in that position until the grip is moved to the opposite maximum position wherein the vehicle begin to turn back toward a "center" position. The advantage of such a design is the reducted inadvertent control actuations committed by the vehicle operator due to severe vibrations and vehicle movements that are anticipated for an LSSM type lunar vehicle.

d. Stick Controller Proportional Operation Mode: Configuration, location and operational advantages of the proportional stick controller steering device are all identical with the discrete controller design. The major difference between the two concepts is in the mode of operation. In contrast to the discrete mode which requires a full right or left displacement of the control before any vehicle wheel movement is initiated, any movement of the proportional control yields a corresponding movement of the vehicle wheels. Thus, the proportional steering controller corresponds in operation to the common automobile steering wheel. As with the discrete controller, all movements of the proportional controller are transferred to the vehicle drive train electrically so that the operator's physical work load required for vehicle control is nominal. The advantage of a proportional controller steering device is its common operating mode with other control devices which the LSSM operator has long been familiar (i.e., automobile, aircraft). The prime disadvantage lies in the ease of inadvertent control actuation by the vehicle operator (with resulting fuel consumption and constant vehicle steering corrections) due to vehicle vibrations and movements.

2. Throttle

a. Stick Controller-Discrete Operation Mode: Operation of the discrete throttle control is similar to the discrete steering device; namely, some fixed distance of control movement is required before any output signal is directed to the vehicle propulsion system. Several discrete throttle design concepts were suggested and reviewed. The one selected for this experiment provided three discrete power positions: a) O Km/hr, b) 5 Km/Hr, and c) full throttle (approximately 10 Km/hr.) The throttle stick controller operates in the following manner. At rest, the throttle remains in the vertical (neutral) position. When the throttle is moved forward to half its maximum travel (7 degrees), the vehicle accelerates to 5 Km/hr and then remains at that speed. The throttle must then be moved to its full forward position (14 degrees) at which time the vehicle accelerates to its maximum speed (approximately 10 Km/hr.). The advantage of such a design is two fold: a) elimination of inadvertant throttle movement by the operator due to vehicle vibrations and, b) the ability to hold the vehicle at a fixed speed with relative ease. This latter factor may be of real significance since a previous study conducted on MOLAB vehicles indicated that the most efficient operating speed in terms of vehicle expendables was a speed approximately one-half that of the vehicles maximum velocity. The disadvantage of such a design is the difficulty of maintaining some speed other than those provided in the discrete modes.

b. Stick Controller Proportional Operation Mode: Configuration and location of the proportional throttle control is identical to the discrete throttle design. The difference between the two is in method of operation. The proportional hand control functions the same as the standard gas pedal on an automobile. Any increase or decrease in stick displacement yields a corresponding increase or decrease in vehicle velocity. The advantage of a proportional throttle control is its commonality in operating procedure with other velocity control devices which the LSSM operator has long been familiar. Principle disadvantage lies in the ease of inadvertent control actuation by the vehicle operator due to vehicle vibrations and movement.

<u>Subjects</u> The 12 male subjects used in Experiment 1 ranged in age from 21 to 36 years and were representative of the 80th through the 95th percentile man. All subjects were right handed and possessed normal, uncorrected vision. All subjects were experienced motor vehicle operators and all were of

college level educational status. All subjects had previous space suit experience, each was in good health and throughout the training and testing period each manifested an attitude of cheerful cooperation.

Design A 2^4 x 3 factorial design with repeated measures was employed The levels of the variables were as follows: Power (proportional vs. discrete), power and steering relationship (integrated and not integrated), wheel turn radius (240° lock-to-lock, and 4 turns lock-to-lock), Steering Control (proportional and discrete) and Rate (Self paced, 5 KPH and 10 KPH). All subjects received all combination of conditions with the exception of power. Each subject received only one power condition. Each subject completed a total of 12 course segments.

Dependent Variables

Mission Variable

Distance

Time

Speed

Dynamic Vehicle Control

Human Energy Output

Independent Variable Description

Mean Driving Distance per Segment of Experimental Traverse

Mean Total Driving Time

Mean Vehicle Speed

Variability of Speed

Mean Number of Steering Movements

Vehicle Deviation from the Desired Course

Mean number of Throttle Movements

Heart Rate

Equipment Vehicle Simulation Systems: NASA/MSFC personnel effected vehicle simulation system characteristics by integration of a physical, fixed base, cockpit system characteristic of LSSM type vehicles with a computer-generated vehicle response capability predicted on a mathematical model of a MOLAB type lunar roving vehicle. Although the two vehicle designs are not identical the particular MOLAB model choosen for this study was very similar to proposed LSSM configurations in terms of motion equations, suspension system design, wheel base, number of wheels, steering operation (Acker man) and overall vehicle design. As a result, vehicle response characteristics were sufficiently developed to provide realistic simulation of the lunar driving tasks for an open cockpit vehicle.

The simulator cockpit structure is of modular construction designed to permit ready exchange of the steering and throttle control units without interfering with the remaining units (i. e., control panel, seats etc.) of the simulator. The static simulator base provided a mounting platform for the control panel and requisite steering and throttle controls required for each experiment. Two adjustable aircraft-type seats with associated restraint harnesses, were placed in normal positional relationship to the display/control systems.

With each steering-throttle control combination seating was arranged to permit maximum comfort for each vehicle operator and was adjusted prior to each test trial. Control of eye position was not required since the vehicle is an open cockpit concept with the subjects field of view, regardless of seat position, limited by the physical dimensions of the simulator projection screen rather than by the subjects eye position. During an experimental trial the subject occupied the left seat, the experimenter the right seat. Thus, the experimenter was clearly able to observe both the subject and the image screen simultaneously.

The display panel used for all steering throttle test trials consisted of a compass, high-low gear indicator, high-low gear switch, a 0 to 20 kilometer-per-hour range speedometer and a forward and reverse switch.

The stick control was mounted on a pedestal which extended from the subject's seat forward to a point beyond the stick. The pedestal top was 27-1/2 inches from the simulator floor, a height which permitted the subject to comfortably rest his arm on the pedestal during the experimental trials. The grip was located 18 inches to the right of seat centerline and approximately 19 inches forward of the seat backline. These dimensions were selected based on previous design studies for open cockpit vehicles and placed the grip controller in a position such that the operator arm fatigue was not a factor during the experimental trails. The grip was a standard-form finger aircraft-type design with a maximum 14^{o} travel in the forward direction and 10^{o} in the backward direction from a straight-up neutral position. A backward motion of the stick served as a brake only. All vehicle velocity control was regulated by forward motions of the stick regardless of whether the vehicle were in a forward or reverse mode of travel.

When the stick served as the steering controller, the maximum angle of travel was 20^{o} to either side. Thus, in the discrete mode of operation the control had to be moved through a full 40^{o} angle of travel to terminate one turning motion and initiate another in the opposite direction.

The steering wheel was located directly in front of the subject on the seat centerline and tilted 45^{o} up with respect to horizontal. A force of 6 pounds at the rim was required to turn the 14 inch diameter wheel under all conditions in which the wheel was used. This force corresponds to that proposed for actual LSSM type vehicles regardless of whether the system were straight mechanical linkage or power assisted. The wheel was constructed so that a minor adjustment could alter the wheel from a 240^{o} full turn design to one requiring 4 turns lock-to-lock to accomplish the same vehicle wheel turning capability. The relationship between wheel location and seat position was established for each subject to provide individually selected positions of maximum comfort. This universally resulted in a seat position which provided a slightly less than fully extended arm when the subject placed his hand on the wheel at the 12 o'clock position. This provided the subject with sufficient clearance between himself and the wheel while remaining within the space suit extention capability.

Since the objective of the experiment was to evaluate various steering and throttel control designs during simulated lunar driving operations it was important to eliminate or hold constant other driving tasks which might mask the effect of the locomotion controller on driver performance. One of the potentially greatest sources of driver confusion was that of navigation. Thus, it was considered important to provide the vehicle operators with sufficient test course identification so that they would have no difficulty in following the test track. Black lines were drawn on the terrain belt to identify the test traverse and small sponge rubber blocks, (each marked to identify to the driver on which side of the block the course traveled) were stationed at each turn on the test course.

Test Course Selection Three test courses were selected which were to serve as the traverses over which each of the subjects had to drive the LSSM vehicle. The courses were established with the intent of being equal in length and difficulty. To verify this fact each course was measured and found to be approximately 1300 meters in length. A very experienced non-subject vehicle operator then drove each course at maximum vehicle speed (with the integrated controller) and covered each in approximately eight minutes. Thus, it was established that the three courses were approximately equal in length and difficulty. Although no attempt was made to equate right and left turns, final course selection yielded a close approximation to this standard. Courses were coded as A, B, C to facilitate experimental design development but the subjects were provided with no maps or other course identification system.

The communication circuits were simple in nature and designed to provide sufficient communication channels to facilitate study execution without distracting subject attention. The only individual in communication with ten subjects other than the experimenter was the suit operator and this was for safety reasons. However, the suit operator was instructed not to speak to the subject once the experimental trial had begun except in the event of some physiological emergency.

Pressure Suit Systems The pressure suits used were U.S. Navy issue high-altitude Mark IV full pressure suits fitted with Mercury gloves. Pressurized runs were made at 3.5 psia with suit temperature at 75° F $\pm$ 5°. Suit pressure and temperature were controlled at a console panel located just outside and to the left of the vehicle simulator. Thus, the suit operator was in visual as well as verbal communication with the subject. The console panel contained flow regulation devices for control of pressure and adjustment of temperature sensing was achieved via insertion of a theristor at the outlet of the suit gas flow and read directly from a display.

Computer Support General Purpose Analog Computers, EA1231R, & EA1221R, and AD256 were used for modeling and simulation of the dynamic responses of the proposed vehicle systems. Supporting equipment and expansion groups to implement the capability included a DR-20 Resolver, CCC Digital 116, amplifiers and attenuators. A detailed description of computer operation is provided in another report (ref NAA report) and will not be repeated here.

Simulator and Program Verification Prior to the initiation of any experimental trials, a series of calibration and validation exercises were performed to ensure the attainment of maximum possible simulation validity and data reliability. The simulator performance characteristics in terms of vehicle speed, turning rate and radius, locking, etc., were established to correspond as closely as possible to expected LSSM vehicle dynamics.

Runs were executed using all control and calibration systems to check the input characteristics of the computer in terms of amplitude, rates, times, accuracy of responses and simulation veracity. Various terrain areas were used to examine the effects of topography on input/output ratios of the computer and their readouts. The primary purpose of these efforts was to increase the probability of collecting reliable, accurate and valid data with which to compare subject performances as a function of steering and throttle design concepts.

Once the test courses had been established it was necessary to verify that each was sufficiently marked to guarantee that test subjects could follow them without becoming lost. This was necessary since it was important that subject performance differences due to locomotion controller design not be confounded by extraeous data acquired while the subject was covering lunar terrain not included in the test course. Accordingly, twelve non-subject operators drove the vehicle over one or all of the test tracks to establish that a naive subject could successfully follow the selected traverse. Results indicated that the courses were sufficiently identified and thus ready for the experimental test runs.

Procedure The procedures may be divided into three basic categories according to the goals of each: 1) training of subjects and operators, 2) testing and 3) subject calibration.

RESULTS

1. Locomotion Controllers

Wheel vs. Hand Controller - When averaged across all other conditions, the hand controllers were slightly superior to the wheels in terms of average time to drive a segment. Subjects when steering with a hand controller, drove faster and varied less than did those who steered with a wheel.

Hand Controller, Proportional vs. Discrete Steering - The proportional stick yielded the fastest trial time with least variability per segment of any condition. The discrete stick yielded longer trial time and required more power movements and steering movements.

Wheel, 240° vs. 4 turns lock to lock - The 240° wheel resulted in the second fastest trial time per segment of any condition. In comparison with the 4 turn wheel, the subjects using the 240° wheel drove faster and with less variability. The worst condition was the 4 turn wheel.

2. Power - Proportional vs. Discrete

The proportional power condition was slightly better than the discrete power condition when averaged across all other conditions with the proportional power condition there were fewer gross power movements and more fine power movements than with discrete power.

EXPERIMENT II

NAVIGATION REQUIREMENTS

Previous studies utilizing the SHK-23 Lunar Driving Simulator have shown that subjects become disoriented ("lost") after a brief period of driving. This is because the lunar environment presents a degradation of visual cues relative to familiar earth stimuli. Since even a single lost case involving an astronaut on the moon would be intolerable, an attempt was made to test several navigation aides with a view to determining minimum requirements for completely safe navigation.

The navigation aides investigated were:

1. Map and compass (Window)

2. Digital readout of Eastings - Northings units, map and compass (DIGITAL).

3. X-Y plotter, map and compass (X-Y plot).

4. Aides #1, 2, and 3 without the compass (COMPASS FAIL).

All navigation aides were examined under three separate ordered speed conditions:

1. Full throttle (approx. 10 KPH)

2. Half throttle (approx. 5 KPH)

3. Self-paced

The design was a 3x2x3 factorial design with repeated measurements. The levels of the variables were as follows: Source of Position Information (window only, digital readout, X-Y plot), compass (operational, failed), speed (full throttle, half throttle, self-paced).

Each subject was tested under all combinations of conditions. This involved 18 runs for each subject (i.e., 6 runs for each of 3 sessions). The order of presentation of positive information was selected to maintain approximate balance of order effects.

The dependent variables for this study were the same as those for the locomotion controller study, with two changes:

1. HR data was not taken as the subjects were not in space suits and the physical task was identical for all runs.

2. A running count of all "lost" cases was kept for each run.

Lost criteria:

a. S was either to the right or the left of his assigned course and made a steering correction of at least a 2-second duration which compounded his error.

b. S was on course, but passed the point where he should have turned to the extent that he couldn't get to his new course leg without either encountering an obstacle or running the vehicle into the limits of the terrain belt.

c. S turned either too early or too late, making a crash unavoidable.

d. S told E that he didn't know where he was.

The apparatus for this study was the same as for the locomotion controller study, with the following exceptions.

1. Only the integrated hand controller was used (in the proportional steering, proportional power mode).

2. Maps were made by constructing a polaroid photograph mossaic of the terrain belt on a scale of 1800 to 1. These maps were utilized under all experimental conditions.

3. A panel containing a digital readout (Eastings - Northings), compass and speedometer (0-20 KPH) was constructed and placed 40° to the right of centerline just forward of the integrated locomotion controller.

4. An X-Y plotter (variplotter) was placed directly in front of the S in place of the steering wheel used in the previous study. The maps for the assigned courses were placed on the plotter for all conditions, with the plotter active only during the X-Y plot condition.

The data from this study are currently undergoing analysis. Preliminary indications are that X-Y plot is the "best case" in that no S ever became lost while driving with this aide. The digital aide is slightly better than the map and compass alone, but both these aides proved inadequate as the S's became disoriented and lost an average of once every 300 meters. Further research is now underway investigating a more sophisticated digital device. The X-Y plot condition also seemed to be best case in terms of speed and accuracy of completing assigned driving tasks.

EXPERIMENT III

CAUTION/WARNING

Present lunar roving vehicle caution/warning (c/w) system concepts use a transmitted tone combined with visual displays as an alerting signal. Considerable doubt still exists as to the optimum caution/warning system stimuli.

An investigation of the problem should attempt to answer the following questions:

1. What should the physical quality of the altering signal be?

2. Is the signal compatible with the Apollo caution/warning system?

3. Can the crewman differentiate between the signal representing vehicle malfunction and the signals used to indicate a PLSS malfunction?

4. What should the duration and reset concepts be?

5. What specific malfunctions should be displayed, and how?

6. Should information, in addition to the basic malfunction notice, be provided (verbal auditory message explaining malfunction and correction required)?

The caution/warning devices investigated were:

1. Voice - a verbal warning.

2. Light - an annunciator light.

3. Tone - a 750 cps - 2000 cps signal (at 60 db, alternating at 1/2-second intervals) coupled with the light.

In addition to the above devices for alerting the S to a vehicle malfunction, there was a 1500 cps signal (frequency varied 20% 15 times per second to give it a warbling sound) which was used to identify Portable Life Support System (PLSS) malfunctions recorded. This tone is currently being used at MSC, Houston as the PLSS warning.

The design was counterbalanced so that each S drove under all caution/warning conditions (voice, light, tone) at two different ordered speeds (8 KPH, self-paced). Thus, each S drove six different courses, one under each combination of caution/warning condition and speed. The PLSS warnings were presented randomly under all conditions.

In order to make the S's task realistically difficult, communications were maintained with an "astronaut" in the lunar module. At various times during the run, the subject was asked questions such as "Explorer, this is Home Base. What is your course?" or "Explorer, this is Home Base. What is your speed?" The S's tasks for each run were:

1. To drive the assigned course safely and as accurately as possible while maintaining the ordered speed.

2. To recognize and react to vehicle and life support malfunctions quickly and accurately.

3. To answer questions from the command module quickly and accurately.

The dependent variables for this study were the same as those for the navigation study, with 3 changes:

1. "Lost" cases were not a factor because all S's used the X-Y plotter, map and compass for navigation and knew at all times where they were.

2. Reaction time to vehicle and life support malfunctions were measured to the nearest 0.1 second.

3. Accuracy of response was recorded.

The apparatus for this study was the same as for the navigation study, with the following exceptions:

1. The digital readout panel was removed as all navigation was accomplished with the aide of the X-Y plotter.

2. A panel containing a series of eight caution/warning annunciator lights, compass and speedometer was placed just forward and to the left of the X-Y plotter 29° off centerline. (this position is within the normal 30° cone of vision).

3. A panel containing two 1 1/4 x 3/4 buttoms labeled "RESET" and "PLSS" was placed at a 45° angle to the left of the X-Y plotter, within easy reach of the subject. By pressing the appropriate button, the subject terminated the c/w signal and stopped the timer which was automatically started at the onset of the signal.

4. A panel containing a series of momentary contact noise-free switches was installed alongside the experimenter in such a way as to be out of the subject's sight. With this panel, the experimenter instigated the desired vehicle and life support malfunctions and automatically started the timer.

5. A variable frequency signal generator was used to generate the desired vehicle and life support malfunction tones.

The data from this study are currently undergoing analysis. Preliminary indications are that reactions to the tone condition are slightly faster than reactions to verbal warnings on the light alone. This difference is not great however (about 0.3 sec.). As the average reaction time was approximately 1.6 seconds, it is worthy of note that on several occasions, under the light condition, the subjects missed the warning for as long as 13.5 seconds. The subjects later said that they were preoccupied with the driving task and "didn't notice" the light. This phenomenon of not noticing did not occur under the tone or verbal conditions and may have implications for the safety of the astronaut during an actual lunar mission.

CONCLUSIONS AND RECOMMENDATIONS

1. Locomotion Controller Experiment

The optimal controller/power combination appears to be the proportional hand controller with proportional power. This system, however, was not greatly superior to the 240° wheel with proportional power. In that a wheel if not the primary system is the backup system in all vehicles proposed at this time, the 240° wheel might be the best alternative.

The 4 turn wheel proved to be a very marginal system both in terms of time to drive the course and subject energy expenditure. It would seem on the basis of this study that fewer turns lock-to-lock should be required.

2. Navigation Requirements

It appears that use of the X-Y plotter for navigation would result in maximum safety for the astronaut and this is of prime importance. The plotter, however, requires additional equipment including a small computer and associated circuitry. This adds to both hardware weight and complexity, which in turn leads to a higher probability of equipment malfunction. Research is now in progress to examine the utility of a more sophisticated digital readout device. If it can be shown that this new equipment is adequate for the task, its reduced weight and low complexity might make it more desirable than the X-Y plot as the primary navigational aide. It might also be considered as a backup to the X-Y plot if the research indicates that it is not sufficiently accurate for primary use.

3. Caution/Warning

It appears that use of the 750 cps/2000 cps alternating tone coupled with the light is the most rapid and consistent method of notifying the astronaut of a vehicle malfunction. The study has also shown that the astronaut would have no difficulty in distinguishing this tone from the 1500 cps "warbling" tone associated with life support malfunctions and thus is highly compatible with existing Apollo hardware. In addition, the vehicle caution/warning tone could be produced by the same equipment used to generate the life support warning tone with minimum modification and would thus minimize hardware weight, complexity, and development costs.

SELF-ROTATION OF ASTRONAUTS BY MEANS OF LIMB MOVEMENTS

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SUMMARY: When an astronaut is in a state of free fall (weightlessness), any motion of one part of his body relative to another causes a change in the state of motion of the entire system. Some investigations concerned with the exploitation of this fact for the development of practically useful self-rotation techniques are discussed.

INTRODUCTION

Gymnasts, trapeze performers, and divers offer living proof of the fact that relative motions of body parts can have profound effects on the over-all attitude motion of the human body in free fall and, furthermore, that human beings are capable of performing certain relative motions of body parts in such a way as to repeatedly obtain well defined results. This observation leads rather naturally to the thought that it may be possible to use predetermined limb motions to control the orientation of an astronaut in space. However, the attitude control problems faced by the astronaut are so different from those encountered by gymnasts, etc., that their solution cannot be found merely by applying old and familiar techniques. Furthermore, new techniques cannot be devised experimentally in conventional ways, because gymnasts, etc. do not remain in a state of free fall for sufficiently long periods of time. Hence one must ultimately resort to orbital experiments, but the value of these can be enhanced considerably by first studying certain questions analytically. For example, it is obvious that the effect of a given limb motion will be modified by attaching weights to various points of the body--and it is preferable to explore questions connected with the optimum size and placement of such weights analytically, rather than during an orbital flight.

With these ideas in mind, a number of my students and I have sought answers to the following questions:* (1) How can an astronaut in free fall move various parts of his body relative to each other in such a way as to bring about both a desired change in orientation and a prescribed disposition of body parts ? (2) Starting from rest, how can one move the limbs to obtain relatively pure pitch, roll, and yaw motions, and what can one do to make these maneuvers as effective as possible? (3) How do limb motions and propulsion devices interact ? A formal answer to the first question was given in a recent paper.[1] The second and third questions are discussed in the sections that follow.

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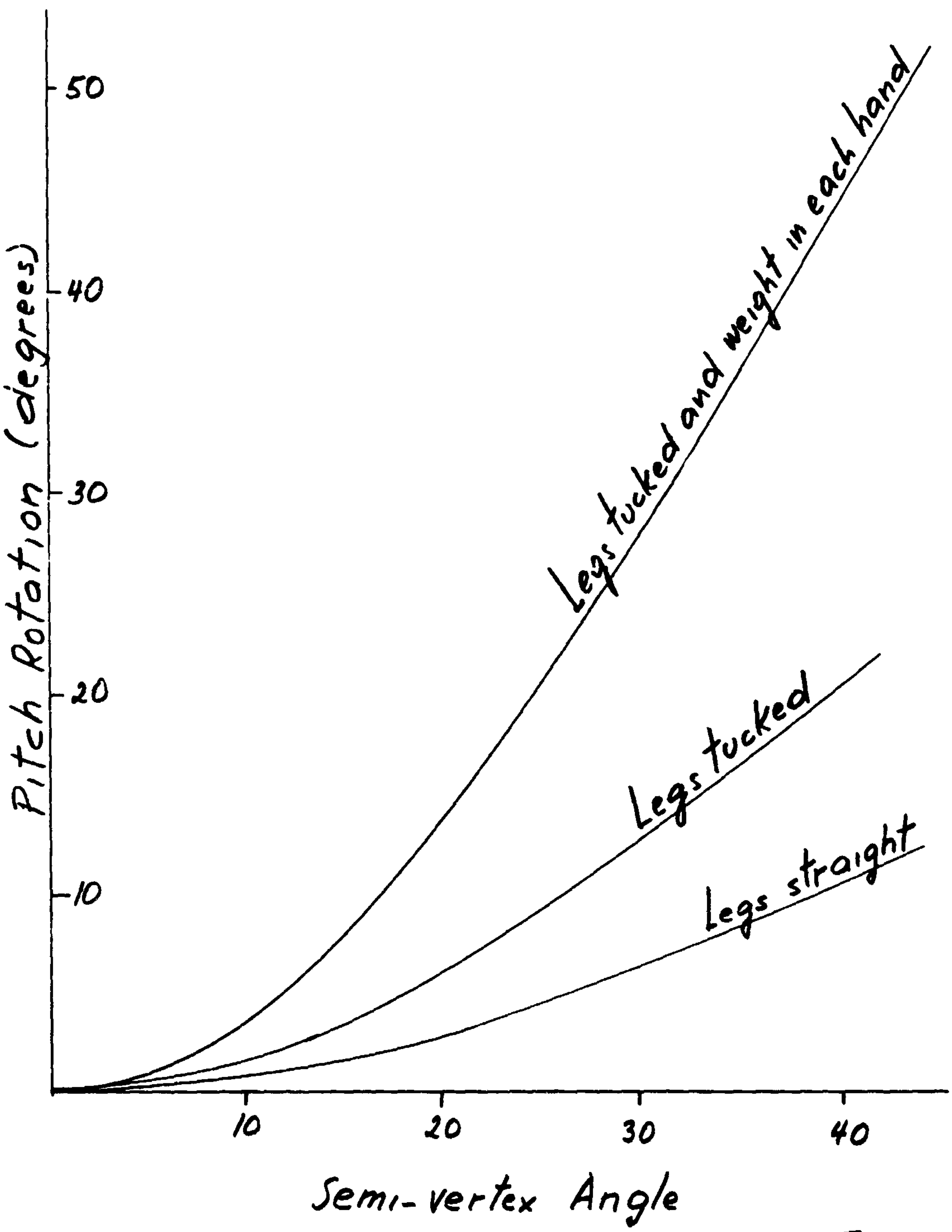

Pitch Rotation (degrees)
50
40
30
20
10
Legs tucked and weight in each hand
Legs tucked
Legs straight
10
20
30
40
Semi-vertex Angle

Fig. 1

fifty degrees. A practical resolution of this dilemma is to follow up the roll maneuver with correcting pitch and yaw maneuvers, both of which are essentially "pure."

YAW MOTION

A cyclic, two-part maneuver that produces yaw reorientations and that may be performed either with the legs or with the arms was suggested by James Jones of NASA, Ames Research Center, and proceeds as follows: With the legs placed as when taking a step in walking, i.e., one to the front, the other to the rear, each leg is moved on the surface of a cone whose axis is parallel to the yaw axis and which has a semi-vertex angle equal to the semi-angle of leg spread; and each leg is permitted to rotate about its own axis in whatever way is necessary to avoid twisting. The second part of the maneuver consists of bringing the forward leg to the rear, and vice - versa, and the entire operation then can be repeated. Fig. 3 shows the reorientation per cycle obtained in this way for various semi-angles of leg spread. (A similar maneuver can be performed with the arms, and its effectiveness has been demonstrated by gymnasts.)

A second method for obtaining yaw motions is to bend the torso relative to the legs successively in various planes passing through the yaw axis, which is the human analog of the righting maneuver performed by a cat that lands on its feet after being released from rest in an upside-down position. (A detailed discussion of this topic will appear in a forthcoming issue of the International Journal of Solids and Structures.) One cycle of this maneuver can produce a reorientation of as much as one hundred and eighty degrees, and this has been demonstrated experimentally. However, the body movements required to produce such a large reorientation may be so arduous as to render them unsuitable for execution by an astronaut, particularly one wearing a space suit. Ultimately, only an orbital experiment will permit one to assess the relative advantages of this and other yaw maneuvers.

PROPULSION DEVICES

The simplest conceivable propulsion device is one consisting of a single thruster that exerts a force of constant magnitude in a direction that is fixed relative to the body on which the force acts. If such a device could be used, perhaps intermittently, to propel an astronaut through space, its simplicity would be its greatest asset. But, even if this mode of propulsion is not used intentionally, it would be desirable to understand how limb motions interact with such a propulsion system, for it is possible that a malfunctioning of a more sophisticated astronaut maneuvering device leads to unintentional use of this simple technique, and limb motions might then provide the only means of achieving even partial control. We have, therefore, begun to study this problem analytically, and, while our investigation has not been completed, a few results can be reported.

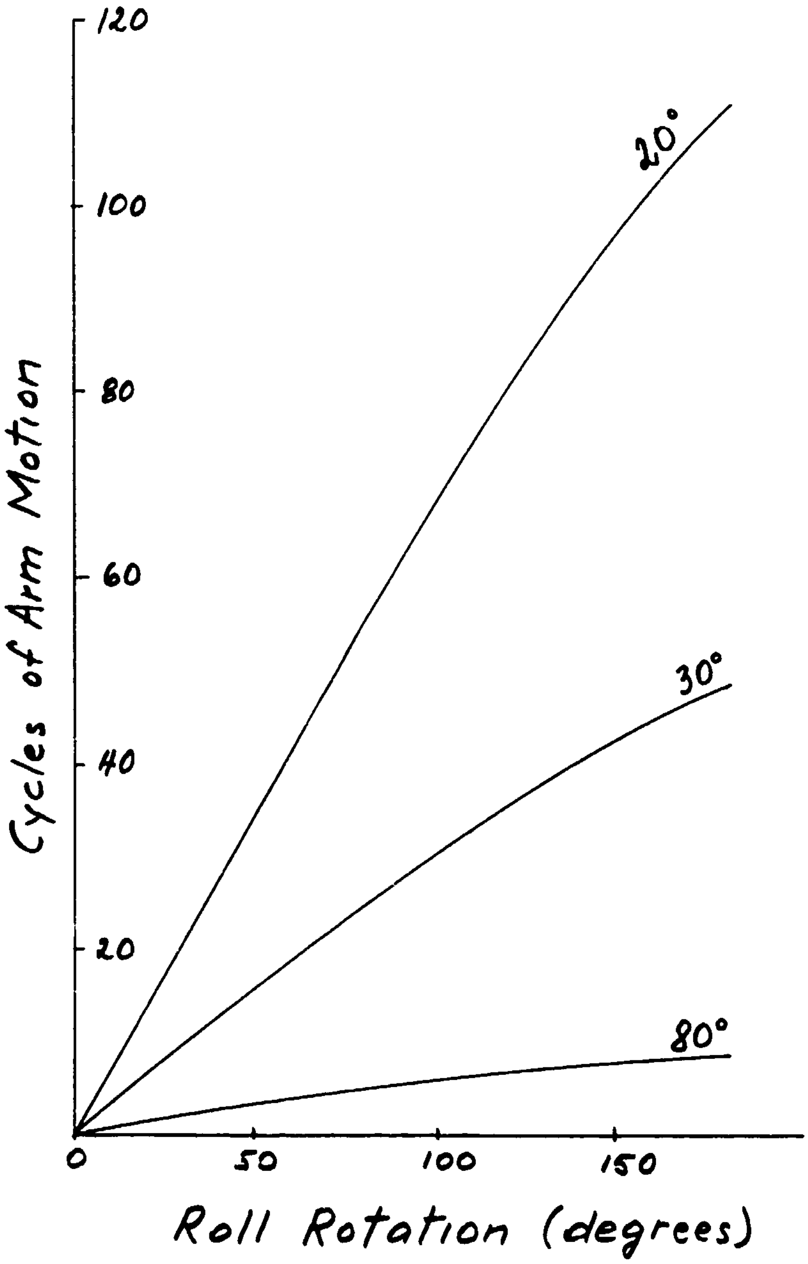
Considering a system of two rigid bodies which are connected at one point but are otherwise free to move relative to each other, one can readily see that rectilinear motion is possible only if the bodies are oriented relative to each other in such a way that the line of action of the thrust passes through the mass

PITCH MOTION

A pitch motion of the torso and legs (here regarded as a single, rigid body) can be produced by rotating the arms symmetrically, each arm moving essentially on the surface of a cone. (This maneuver was discussed both by Kulwicki[2] and by McCrank and Segar.[3]) To study such motions, the governing differential equation was written for a system comprised of three rigid bodies. The solution of this equation yields a relationship between the reorientation (in degrees) per cycle of arm motion, the geometric and inertia properties of the rigid bodies, and the three angles that characterize the arm motion, namely the semi-vertex angle of the conical surface on which each arm moves and two angles which locate the axis of one cone. (Symmetry considerations permit one to locate the axis of the second cone.) Details of this analysis will be published in a forthcoming Technical Report of the Department of Applied Mechanics, Stanford University. Fig. 1, which shows some of the results of the analysis, deals with the case in which the cone axis for each arm is parallel to the pitch axes. The three curves in the figure correspond to three sets of inertia properties, the first being applicable when the torso and legs occupy the same relative position as when a man is standing at attention, the second representing a "tucked" position, and the third accounting for changes in the inertia properties associated with holding a five pound weight in each hand. Not surprisingly, it turns out that both the use of hand-held weights and the tucked position enhance the effectiveness of the maneuver. Perhaps less obvious, but fortuitous, is the following fact which emerges from the analysis: The maneuver can be made more effective by choosing the axes of the conical surfaces on which the arms move in such a way that these axes are not parallel to the pitch axis. The optimum orientation of the cone axis is one in which this axis lies in the roll plane and makes an angle of about fifteen degrees with the pitch axis. This is a rather comfortable position, even in some EVA suits, and reorientations as large as twenty-five degrees (and twice that, if five-pound weights are attached at the wrists) can be obtained in a single cycle of arm motion.

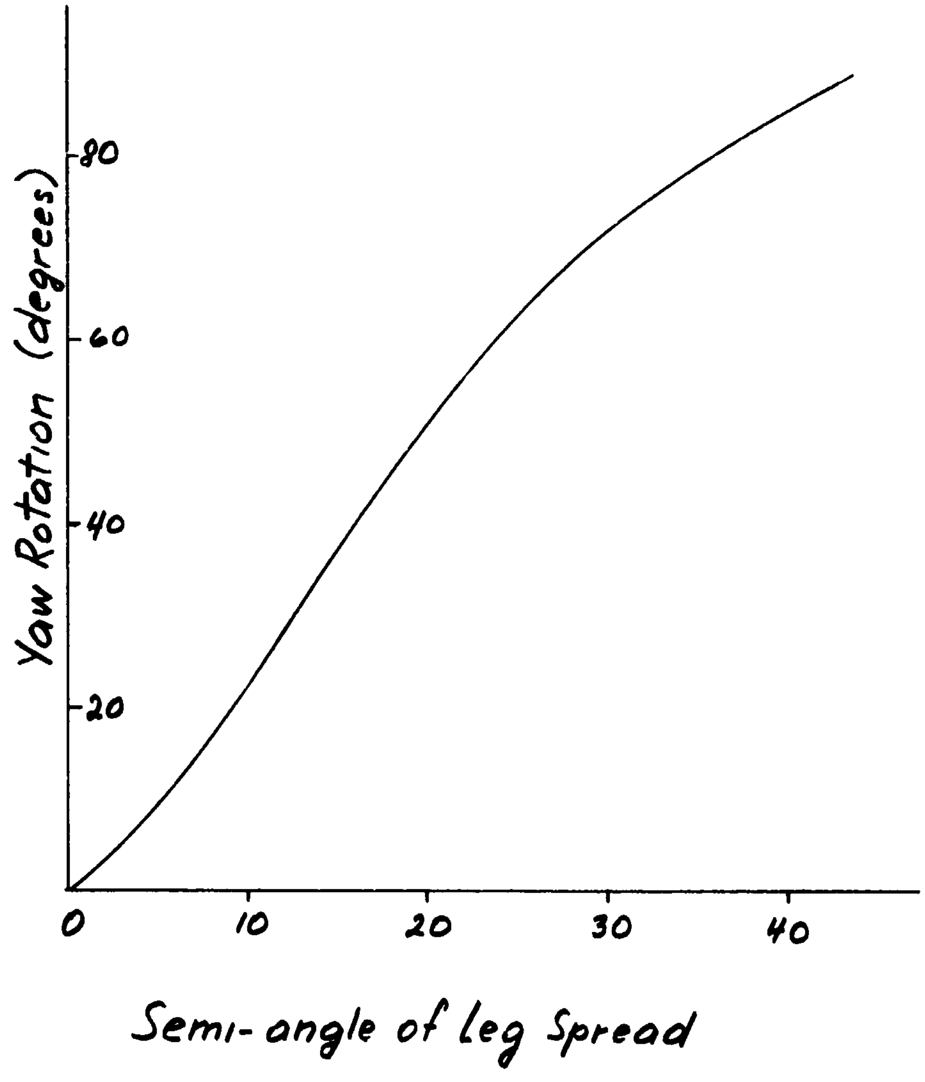
ROLL MOTION

The simplest way to obtain roll appears to be to move the arms on identical conical surfaces, but keeping their axes parallel to each other, so that the arm motions are now unsymmetrical relative to the torso. This lack of symmetry complicates the analysis considerably. In fact, it becomes necessary to solve a set of three simultaneous, nonlinear differential equations in order to determine the reorientation per cycle of arm motion. Such an analysis is described in detail in Technical Report No. 182 of the Department of Applied Mechanics of Stanford University. Fig. 2 shows the number of cycles of arm motion required to produce a given roll reorientation for various semi-vertex angles and with the axes of the cones parallel to the roll axis. Of course, the maneuver can once again be rendered more effective by adding weights to the arms; but, in doing so, one pays a penalty in the following way: The desired roll motion is inevitably accompanied by undesired pitch and yaw motions, and these tend to become more pronounced when the roll reorientation per cycle of arm motion is increased. Even without additional weights, these parasitic pitch and yaw rotations can have values of forty to



120
100
80
60
40
20
Cycles of Arm Motion
20°
30°
80°
0
50
100
150
Roll Rotation (degrees)

Fig. 2



Yaw Rotation (degrees)
80
60
40
20
0
10
20
30
40
Semi-angle of Leg Spread

Fig. 3

center of the system. For a given orientation of the force vector relative to one of the bodies, there may exist two, one, or no orientations of the second body such that this condition is fulfilled. Considering only planar motions, one finds that, when there exist two configurations which are compatible with rectilinear motion, then the motion associated with one of these is stable whereas that associated with the other is unstable; and when there exists but one such configuration, the associated rectilinear motion is always unstable. Finally, still for planar motions, it is found that the body to which the force is applied can rotate at a constant rate in inertial space when the second body performs an oscillatory motion relative to the first body in the neighborhood of the position associated with rectilinear motion.

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SESSION IV

SPACE MAINTENANCE TECHNOLOGY

Session Chairman: Colonel F. X. Kane
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SPACE TOOL POWER SOURCE SAFETY INVESTIGATION

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SUMMARY: A study was accomplished to define the hazards associated with poentially useful power sources for spacetool applications. It was found that all power sources offer some specific hazards. It has been recommended that the brushless D.C. electric power drive be developed to further decrease its potential as a hazardous device for use in spaceflight.

INTRODUCTION

A previous study compared various space tool power sources with respect to power/mass requirements.* This report presents the results of a continuing study directed primarily at the safety and reliability of the possible power sources to be used as a multi-purpose driving unit for hand-operated tools.

The potential power sources are divided into two basic safety groups, those involving fuels and/or power cycles which are normally considered to be inherently extra-hazardous and those involving fuels and/or power cycles which are normally considered to be of low inherent hazard. Other areas of discussion included in this report are:

1. Safety considerations for gas, electric, and hydraulic power.
2. Fire and blast hazards.
3. Toxicological hazard.
4. Special tool hazards.

Conclusions with recommendations are drawn as to the safest and most reliable power source for space use.

INITIAL SELECTION

All the potential power sources can easily be placed initially into two basic safety groupings.

*Space Tool Power Source Investigation. Internal Note R-ME-67-4.

1. Those power sources involving fuels and/or power cycles which are normally considered to be inherently extra-hazardous.

2. Those power sources involving fuels and/or power cycles which are normally considered to be of low inherent hazard.

The inherently extra-hazardous class includes all the solid and liquid rocket propellant-oxidizer combinations and the monopropellants such as hydrazine and hydrogen peroxide.

These high energy materials are inherently hazardous since they are highly reactive, frequently unstable, and relatively toxic with very low threshold limit values (TLV). Low allowable concentration percentages apply to these materials as separately stored fuel or oxidizer components and as reaction products. Their maximum allowable concentrations (MAC) from a flammability-limit standpoint are also low. As highly reactive agents they may be ignited by low-energy ignition sources; some are hypergolic. They usually combine or disassociate at high temperatures; therefore, the reaction chambers, and frequently the exhaust products, must be carefully and completely isolated from astronaut proximity. These extra-hazardous power sources have been developed for specialized space use and previously have been used on manned space flights. However, the location requirements for thrust engines, spacecraft attitude controllers, extravehicular activity (EVA) manuevering units, and other special uses are much less severe than the location and use requirements placed on the space tool power source.

It is possible that this category of power source could qualify for specialized EVA such as a high power/mass emergency tool.[1] However, as long as the power source may be used within the cabin, these extra-hazardous systems should be eliminated from the candidate list. This conclusion and the considerations in arriving at it are in agreement with that proposed by Roth.[2]

The initial selection leaves as potentially useable power sources only the normally low-hazard power systems. This category includes gas (air) powered, electric powered, and hydraulic power systems.[3] All three of these systems are used on Earth as tool power sources.

Several previous space tool power sources have used dc electric motors.[4,5] The characteristics of this type electric motor are very favorable for use as a tool power source. The dc electric motor's high efficiency, high power-to-weight ratio, and torque/load characteristics make it one of the most suitable electric motors for hand tool use. In addition, spacecraft electric power systems usually include several dc prime power sources. This is true because of the basic dc nature of these devices, i.e., solar cells, thermoelectric, batteries, and/or fuel cells. The dc motor is a natural choice since it can use power directly from these prime spacecraft power sources.

Previous considerations have eliminated all the exotic high-energy gas cycles. This leaves the low-energy, pressure-work type gas cycle as used in all terrestrial gas (air) powered hand tools. These motors use air at line pressure developing power across a work-surface area; little or no work is obtained by gas expansion and the gas reaches the exhaust port at practically line pressure.

True turbine motors do not operate well on this power source.[6] They have even lower starting torques than turbine motors operating on more effective turbine power cycles. Except for very specialized light loads, high-speed gas-powered turbines are not considered to be valuable as a space power system. Only two types, the piston and the rotating vane, remain as of major importance. The piston motor is generally heavier and bulkier, and is usually not offered in tools under 373 W to 559 W (0.5 to 0.75 hp). The vane-type gas pressure motor will be the major gas motor considered in this report.

In the early stages of investigating the safety aspects of pneumatic power sources it was felt that a closed fluid working cycle might offer several safety advantages over the pneumatic. Therefore the safety aspects of hydraulic tool power will also be considered.

Hydraulic systems are mainly power transmission devices and are not prime power sources. The hydraulic system will have to be driven by a gas or electric prime power source and it will have some of the advantages and disadvantages associated with the chosen prime power source.

SAFETY CONSIDERATIONS FOR GAS, ELECTRIC, AND HYDRAULIC POWER

Even though the three remaining candidate power systems are considered safe in normal terrestrial use, their use in space flight poses several unusual problems of some potential danger. These specialized problems can be reduced to three categories:

1. Fire and blast.

2. Toxicological contamination.

3. Safety hazards peculiar to the type of tool.

The three types of power sources will now be considered within these three potential hazard areas.

Fire and Blast

Two very different use environments apply to the proposed tool power source, EVA or intravehicular activity (IVA). In EVA use, the accidental fire or blast hazard is much reduced from that found for similar tools in terrestrial use.[2,7] This is primarily a result of the unavailability of a natural supply of oxygen in the space vacuum environment. Most liquid or gaseous fuels or oxidizers will be vented naturally and dispersed by this very low pressure. Also, a decrease in pressure usually narrows the flammability range and increases the auto-ignition temperature.[8]

Another factor which will reduce the chance of orbital EVA fire is the unusual "zero-gravity" gravitation field. Zero gravity, in the absence of induced relative velocities, will reduce the mixing process to either a random low-velocity mixing or diffusion-controlled mixing. Either process is less effective than mixing driven by displacement convection between "heavier and lighter" liquids and gasses.

The normal terrestrial convection-burning process can also be affected by zero gravity. Spalding lists the equation for the burning of fuel vapor droplets in air.[9]

$$\frac{Mdc}{k} = (45B)\ 0.75 \left[\frac{(gd^3)^{0.25}}{a}\right]$$

where M = Vaporization rate per unit face area.

d = Droplet sphere diameter

c = Specific heat

k = Thermal conductivity

B = Transfer number, a fuel function

g = Acceleration due to gravity

a = Thermal diffusivity

Even though the equation indicates so, the conclusion obviously cannot be made that the burning rate is zero when g is zero. Roth suggests an equation of the form[2]

$$M_g = M_o\ [1 + f\ (g)]$$

The function f (g) would be relatively small compared with unity, and the subscripts g and o refer to gravity and zero-gravity cases. In any case the lack of normal convection current per se will reduce the major natural oxygen replenishment mechanism found operating in an Earth atmosphere (convective) fire.

The extravehicular environment is considered by most authors as a natural fire extinguisher for fires that may break out even inside the cabin. The procedure suggested is to depressurize the cabin to the exterior vacuum in case of internal cabin fire.

It has been shown that fire hazards increase in space flight; therefore, the major increased fire hazard in space flight must arise from the special conditions of IVA and not EVA.

Several extensive studies have been accomplished aimed at defining the extra hazards introduced within the space cabin in space flight. Final and exact answers to these questions must await combustion experiments scheduled for future flights. It is known that past flights with 100 percent oxygen cabin atmospheres were relatively more hazardous than an Earth environment. Not only may more oxygen be available but the lack of a diluent atmospheric gas also contributes to the shortened time scale and higher temperature of combustion found in cabin fires under 100 percent oxygen atmospheres.

No attempt will be made here to reassess or restate these extensive studies except to accept the environment as a potentially more positive fire risk and to reference prior test data as it specifically applies to operation of or to materials of construction of the space tool power source.

The Power Tool as an Ignition Source

All power tools have certain common potentials as ignition sources. In this function the tool would serve to ignite another fuel or combustible. This is the major terrestrial danger considered for constructing and using nonsparking tools in highly combustible or explosive atmospheres. The problem may be more severe in space cabin atmospheres. Minimum ignition energies are lowered with increased oxygen partial pressure.[10] Huggett et al. showed slightly lower ignition energies for some common materials required in space cabin atmospheres as compared to ignition in air.[11]

According to Voigtsberger[12] and Roth[2] the spark energies required to ignite common clothing materials may be decreased more than one thousand fold in pure oxygen to levels similar to those of electrostatic sparks from the human frame. The rate of burning after ignition of some common spa... cabin materials may increase b, ive fold on replacing air with oxygen. Propagation of flame may be much faster in the gas phase.[13] There are certain ignition source modes which are common to every tool regardless of the type of power source. These are discussed in the paragraphs below.

Electrostatic Sparks. In general, the grounded all-metal tool housing would not be expected to build up or hold an electrostatic charge in normal use. Electrostatic spark potentials can be accumulated only on parts of a device which remain electrically isolated for a sufficient period. Under certain conditions aluminum alloy housings, as might be used in all types of tool housing could become electrostatically dangerous.

Aluminum is a chemically reactive material, especially with respect to its combination with oxygen. The aluminum oxide product is highly adherent to the base aluminum, is chemically inert, hard, dense, mechanically strong, and serves to protect the base aluminum from further oxidation and other chemical attack. Aluminum oxide is also a dielectric material. Its formation on the aluminum surface, especially under dry atmosphere conditions, can create an electrically isolated surface which could store an electrostatic charge.

Frequently aluminum products are processed through one of the anodizing processes. Anodizing processes control the formation of aluminum oxide on the aluminum surface; usually it is done to give the surface a harder, more wear resistant finish of from 25 to 26 μm (1 to 3 mils) thickness. The "hard coat" process is a special anodizing process which gives a surface from 177.8 to 304.8 μm (7 to 12 mils) or more and is especially long-wearing and abrasion-resistant.

Such coatings pose a hazard and should be avoided to reduce the electrostatic spark hazard from aluminum surfaces. Further, should aluminum be perferred because of its other properties to another nonsparking alloy, such as beryllium-copper, which does not exhibit the tendency to auto-oxidize to a dielectric surface, then special surface finishes should be developed for the aluminum housing. Such surface finishes would coat the aluminum with a thin, conductive, nonsparking, non-auto-oxidizing material. Care must be taken in such a coating process to insure that the coating is applied directly on the aluminum base metal. An equally dangerous capacitive spark source can be created should a conductive coating be placed over the aluminum oxide coating.

The vane air tool has another type potential electrostatic hazard. The rotating vanes themselves are usually manufactured to phenolic-impregnated linen-fiber material.

This material can generate electrostatic sparks which would discharge vane-to-housing. The high moisture conditions of most delivery air in terrestrial maintenance shops would serve to reduce or eliminate such a conditions here on Earth. In a dry atmosphere the high rotational speed might help compensate for the low insulated surface (vane) area and an electrostatic spark hazard would exist within the air tool. Development of conductive vane materials or conductive surface coatings would eliminate this hazard in the air tool.

Switch Sparks. The electric motor as a primary space tool power source, or as the power drive in a hydraulic system, would have whatever ignition hazard is offered by sparking at an on-off motor switch. The gas power source does not offer this particular hazard. By definition such sparking can occur only between the contact points in the switch. The problem of the ejecta-spark, or ejected hot particle, as an ignition source is treated separately in this report. The physical arrangement of such electrodes is such that the practical danger of a fire being initiated by this spark is offered only to ignite that fuel supply which can be brought to the spark, i.e., passed between the electrodes. Gaseous fuel-oxidizer mixes meet this requirement and will be treated as the only probable fire threat offered by the switch spark. The major space flight atmospheric variables of the type of gaseous composition, the percentage of oxygen, and the total pressure affect both the minimum electric gap length required to ignite a given gaseous fuel-oxygen mix and also the minimum voltage for the production of the spark.[11, 14, 15]

Short-time sparks supply the energy necessary for ignition in a few microseconds. This energy triggers the chemical reaction (flame) in a very small sphere of the combustible mixture. For some time it has been known that continuation of the flame front and development of a general fire will depend on whether the small initial sphere can propagate without being extinguished.[2] The electrode gap may act as a quenching agent on this small flame. Figure 1 shows the dependence of the critical (minimum) energy for ignition on the gap length.

From this graph it can be seen that gap lengths shorter than the minimum will require greater spark energies to propagate combustion. A spark gap will begin to dissipate its energy almost immediately in the case of "break" sparks, the left hand branch of the curve. In this case the close spacing of the operating contacts may serve to quench the process. This indicates that "make" sparks will be more dangerous than "break" sparks. There are other factors operating in break sparks in motor circuits, however, which makes the energy available in motor circuit "break" sparks more than that energy available to the switch in "make" sparks. These effects will be considered later.

The minimum ignition energy also depends on the fuel-air ratio for any given combustible mixture (Figure 2).

The relative diffusivity of the fuel is also a control on the minimum ignition energy. For a homologous series of hydrocarbon fuels the minimum ignition energy shifts toward higher stoichiometric fuel-air ratios.

A practical demonstration of the wide variation of minimum ignition-gap length on oxygen

FIGURE 1

FIGURE MINIMUM IGNITION ENERGY VERSUS FUEL AIR RATIO

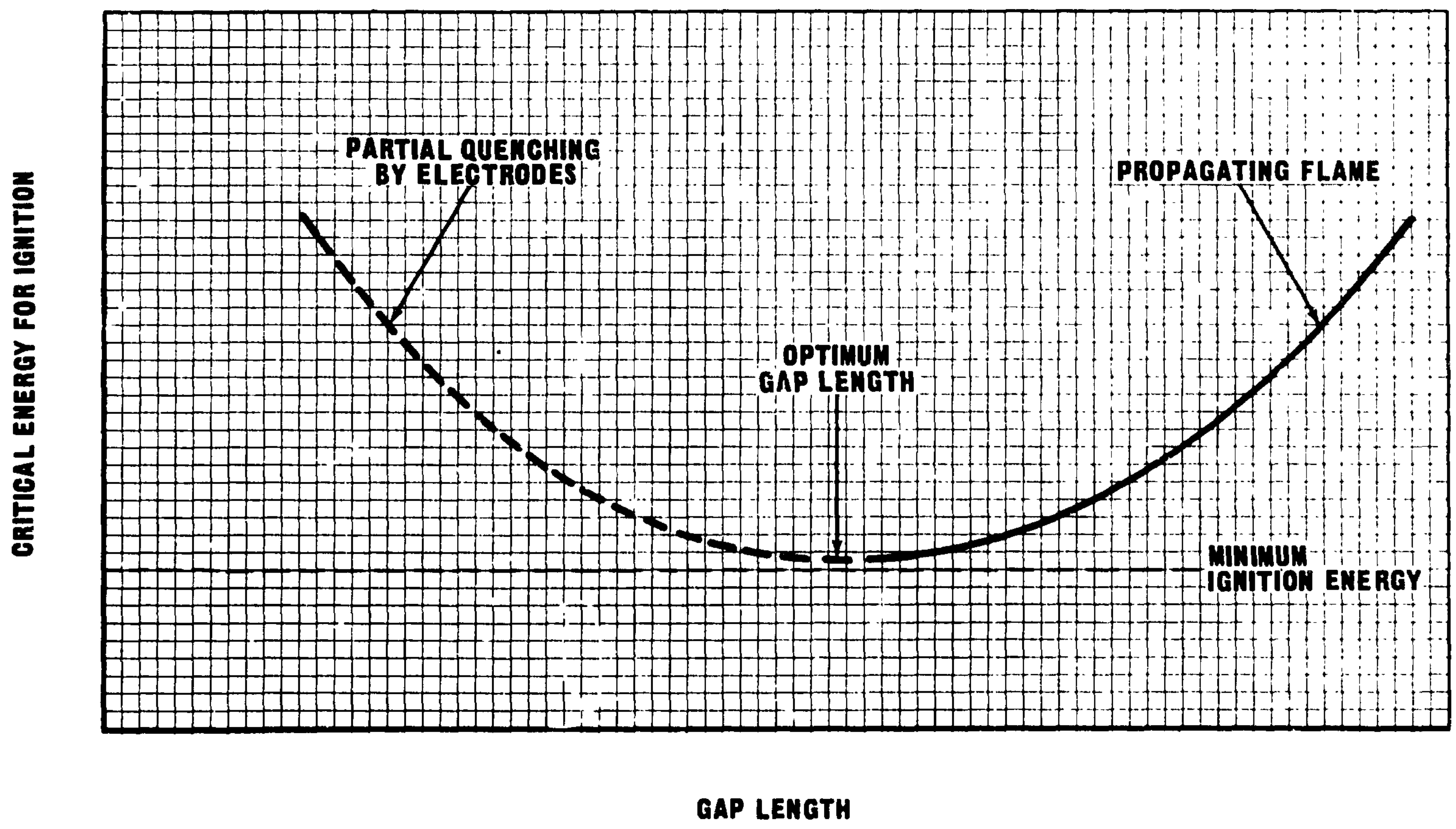
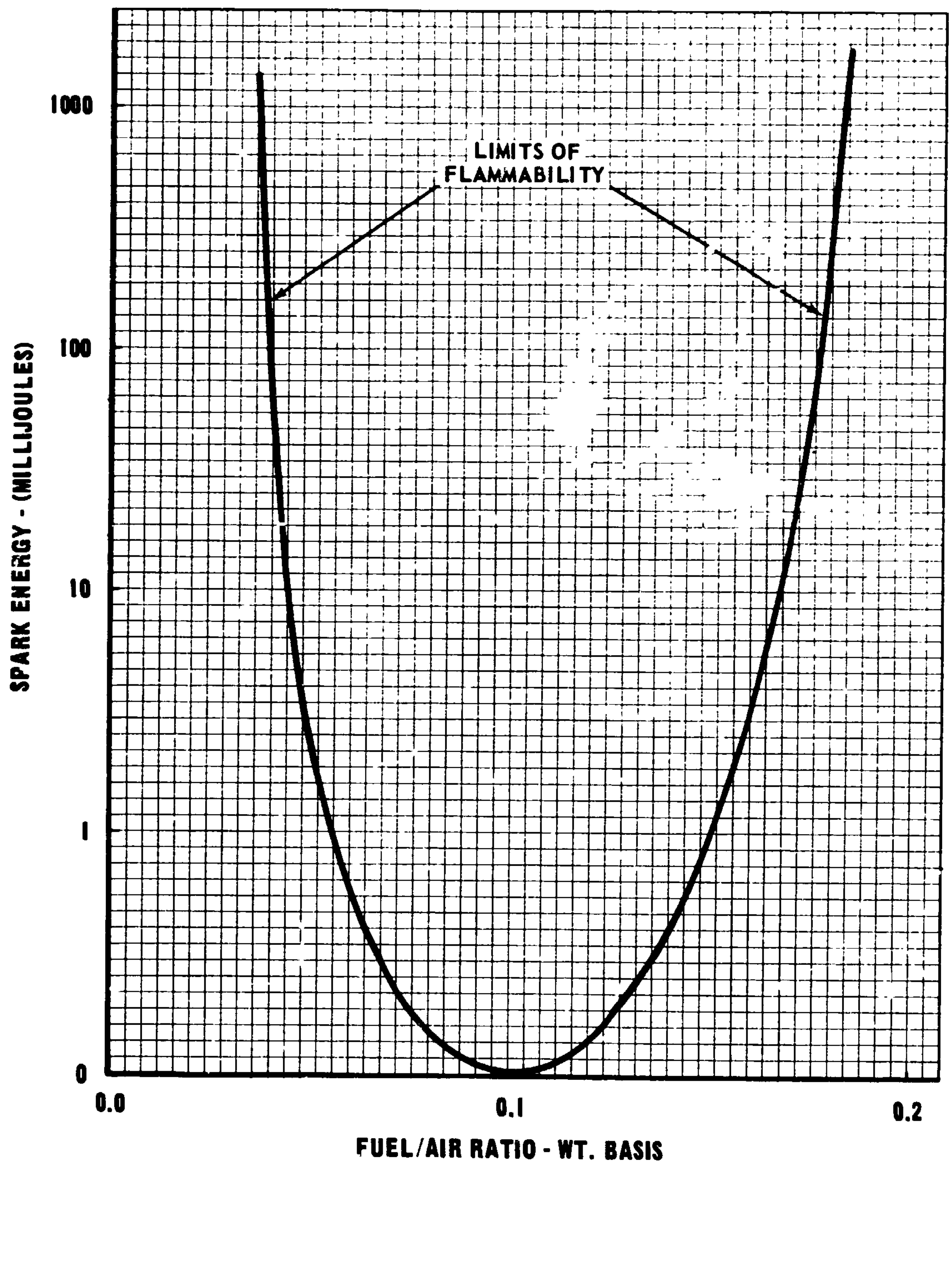


FIGURE 2

MINIMUM IGNITION ENERGY VERSUS GAP LENGTH



concentration for two common gaseous hydrocarbon fuels (gasoline and ethyl ether) is shown in Figure 3. As the percentage of oxygen in the air is varied from 20 to 75 percent, the minimum spark gap for ignition of gasoline decreases from 0.089 mm down to 0.001 mm. These test results show the minimum spark gap required to ignite a gaseous fuel at atmospheric pressure decreases by a factor of more than 10 when the oxygen concentration is increased from 20 percent to above 75 percent.

Figure 4 shows the effective increase of the fire-ignition hazard with increasing oxygen concentration. It decreases with decreasing total pressure. From atmospheric pressure $101.35 \times 10^3 N/m^2$ absolute (14.7 psia) down to $19.99 \times 10^3 N/m^2$ absolute (2.9 psia) the minimum ignition energy decreases by a factor of 10.

The data of Figures 3, 5, and 6 taken together indicate that in the current space cabin atmosphere and those found on projects Mercury and Gemini, the overall result of increasing the oxygen concentration to 100 percent and decreasing the total pressure to 24.13×10^3 to $37.92 \times 10^3 N/m^2$ absolute (2.5 to 5.5 psia), decreases the minimum spark ignition energy by an overall factor of 12 to 70.

In considering the development of sparks at switch contacts there are two separate and different conditions: The "make" spark and the "break" spark. The "make" spark, or breakdown potential, V_b, is a function of the type gas and the product, δS, where δ is the density of the gas and S the gap width. Considering the temperature to be constant, we may replace this produce with pS, where p is atmospheric (or space cabin) pressure and S is electrode gap width. This similitude law is known as Paschen's law. Figure 6 shows a plot of V_b versus the produce pS.

The minimum breakdown voltage of 330 volts, occurs at pS = 9.117 N/m^2-cm(2×10^{-3} atm-cm). This general curve of Paschen has been reproduced by the data of more recent investigators.[16] Recently Germer [17-19] investigated very closely spaced electrodes down to a spacing of 1×10^{-5}cm (1000Å). He found arc ignition below Paschen's minimum of 330 volts, even down as low as 50 volts. Arc ignition at these very close spacings is explained as resulting from very high field concentrations occurring at surface asperity peaks.

Most electric tools in the range of consideration of this report will use power circuit voltages much lower than Paschen's minimum ignition voltage and even considerably lower than the 50 volts found at extremely close spacings by Germer. The conclusion is that "make" sparks in tools using 30 volts or less do not constitute an ignition hazard from the "make" spark switch source.

An interesting aspect of combining the information from the data obtained from Figures 3,4, and 6 is presented in Figure 5. Considering a cabin atmosphere of more than 80 percent oxygen content and a minimum gap length in this atmosphere which will ignite most low molecular weight hydrocarbons 0.0025 cm (0.001 in.) we calculate and plot the curve of Paschen's breakdown potential versus gap-pressure. This composite curve shows several important facts relevant to the operation of electrical switches under space cabin and space flight conditions:

1. The electrical breakdown

FIGURE 3

MINIMUM IGNITION GAP LENGTH VERSUS
OXYGEN CONCENTRATION

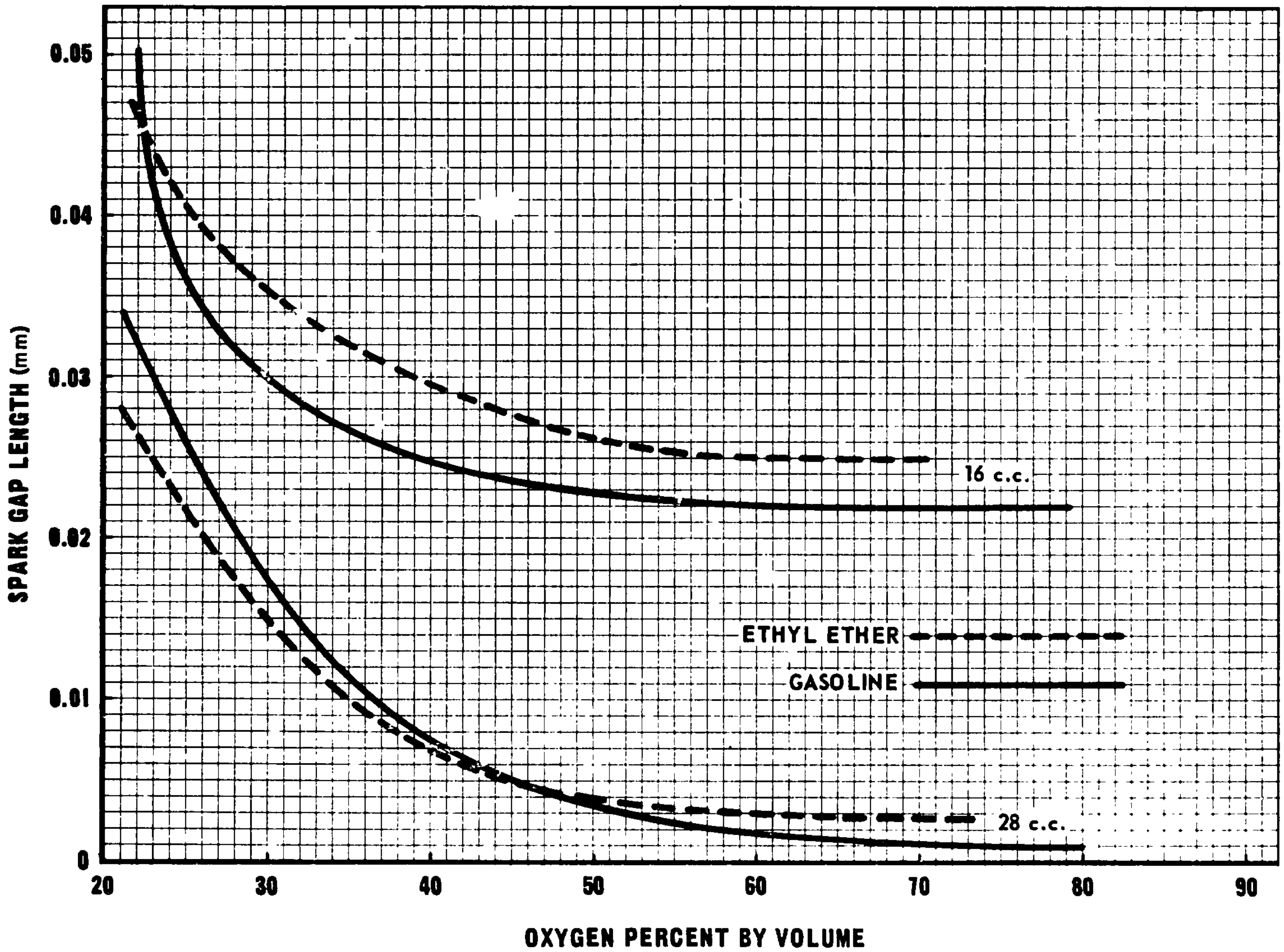


FIGURE 4

MINIMUM SPARK ENERGY AS A FUNCTION OF
ATMOSPHERE COMPOSITION

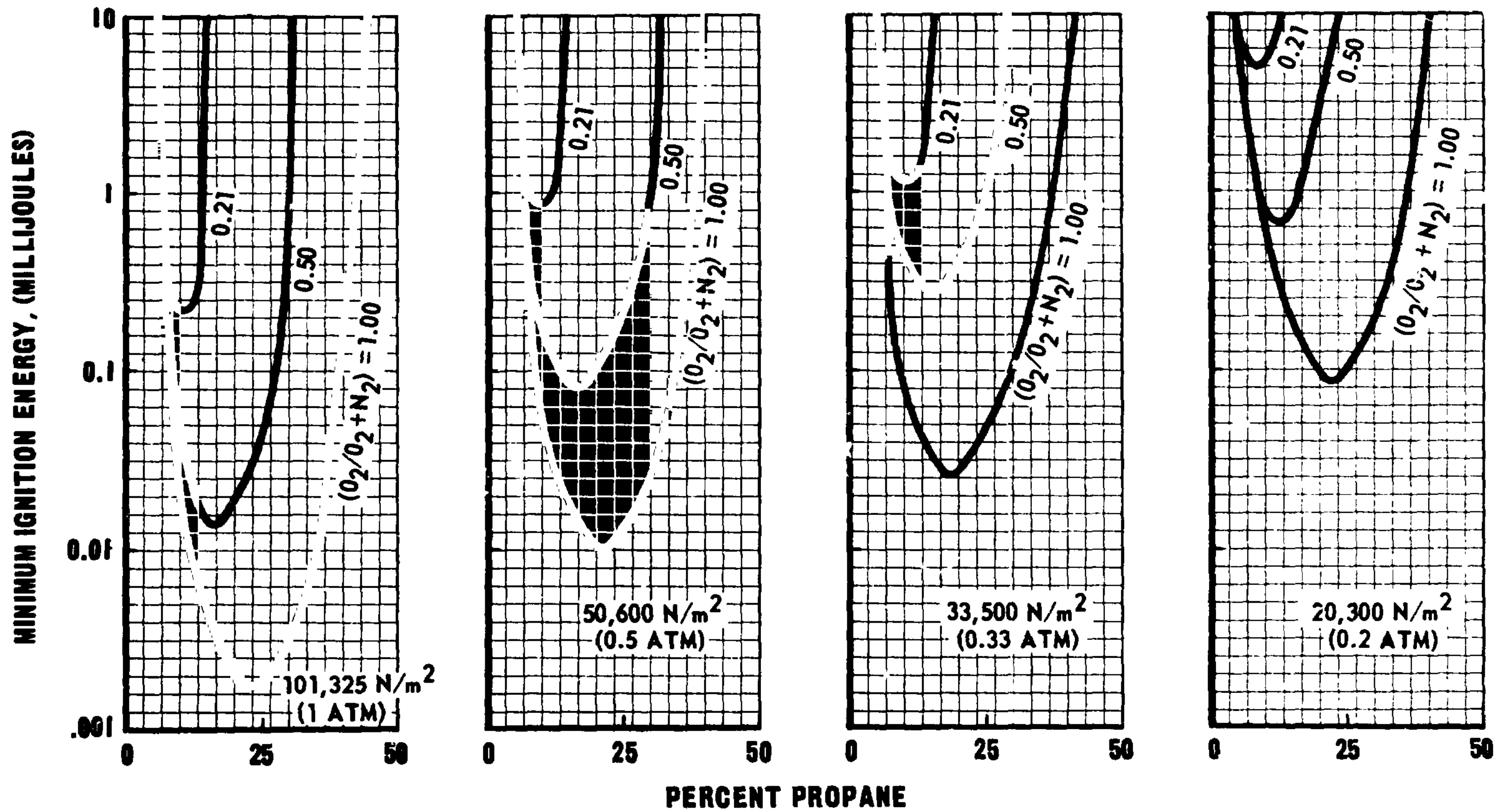


FIGURE 5

ARC BREAKDOWN POTENTIAL AS A
FUNCTION OF GAP PRESSURE

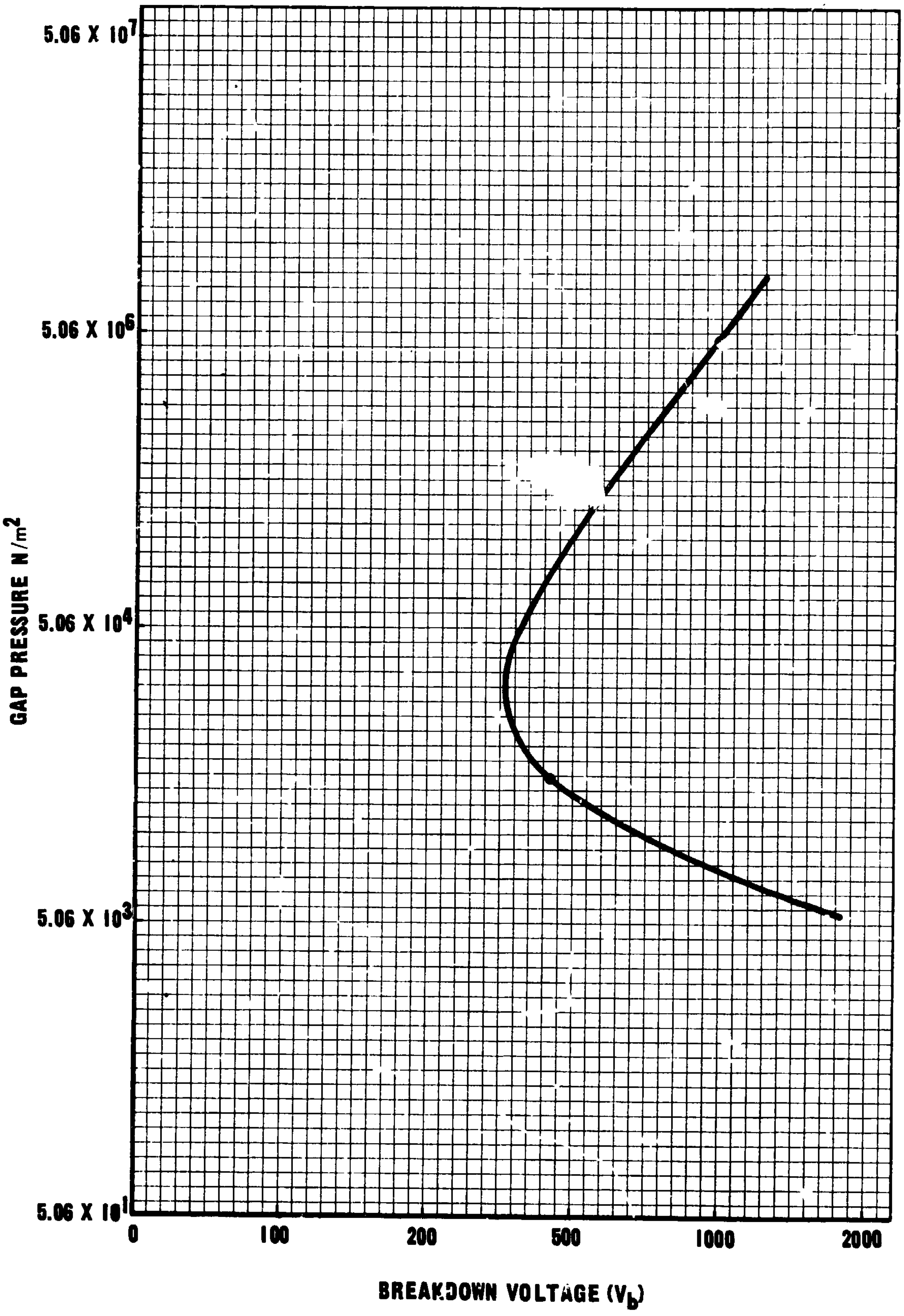
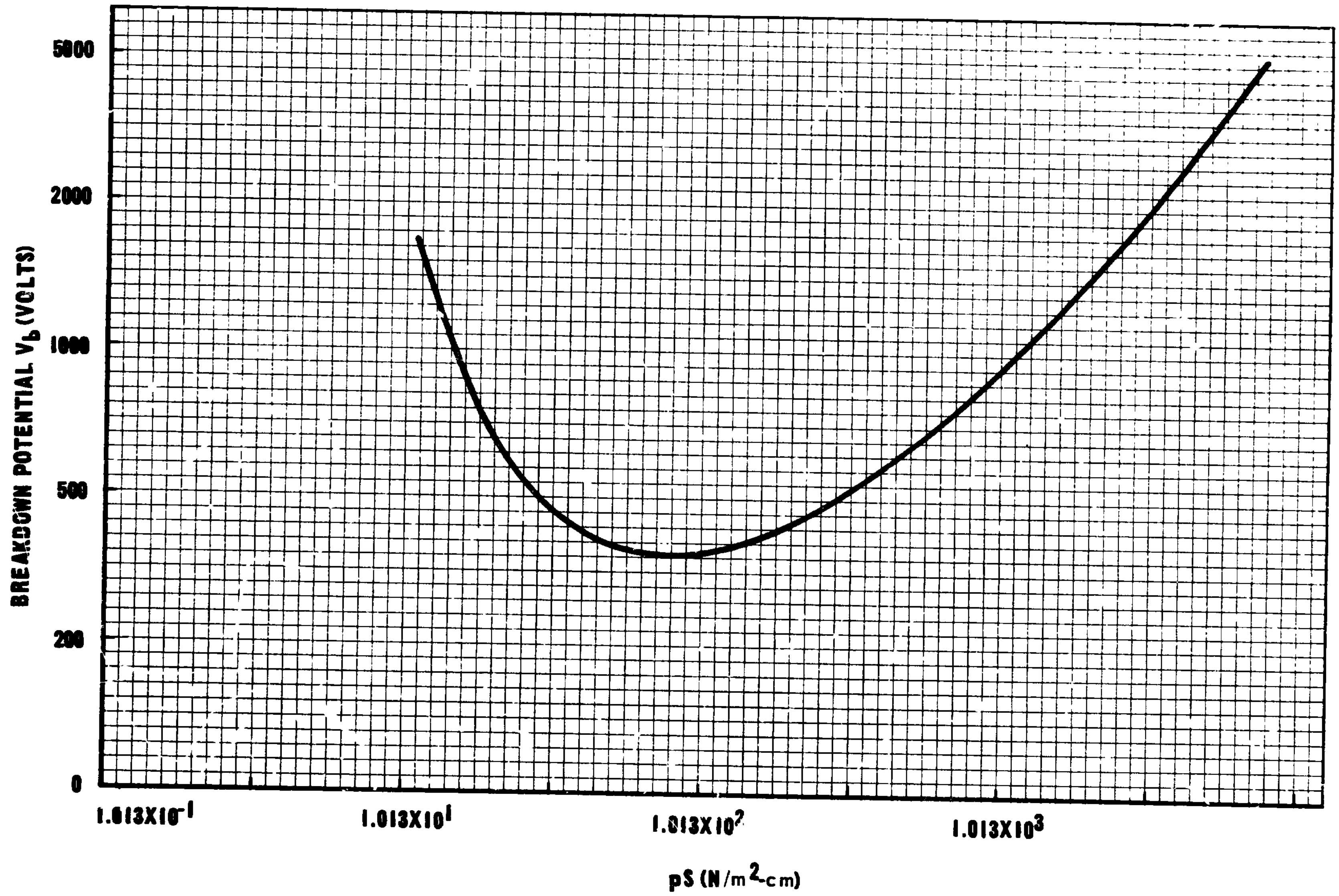


FIGURE 6

BREAKDOWN POTENTIAL AS A FUNCTION
OF GAP WIDTH AND PRESSURE



gap length is a minimum in the 100 percent oxygen atmosphere at between 39.99 x $10^3 N/m^2$ absolute (5.8 psia) and 20.68 x $10^3 N/m^2$ absolute (3.0 psia). This pressure range is precisely that range in which past and present spacecraft atmospheres have been operated.

2. Pressures lower or higher than the range 20.68 x 10^3 to 39.99 x $10^3 N/m^2$ absolute decrease the danger of development and ignition of "make" sparks. The lowest pressure (EVA) is considered a much lower risk. The reduction of "make spark" risk at high pressures suggests that the solution should be a hermetically-sealed pressurized switch.

3. Reducing the electrode voltages below 330 volts, especially below 50 volts, practically eliminates "make" spark risk.

The "break" spark is fundamentally different from the "make" spark. There are two major differences which apply to the conditions just preceeding the creation of the "break" phenomenon as compared to the " make" condition:

1. The surfaces are in mechanical contact.

2. Some parts of the surfaces are carrying the circuit current (I_c).

Considerable work has been done on the nature of contact of these surfaces.[15,16,20] The actual area of mechanical contact, even for the smoothest surfaces, is at most about 0.1 percent of the mechanical contact surface. The conducting spots or constriction areas are thin highly-conductive bridges. The bridges are formed by a process called fritting. "A" fritting generally means the breakdown of insulating films originally found covering the contact surfaces. This breakdown is accompanied by local conduction, intense localized heating of these conducting points, and melting of the fritting spot with subsequent formation of the molten conducting bridges.[15] The actual conducting area is small and the current density in the areas is very high. Contacts carrying only a few amperes, such as a switch cutting the power to an electrically powered hand tool, may have current densities of the order of 10^8 amps/cm^2. The importance of this high current density passing through the resistance is not found in its electrical resistance, which is usually only milliohms, but in its thermal capacity. Most of the energy lost in passing through the contact surfaces is lost as heat energy ($I_c^2 R_c$) generated in the immediate vicinity of the conducting aspherities. Because of the small thermal capacity the spots exist at higher temperatures. The second stage of the process of fritting, called "B" fritting, operates in such a way as to keep the bridges molted. If the current increases, "B" fritting generally results in the molten spots increasing in size to maintain nearly the same value of constriction resistance and to carry the extra current.[15]

Under breaking the contact we start with the molten conducting bridges and begin to reduce the normal load. The area of contact decreases and consequently the constriction resistance, current density, and temperature increases, and the material in the bridge vaporizes as the switch opens. The opening switch thus creates its own high-temperature, highly-conductive, vaporized metal, or plasma path which constitutes the beginning points for the drawn spark or arc. Once such a high-temperature vapor path is formed it will draw out a spark or arc until either the current or voltage goes below a value necessary to maintain the conducting path. These shortest-arc minimum values are known as minimum arc current (I_m) and minimum arc voltage (V_m). Both of these important limits depend on the cathode materials. Values of I_m and V_m for typical switch contact materials are:

	I_m Amps	V_m Amps
Carbon	0.01	15-20
Silver	0.45	8-12
Copper	0.43	12
Tungsten	1.0	10-15

These minimum values are within a range which can be developed in a dc space tool power circuit. The conclusion is that "break" sparks are possible in the low voltage space tool power circuits and constitute a potential ignition hazard.

The extinguishment of the break-spark hazard is possible if we could reduce either the current or voltage before contact separation below the minimums. It should be recognized that in power tool systems a further consideration must be made. These circuits contain inductance and capacitance, and in dc systems undergoing a transient (switching) condition the inductance serves to supply higher transient circuit voltage values than occur during steady electrical conditions. Even though the power circuit could be designed and operated below the minimum arc current (I_m), on switch opening the inductance would operate to prevent any circuit changes, and arc would ignite at: $I_o = \frac{E-E_c}{R}$

where E_c is the last voltage across the switch contacts. After ignition the circuit varies as:

$$E = IR + L\frac{dI}{dt} + V\ (I,S)$$

where V(I,S), the spark voltage, is a function of I and arc length S.

The presence of capacitance and inductance in most power circuits may lead to arcing and sparking in circuits otherwise operating below I_m and V_m. The inductance may provide voltages higher than the prime supply voltage and pulse currents higher than I_m may be drawn from the capacitance.

Ejecta Particle Sparks. Several types of hot incandescent particles may be created and ejected by tools:

Metal strike sparks. "Sparking" and "non-sparking" metal tools have long been considered in explosive or potentially dangerous atmospheres on Earth. There are two major chances for developing metal strike sparks in space, striking the tool housing and metal sparks generated in the impactor mechanism of some tools.

The U. S. Department of Commerce has done research on the sparking of metals in an atmosphere which has some relevance to space cabin atmospheres. In this work the sparking characteristics and the ignitability of flammable mixtures were tested under increasing concentrations of oxygen. The results of these tests are summarized in Tables I and II. These tests showed that metals safe from strike-sparking, flammable, and explosive high-oxygen atmospheres include maganese bronze, phosphorus bronze, aluminum

TABLE I. SPARKING CHARACTERISTICS OF VARIOUS METALS IN A GASOLINE AND OXYGEN ENRICHED AIR MIXTURE ($0.13\ M^3$ [$4.5\ ft^3$] OF AIR WITH 50 PERCENT OXYGEN)

Rod Specimen		Metal Wheels					
Material	Rockwell Hardness	Carbon Tool Steel Rc 63	H.S. Tool Steel Rc 66	Alloy Steel (Bolt Rock) Rc 56	Stainless Steel Rc 37	Carbon Steel Rc 62	Abrasive Wheel
16 cc Gasoline							
Carbon Tool Steel	B72	X	X	X	X	X	X
H.S. Tool Steel	B92	X	X	X	X	X	X
Stainless Steel	B93	X	X	X	X	X	X
Carbon Steel	B82	X	X	X	X	X	X
Monel (Nickel Copper)	B90	X	X	X	X	X	S(1)
28 cc Gasoline							
Manganese Bronze	B89	N	N	N	N	N	N
Phosphorus Bronze	B90	N	N	N	N	N	N
Aluminum Bronze	B93	N(2)	N	N	N	N(2)	N
Commercial Brass	B72	N	N	N	N	N	N
Aluminum	B56	N	N	N	N	N	N
Beryllium Copper	C22	N(2)	N	N	N	N(2)	N

Keys: X - Visible sparks and explosion.
S - Visible sparks, no explosion.
N - No visible sparks, no explosion.

Notes: (1) This result is for 28 cc of gasoline.
(2) Second test conducted on a rusted wheel; no visible sparks, no explosion.

TABLE II. SPARKING CHARACTERISTICS OF VARIOUS METALS IN A GASOLINE AND AIR MIXTURE (20 cc Gasoline 0.13 M^3 [4.5 ft^3] OF AIR)

Rod Specimen		Metal Wheels						
Material	Rockwell Hardness	Carbon Tool Steel Rc 63	Carbon Tool Steel Rb 72	H.S. Tool Steel Rc 66	Alloy Steel (Bolt Stock) Rc 56	Stainless Steel Rc 37	Carbon Steel Rc 62	Abrasive Wheel
Carbon Tool Steel	B72	X	S	S	S	S	S	X
H.S. Tool Steel	B92	S	S	S	S	S	S	S
Stainless Steel	B93	X	[illegible]	S	X	S	S	X
Carbon Steel	B82	S	S	S	S	S	S	X
Monel (Nickel Copper)	B90	S	S	S	S	S	S	S
Manganese Bronze	B89	N	N	N	N	N	N	N
Phosphorus Bronze	B90	N	N	N	N	N	N	N
Aluminum Bronze	B93	N	N	N	N	N	N	N
Commercial Brass	B72	N	N	N	N	N	N	N
Aluminum	B56	N	N	N	N	N	N	N
Beryllium Copper	C22	N	N	N	N	N	N	N

Symbols: X - Visible sparks and violent explosion.
S - Visible sparks, no explosion.
N - No visible sparks, no explosion.

bronze, commercial brass, aluminum, and beryllium copper. Unsafe metals include carbon steel, carbon tool steel, stainless steel, and monel (nickel copper).

A fortunate result from these tests is that aluminum and high-aluminum alloys frequently used in the tool housing are non-sparking with respect to strike- or abrasion-generated metal sparks. Since regular prediction of metal strike-sparks from a tool housing under use by a man is impossible, the only approach that can be accepted is to manufacture the housing of such non-sparking materials.

The condition in the impactor mechanism is more predictable than in the tool housing, is more controllable in design, and can be tested for possible sparking after construction. Most impactor mechanisms have relatively flat anvil/hammer surfaces, with little abrasion and surface shear occurring in use. Simple substitution of non-sparking alloys may be possible since the hardness of non-sparking aluminum-bronze alloys (Rockwell B93) compares favorably with the steel materials (B90 to B92) sometimes used. However, only the tool designer in the original design process can adequately evaluate the substitution of these specified non-sparking alloys in the impactor section. Other factors under the designer's control include maximum impact pressure (the impactor surface area), the shape of the impactor surfaces, and the geometric impacting conditions. Using all the available design conditions the impact mechanism can be made safe from metal strike sparking. Test procedures simulating IVA and EVA use under long term normal and possible failure mode should be used to verify the adequacy of any materials/design compromise, should this become necessary.

Incandescent carbon wear particles. In some space power tool machinery we have situations where carbon elements are in sliding contact with metal surfaces. This occurs in the electric motor where carbon brushes run against metal slip rings or against metal commutator segments. In gas (air) powered tool motors the vanes may be carbon or they may be of phenolic-linen composition which can produce small fragments with high carbon content. The question is whether these particles may become incandescent and serve as a potential ignition hazard.

The literature search shows only two cases translatable to potential space cabin hazards. One case is poor commutation of the electric motor. Poor commutation can be caused by vibration of the brushes, mechanical and electrical defects in the motor, high altitude effects in the brush, etc. Under poor commutation, streamers of hot particles thrown out from under the brush are observed. These are organic impregnations in the brush material which are heated by sparking and arcing during the deficient commutation. A similar mechanism was observed by Buckley,[21] whose group investigated sliding carbon wear surfaces on metal both with and without an electrical potential across the carbon-metal interface. Fires in combustible mixtures were generated by incandescent wear particles in these tests, but only when an electrical potential was placed across the carbon-metal interface. For incandescence, the values of voltage and current had to be above 106 Vac and 0.3 ampere. An electric power tool in normal use would carry more than this minimum current, but dc power tools would not be expected to carry this order of voltage magnitude. The hot incandescent ejecta particles observed in electric motors undergoing poor commutation may be caused by momentary surge voltages generated by the motor circuit inductance and capacitance, which can operate during transient high-load electrical conditions and create surge voltages above this minimum. Under this condition the brush may be operating as a special case of the "break" spark found in the ordinary switch.

Electrical power tool motors are known to offer the hazard of hot incandescent ejecta particles from the carbon-brush metal interface. The gas powered vane motor is not known to show this source of ignition hazard when operated with compressed air.

Solid state reaction sparks. There are two solid-state chemical reactions which are possible within a space cabin. These solid-state reactions would not be directly dependent on the oxygen atmosphere and would therefore be an equal risk as a spark ignition source within or without the cabin. The two reactions are:

$$Fe_2O_3 + 2Al \rightarrow Al_2O_3 + 2Fe + \Delta H \text{ (Exotherm)}$$

$$Ni + Al \rightarrow NiAl + \Delta H \text{ (Exotherm)}$$

These two reactions are considered not only because they are thermodynamically capable of producing incandescent particles but also because the solid phases necessary for the reactions may be found throughout the construction of the spacecraft. The iron oxide reaction with aluminum is characteristic of several metals that can be replaced from their oxide lattice crystal formation by the very active aluminum atom. The oxides of manganese, chromium, vanadium, lead, and nickel are also capable of "thermite" reaction with aluminum. Aluminum reacting with iron oxide is the reaction which offers the greatest hazard potential since iron alloys and aluminum are frequent materials of construction within the space cabin and power tools.

The oxide of iron forms normally on most steels and, since the reaction product occupies considerably more volume than the unreacted iron, has little adherence to the underlying base metal, the oxide is usually found scaling off as small particles. The oxides of iron progress from FeO to Fe_3O_4 and finally to Fe_2O_3, all of which will react. Such small particles, when struck by or against aluminum could become incandescent thermite spark sources. That impact is sufficient to ignite these sparks was quite well pointed out by Kingman et al.[22], working with aluminum paint on rusty steel. These investigators found no difficulty in striking thermite sparks of sufficient incandiveness to ignite combustible gases. All that was found necessary was two reacting components and sufficient impact to start the reaction. Sources of energy such as electric sparks, hot surfaces, friction, and shear would also initiate the reaction in air. The most expected reaction would be between Fe_3O_4 or Fe_2O_3 aluminum. The typical exotherm is:

$$3Fe_3O_4 + 8Al \rightarrow 4Al_2O_3 + 9Fe + 793 \frac{\text{kcal}}{\text{kg mol}}$$

Based on this exotherm the reaction can be classed among the high heat fuel reactions and such small incandescent particles would be a definite ignition hazard.

Visible areas of rusty steel such as those with which these researchers worked are not expected inside a space cabin. However, some quantity of oxidation product from the several steels present could be expected and the presence of larger areas of aluminum would also present a situation favorable for the reaction. Such situations might be found where the aluminum housing of the hand tool (gas, electric, or hydraulic powered) might be struck against a steel surface. The ordinary impact metal-sparking characteristics of aluminum against iron are considered to be a "safe" or "non-sparking" combination (discussed elsewhere in this report as metal "strike" sparks); however, slight iron rust would change this.

While oxygen does not enter directly into this reaction, the oxygen atmosphere would be conducive to formation of iron rust; thus the cabin atmosphere is indirectly involved in forming one of the reactants.

Other areas where this reaction might offer danger in tool operation would be inside the air tool rotating mechanism and at the anvil-hammer interface on ordinary impactor mechanisms. The vane-to-housing interface in the air tool usually finds the vanes running a tight fit with high rotational speed against the aluminum tool housing. A particle of Fe_2O_3 would have opportunity to find sufficient aluminum and initiation from impact/shear for thermite reform of steel. Here an aluminum flake could find both impact and iron oxide particles on the impact surfaces.

The nickel aluminides are formed as a metallurgical solid-state reaction from pure nickel and aluminum.[23] The reaction is highly exothermic and small particles can become incandescent. This material is used as a substrate bond coat in sprayed metal systems; part of its unique ability as such a universal bond coating is that it arrives on the metal surface in such a highly active exothermic condition. When pure aluminum powder particles coated with pure nickel are sprayed through an ordinary flame spray gun the metal particles are observed to be of maximum incandescence beyond the hottest part of the flame. They are found, in fact, to increase their temperature after passing through the flame because of the high exotherm of the Ni-Al solid state reaction. A temperature of 922°K (1200°F) can initiate this exotherm but no information is available indicating the impact sensitivity of the reaction.[23,24]

A complete metallurgical solid solution series is formed between 100 percent aluminum toward 100 percent nickel. This metallurgical formula is usually written:

$$(1-X)Al_{(S)} + X\ Ni_{(S)} \rightarrow Al_{(1-X)} + \Delta H$$

where X = Atom fraction of the component
and S indicates reaction in the solid phase
ΔH = Enthalpy change in cal/g-atom

In the Ni-Al series four intermediate pahses are known; all combinations are highly exothermic. Any combination above 0.1 mole fraction of either material in the order exceeds 16 736 joule/g-mole (4000 cal/g-mole) in value for ΔH. For most exothermic solid-state pure metal reactions 16 736 to 25 104 joule/g-mole (4000 to 6000 cal/g-mole) is a high exotherm. The formation of nickel aluminides fines a maximum exotherm between 0.4 to 0.6 mole fraction of nickel reacting with aluminum, with the peak exotherm going over 58 576 joule/g-mole (14 000 cal/g-mole).

Aluminum is a common space cabin material, but free nickel is not as common. Nickel is sometimes used to plate aluminum which is to be soldered or have a low temperature braze accomplished later. A spark drawn to such a housing would raise the temperature of the reactants and initiate the self sustaining exotherm. No data on the action of these two materials under impact is available from the literature. Comparing the thermodynamic data of the reaction, the reactants, and the products of the nickel-aluminide to the thermite reaction suggests that Ni-Al would be an impact-sensitive reaction. As such it would operate much as the strike sparks for sparking metals or for impact generated thermite sparks. In the range of approximately equal mole fractions, especially when small flakes of either material were to be brought into intimate contact with the other reactant, an incandescent spark source could result which would be capable of serving as an ignition source.

The pure chemistry and metallurgy of these two different solid-state reaction ignition hazards give only a part of the true picture. The conditions of 100 percent oxygen in the space cabin atmosphere and the EVA use of space power tools bring up additional physical considerations which will now be considered.

Atmospheric constituents do not enter directly into either thermite or Ni-Al reactions. Once initiated, they occur euqally well in any atmosphere, in any inert gas, or in the space vacuum (EVA) condition. Their direct reaction hazard does not depend on the presence of freely available oxygen. However, oxygen and the hard vacuum will affect the physical aspects of these reactions. First, the iron oxide reactant which is expected to be the only probable thermite reaction is formed from free atmospheric oxygen. The ignition hazard of this particular thermite reaction depends on the prior atmospheric oxygen history of the iron rust source.

Since these solid-state reactions are hazard ignition sources the total risk involves the potential for developing a fire in other fuel or combustible materials. The presence of free atmospheric oxygen will determine the potential effectiveness of these hot particles as fire starters. In the space vacuum the risk of fire from ignition by these particles is much lower; perhaps the only risk here is the potential burn-through of the pressure suit.

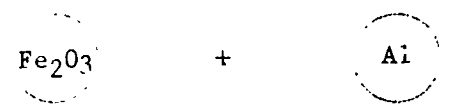
The oxygen exposure histroy of the aluminum reactant in both reactions will also operate in a very special way to influence the potential hazard. The representation for these reactions is usually written as the chemical reaction:

$Fe_2O_3 + Al \rightarrow$

or

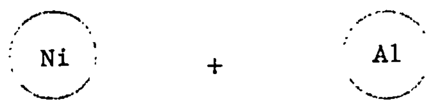
$Ni + Al \rightarrow$

These type formulae alone imply the physical conditions:



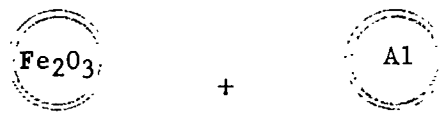
Iron rust particle — Pure aluminum reactant particle

or



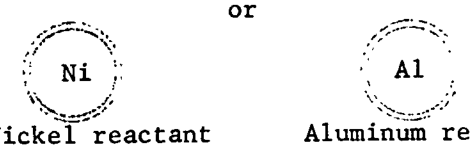
Nickel reactant particle — Aluminum reactant particles

These are not true physical states of materials under the usual Earth-bound or space cabin atmosphere reacting conditions. A more precise physical picture would be:



Iron rust particle with adsorbed film. — Aluminum reactant particle with aluminum oxide film & adsorbed film.

or



Nickel reactant material with adsorbed film. — Aluminum reactant material with oxidized aluminum film & adsorbed film.

These sketches represent the more correct physical conditions of the atmospheric reactants.

The adsorbed films are only lightly held with forces on the order of Van der Wall's bording levels. The most tightly held of these would be the polar adsorbants such as water. These films are removed with only small energy levels, can be penetrated rather easily, and do not serve to influence the reaction strongly. They are effectively destroyed by heating to a few hundred degrees. The aluminum oxide film is quite different from these other adsorbed films. The forces bonding the Al_2O_3 within itself and to the base metal are quite high and very stable. The oxide surface is chemically inert and a highly refractory material. It is not broken or penetrated easily and if penetrated will reform almost immediately in the presence of oxygen.

There is no large volume change for aluminum oxidizing to aluminum oxide; therefore, there is no inherent scaling off of the oxide film as occurs during the formation of rust and other oxides. The result of increasing the energy level (heating, etc.) in the presence of atmospheric oxygen is to increase the diffusivity reaction of oxygen through the Al_2O_3 film; and consequently increase the depth of the protective alumina film. In the research by Pilling and Bedworth,[25] the actual reaction of metals with oxygen in forming or not forming a protective film is indicated. Metals fall into two categories; the category to which aluminum belongs is among metals which do not ignite until after they melt. Melting and resultant liquid mobility causes rupture in the protective oxide film; effective high-temperature combustion (rapid oxidation) is also suppressed by the adherent alumina film and does not proceed until that film formation process is disrupted. The basic reactivity of pure aluminum metal is much higher than the reactivity found and expected in Earth atmosphere use. This reduction of apparent reactivity and the performance and uses of aluminum on Earth is dependent on the peculiar combination of properties of the aluminum oxide surface, and the normal atmosphere in which we ordinarily use aluminum.

In the space vacuum we may have the physical conditions:

$$Fe_2O_3 \quad + \quad Al$$

or

$$Ni \quad + \quad Al$$

Here pure aluminum surfaces would not form or reform the protective film. Under conditions of hard vacuum the thermite and exothermic metallurgical reactions could proceed with:

1. Lower initiation energies than these same reactions in Earth atmospheres.

2. Higher reaction rates than these same reactions in Earth atmospheres.

Considering that these exotherms are equal to or above the energy levels for many fuel-oxygen combustion reactions they should be investigated as carefully and approached with as much caution as the increased risk of fire in 100 percent oxygen. Under space vacuum, in drilling through aluminum with a steel bit or in cutting or shearing aluminum sheet with high-iron alloy blades, in chiseling, hammering, and other maintenance operations, whether by powered or hand tools, the extra reactivity of non-oxide aluminum surfaces can offer a greater potential hazard for producing incandescent solid-state reaction sparks than the same operations conducted inside the cabin or in Earth atmosphere.

Hot Surfaces.

Gas and hydraulic power. A survey was made of the minimum plate ignition temperature for various combustible fluids and gases which might be expected within the space cabin. This survey showed that under the maximum oxygen concentration expected in any space cabin the minimum ignition temperature for hot surfaces would be 461 to 478°K (370 to 400°F). From this the maximum safe surface temperature for any exposed tool surface was considered to be 408 to 422°K (275 to 300°F) in this report.

The continuous flow of fluid in both hydraulic and gas powered tools reduces the potential tool housing surface temperature rise well below the ignition temperature during normal operation of the tools. These two types of tools also show safe performance under high-load or stalled-loading conditions. Gas and hydraulic powered tools offer no hazard as potential hot-plate ignition sources under either normal or overload operation.

Electric power. The electric tool has no internal flow of fluid which will carry off heat. The tool generates heat from two major sources, frictional heat in bearings and brushes and electric power resistance heat (I^2R). Heat dissipation is by convection to the atmosphere and by radiation. Space conditions affect frictional heat, developed mainly in the brushes, and both major methods of heat dissipation. The interrelation between these factors is complex; therefore, the several vacuum tests on previous dc electric power tools and the well documented performance of brushes under vacuum were consulted to determine whether the maximum hot-surface temperature of 422°K (300°F) could be expected under vacuum conditions.

USAF report TDR-63-4227 documented the vacuum tests run on a 186-W (0.25 hp) electric impact tool developed by the USAF for space uses. Tool housing temperature was measured at six points on the tool housing and an atmosphere of near vacuum was held 6.7×10^{-3} N/m^2 (5×10^{-5} mm Hg). The test was run for over 2 hours on a duty cycle of 4 seconds on and 3 seconds off. A locking device prohibited the output shaft from rotating, which simulated a maximum load condition. During the two-hour test the thermocouples showed a maximum rise in temperature from around 292°K (65°F) to 353 to 355°K (175 to 180°F).

During this test several problems were noted concerning the operation of the commutator brushes. Increased arcing was observed in all tests. In some tests the brush arcing was considered excessive and the test was halted because of it. Motor brushes were replaced with brushes developed by Stackpole Carbon Co. especially for exposed high-altitude use. These brushes also arced noticeable during the vacuum tests. Although the tests were not halted by test personnel because of visible arcing, the motor did run erratically during tests using these special brushes.

Inspection after the motor stalled showed that the copper brush leads had softened during the test and the brush solder had also melted. Since most solders melt at temperatures above 408 to 422°K (275 to 300°F) and the softening point of copper is also over this temperature, the indication is that these motor brush temperatures were higher than the safe surface temperatures for hot-surface ignition. The brush leads and solder are near the top of the brushes away from the running friction surface. It can be expected that the friction surface, the area on which the friction heat and resistance heat losses are developed, attained a higher temperature than did the soldered end of the brush.

The Martin Company also developed other electric power tools for NASA and for experiments on the Gemini flights.[5] Tool housing temperatures were measured during vacuum tests in both development programs. Operations were usually conducted so that the tool was overloaded beyond its expected use. Case temperatures did not exceed the 344 to 355°K (160 to 180°F) levels and show that the overall motor housing is not normally a hot-surface ignition problem. Problems were encountered in the same area of motor stalling, unexpected low speed operation and excessive electromagnetic radiation (interference). These problems were traced to the motor brushes. No information as to the temperature of the brushes was given in these two tests.

The higher brush temperatures and erratic brush operation experienced in all these tests can be expected. All the available literature on brushes shows that special problems are inherent in brushes operated at high altitude These special problems have had considerable attention since the early 1940's. Since then regular operation of large numbers of motors on aircraft at high altitudes has been common and the problem and solutions have been studied.

The major basic changes in high-altitude operation of brushes comes with increased arcing resulting from low gas pressure (already discussed in the "switch sparks" section of this report) and the increase in friction of sliding solid lubricants operated at very low pressures. The basic studies have shown that solid lubricants depend on small amounts of certain adsorbed impurities to show the low friction characteristics.[15,20,28]

Carbon brushes, with the proper solid-state lubricant adjuvants, show normal friction coefficients of 0.1. In high altitude operation, where all water of hydration is driven out of the brushes, the friction coefficients will suddenly increase to values 5 to 10 times normal. The result is catastrophic wear, excessive heat, and generally poor operation of the brushes.[29]

The potential (voltage) drop across the brush-collector interface is also subject to small changes in the constriction resistance, which is in large measure a function of the condition of the solid films formed at the interface. Thus, constriction resistance becomes important in determining the electrical temperature generation. In the case of high-altitude operation, the temperature is dependent on only small changes in the constriction resistance. This dependence is shown in Figure 7.

Considering that both friction heat and electrical heat generated at the brush-collector interface are higher and that either heat source may exceed the maximum safe ignition temperature (also considering that convective heat transfer in the vacuum is practically nil), it should be expected that the brush troubles evidenced in the tests are to be incurred under vacuum operation.

Under stalled rotor or other failure mode of operation the tool housing surfaces would become excessively hot, but only after several minutes under stalled conditions. Since the tool will be hand held in use, continued long-time operation at stall can be avoided. The effects of shorted windings and even stall overload can be avoided by fusing the tool. The pressure suit and other protective outer gear and operation within a vacuum will prevent the astronaut from easily sensing a tool housing temperature rise. Therefore some additional defice(s) should be built into the tool to indicate tool housing temperature to the astronaut during use.

The Power Tool as a Source of Fuel

Metal Fuels. The major mass of material is composed of the metals of which the tools are manufactured and the metal structures on which the tools are used. Even though metals are not generally classed as fuels, we must be concerned with the fuel potential of metals in combustion reactions. From the standpoint of heats of combustion, metals compare favorably with recognized fuels. Table III shows the heats of combustion of several spacecraft materials compared with several fuels.

In determining the relative value of a material as a practical fuel or as a hazard as a fuel, other properties must be considered. The reaction products of the combustion reaction have a large influence on the character of the

FIGURE 7

CONSTRICTION RESISTANCE AS A FUNCTION
OF TEMPERATURE

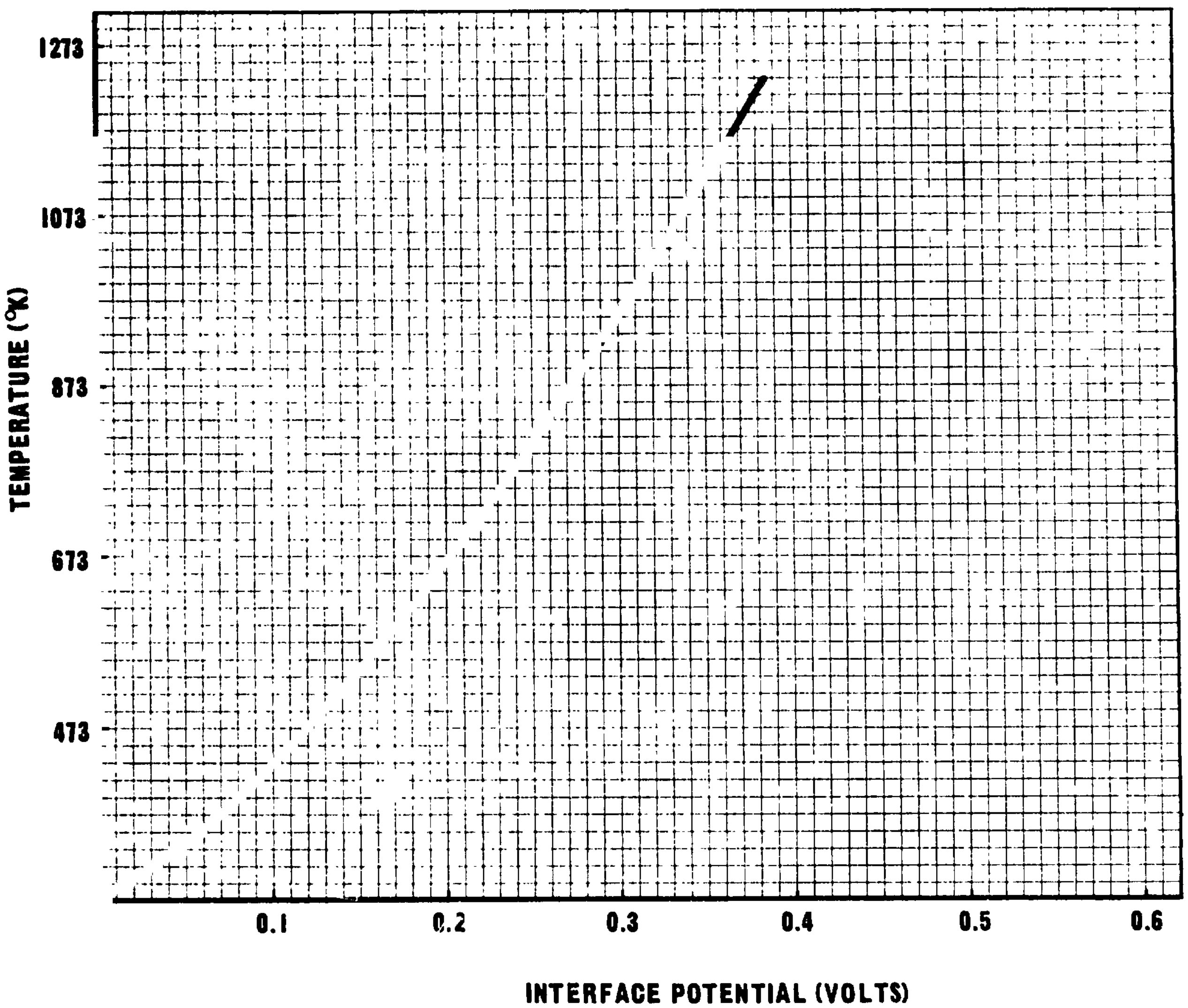


TABLE III. HEATS OF COMBUSTION

Fuel	Combustion Products	At. or Mol. Wt.	J/kg Fuel	BTU/lb Fuel
Carbon (C)	CO_2, CO	12	32 768 400	14 100
Methane (CH_4)	CO_2, H_2O	16	55 776 000	24 000
Acetylene (C_2H_2)	CO_2, H_2O	26	48 804 000	21 000
Beryllium (Be)	BeO	9	67 396 000	29 000
Magnesium (Mg)	MgO	24	25 564 000	11 000
Aluminum (Al)	Al_2O_3	26	30 212 000	13 000
Titanium (Ti)	Ti_2O_3	47	15 803 200	6800
Iron (Fe)	FeO	56	4 648 000	2000

reaction. Combustion of the common fuels listed above, such as carbon, methane, and acetylene, produce gaseous combustion products which do not interfere with the access of oxygen to the remaining hot fuel surface. Magnesium is a metal fuel which has a solid combustion product (MgO). The cubic oxide crystal particles formed, however, are not compatible with the hexagonal crystal structure of the basic metal, and evolve from the combustion in a cloud of smoke composed of small white particles. This process is very similar to the combustion of the carbonaceous fuels listed above, and no interference is offered by the MgO product to the further combustion of the parent metal surface. Magnesium is therefore a threat as a metal fuel. This particular reaction is well known and usually is a major consideration in projected aerospace uses of magnesium.

Aluminum's reaction with oxygen has been mentioned previously. The reaction product is tightly adherent to the base metal and serves effectively to block further rapid oxidation at normal temperatures. The temperature of aluminum must be brought well above the melting point of 933°K (1220°F), or to about 1273°K (1832°F), in the highly molten state, before the aluminum oxidation reaction will proceed as a self-supporting oxidation reaction; i.e., before combustion will proceed.[30] While aluminum, from a chemical standpoint, is as reactive as magnesium, and aluminum will supply more heat per pound of oxidized metal, because of the basic nature of its combustion aluminum is not a practical fuel. As a safety hazard, combustion of significant amounts of aluminum will not proceed unless there is another large combustion reaction preceding the high-temperature combustion of molten aluminum. Therefore aluminum is not a practical safety hazard.

Titanium is similar to aluminum. Although the oxide reaction product is not as adherent, it will tend to protect the metal. Research on titanium combustion in 100 percent oxygen showed that titanium would ignite spontaneously under static conditions in pure oxygen and at pressures of $24.13 \times 10^5 N/m^2$ (350 psi) or more, but only if a fresh surface is created, such as by scraping the surface. Under dynamic conditions, such as material rupture under stress, the same condition applies, but down to pressures as low as $34.47 \times 10^4 N/m^2$ absolute (50 psia).

The practical value of any metal fuel, then, is complex and will depend on the heat of reaction; rate of reaction; ignition temperature; stability, physical, and chemical nature of the reaction products; dissociation pressure and heat capacity of the reaction products; and the specific conditions under which the fuel and oxygen are supplied. For solid metal fuels, no special differences in their conbustibilities can be found in space flight except for the potential for a much higher reaction rate in 100 percent oxygen. Safety considerations as are usual in aerospace work should be sufficient in selecting metal materials for power tools.

Powdered metals offer a very different degree of hazard as a fuel supply than do solid metal fuels. Powdered metals dispersed in air form explosive mixtures with ignition temperatures much lower than those of the corresponding bulk metals.[30] Since a relatively high ignition temperature is the one common characteristic which puts most solid metals out of the class of a practical fuel hazard, the drastic lowering of ignition temperatures in the finely divided metal powders brings powdered metals definitely into the high-hazard category. Higher reaction rates when exposed in 100 percent oxygen atmospheres will add to this hazard. In addition, under zero or subgravity conditions the larger particles of metal will tend to "float" and remain a part of the dispersed metal powder, with no natural falldown of such particles. Therefore, any process which produces small metal particles will be accumulative toward a hazardous situation. Production of metal chips and metal powder by tool operations, unless completely controlled, will bring about a definite safety hazard. Such operations will produce a dispersed metal fuel with inherent high heat release and inherent high rates of reaction, a fuel which can be ignited by common ignition sources. Such dispersed metal fuels should be considered as a hazard equal to highly combustible gaseous mixtures.

<u>The Electric Power Tool as a Source of Fuel</u>. Provided the power tool is manufactured of metals ordinarily reasonably safe as fuels under 100 percent oxygen(i.e., not made of magnesium), then the only potential fuel supply from an electric tool will be contained in the content of the lubricant and the wiring insulation.

The amount of lubrication required in an electric power tool is small and under normal circumstances solid lubricants are used. Polytetron fluorethylene (PTFE) and Teflon represent almost completely nonflammable lubricating materials. These materials are ignition-safe even in 100 percent oxygen. Dry molybdenum disulfide will not burn in air, but incandesces slowly in oxygen. Under these conditions the binders are expected to be the fuel contributor and may be controlled by specification to the MoS supplied. Tri-cresyl phospate is accepted by the Canadian Fire Research Organization as the lubricant for work in oxygen, even though it will burn slowly under 100 percent oxygen. The quantity required in small electrical power tools is so small that this lubricant does not present a fuel hazard.

There are several military standard and commercial standard types of wiring insulation which are accepted as nonburning or generally considered noncombustible. Some of these materials will burn in 100 percent oxygen. Those that will burn include polyvinyl chloride, glass fiber and asbestos. PVC is typically one of the class of safe materials in air but unsafe in 100 percent oxygen. The glass fiber and asbestos materials are not inherently unsafe; they will burn only to the extent that they contain certain binders added in their manufacture which will burn in 100 percent oxygen but will not burn in air. Teflon and PTFE appear as good noncombustible electrical insulators along with specially made glass or asbestos materials.

The mass of fuels represented by the electrical insulation is large enough so that it does offer a threat as a fuel. Insulation must be carefully selected and tested under 100 percent oxygen conditions so that this threat is removed from the tool in its original design stage.

<u>The Gas Power Tool as a Fuel</u>

Source. Provided the gas power tool is manufactured of metals ordinarily reasonably safe as a fuel when used under 100 percent oxygen (i.e., not manufactured of magnesium), then the only potential fuel supply from the gas tool will be contained in the lubrication required.

The requirement for lubrication in the air tool is critical to its operation and is a continuous "flow through" type requirement. The last item in the air line feeding the air supply to the tool will be the lubricant reservoir and injector. Additionally, the tool may have its own internal lubrication reservoir. Most of these systems operate with the amount being supplied giving an "oil-wet" exhaust condition, when being properly lubricated. This excess oil is carried off with the exhaust air and diluted into the terrestrial shop atmosphere.

Lubrication in the gas tool performs several vital functions other than reducing friction and wear.[6,31,32] The liquid lubrication medium is used as a thinline liquid pressure seal running at the vane-housing interface. It also assists in removing heat directly from the sliding vane-metal housing friction surfaces, and it flushes out particles in the motor.

Oil is vitally necessary for developing power in the air tool. This is illustrated by following the progressive deterioration sequence of an air motor running without lubrication:

1. First, there is a drop in speed and power immediately upon losing the pressure sealing function in the motor.

2. The cylinder liner heats as a result of increased blade friction and the resulting charring of the vane-blade.

3. An additional power drop which is caused by the char and dirt, which is no longer flushed out.

4. There is scoring and excessive wear by accelerated abrasion.

5. Further damage is done by worn blades riding at an angle and gouging the housing liner or by blades breaking and chipping off.

The minimum amount of oil usuage appears to be two drops per minute 0.0024 m^3/s (25 cfm) being used. Approximately 0.00316 $m^3/s/W$ (50 cfm per horsepower) is required in the small fractional size motors. For a motor with 186 watts (0.25 horsepower), about 0.0057 m^3/s (12 cfm) will be used near load speed, or an oil usage of one drop per minute per tool. If only a small percentage of this amount of ordinary lubricating oil accumulates in some part of the tool, it will represent a very hazardous fire situation. At some point the oil must be exposed to the 100 percent IVA environment, unless the gas exhaust system is completely sealed and vented overboard. Such an oil vapor in 100 percent oxygen is one of the most volatile fuel-oxidizer mixes available.

The gas power tool offers a fuel supply hazard that is inherent in the relatively large amount of "flow through" lubrication. The safety hazard is offered as pooled oil within the tool or as a vaporized fuel-oxidizer mix within the cabin.

The Hydraulic Power Tool as a

Fuel Source. The hydraulic power tool will have whatever fuel hazard is offered by the type of prime power used, whether electric or gas drive. In addition, it will have its own special potential contribution to the fuel hazard.

The major contribution to a combustible fuel supply in all the types of power tools considered would easily come from the hydraulic power source. The amount of fluid circulating would cause a major problem if damage to the tool occurred. Ordinary leaks that are standard in such equipment would be excessive within a closed ecological system. The condition of being able to collect easily at a single point and to vent this effluent from ordinary leaks would not be as readily accomplished in the hydraulic system as it was for the pneumatic.

Research has shown that the conditions in space cabins greatly increase the danger of conflagration should hydraulic fluid be let into the cabin. From this research, Figure 8 shows that all hydraulic fluids decrease in spontaneous ignition temperature (S.I.T.) when the atmospheric oxygen content increases. This decrease is from S.I.T. of 644 to 672°K (700 to 750°F) for most high-temperature hydraulic fluids at normal oxygen concentrations to 519 to 533°K (475 to 500°F) at high (space cabin) oxygen concentrations. These so-called high-temperature fluids ignite at about the same temperature as the standard fluids under the high-oxygen condition.

The combination of high probability and the consequences of a hydraulic system fuel fire was considered to be too great to allow the use of hydraulics within the space cabin.[2] Another USAF study (as yet incomplete) recommended that only solid, brazed, pressure-tested hydraulic joints be considered, and that no flexible hydraulic lines be allowed inside the manned compartment.

The conclusion is that probably the most serious fire hazard is offered by the hydraulic power source. From a safety standpoint, this source is probably too dangerous for consideration.

Toxicological Hazard

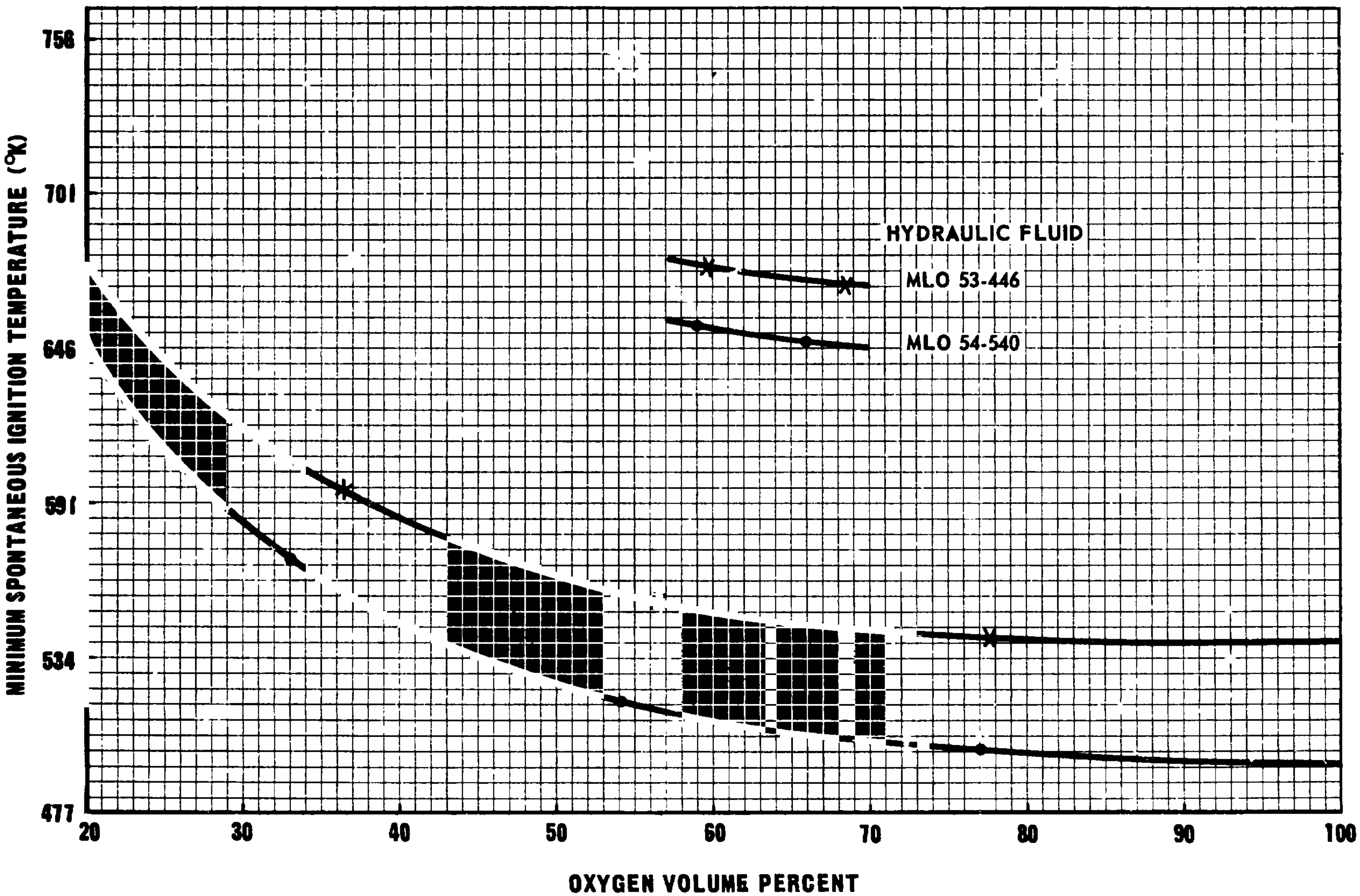
The Electric Power Tool as a Toxic Producer - The electric power tool has two conditions under which it will produce potentially toxic substances, ozone produced at the brush-commutator interface, and pyrolysis products from overload failure or electrical fire in the tool.

The production of ozone from oxygen fed through an electrical arc is a well-known phenomenon. The increased susceptibility toward arcing and sparking at the brush-commutator interface at low atmospheric pressures is also well documented. These two facts would support the potential for the brushes in an ordinary electric motor to produce ozone during use.

The Martin Company, in developing the electric-drive Multipurpose Space Tool for both the NASA and USAF programs, tested their tools for ozone production. These tests as described were not complete enough to give assurance that all such electric tools, or even those tested, were sufficiently free of production of harmful amounts of ozone. This remains a little-defined problem which must be fully proven by test before any open-brush motor is declared sufficiently free of ozone production to be safe from this hazard.

FIGURE 8

MINIMUM IGNITION TEMPERATURE AS A FUNCTION OF OXYGEN CONCENTRATION



The pyrolysis products from many electrical wiring insulations are more dangerous than the small fires which produced them. While the fire may be small, the contribution within a small-volume closed ecological system by pyrolysis of the windings of only one motor would add to, approach, or exceed the threshold limit values set for these products. Most toxicological threshold limit values are based on 8-hour exposure for a working week. Toxicologists agree that these values must be drastically reduced within the space cabin. Here the exposed continuously, and the accumulative effects of many toxicants never before placed simultaneously must be considered. In addition, the environmental control systems have only limited capacity to remove solids and certain gases, and may be overloaded with only small amounts of unexpected toxicants.

Part 1 of the Space Cabin Atmospheres study is devoted entirely to oxygen toxicity. Under the exposure conditions of spaceflight, especially long duration 100 percent oxygen, even oxygen becomes suspect as a toxic gas. The additive effects of toxicants plus the relatively small real time experience for humans under such conditions requires the reduction to zero or near zero for foreign toxic materials.

The electric tool may produce ozone at the brush-commutator interface in amounts which can be dangerous and under fire-failure mode the pyrolysis of electrical insulations may be a greater hazard than the fire itself.

The Gas Powered Tool as a Toxic Producer - The lubrication added to the tool appears as a toxic irritant even in small amounts, and in accumulated running of a single tool over one hour (in the Apollo volume) will represent a systemic toxic level.

The gas power tool produces a toxic condition inherent in its present design and operation.

The Hydraulic Power Tool as a Toxic Producer - The hydraulic power tool will have the hazard inherent in the type of prime power driver. In addition, it will have the added hydraulic fluid from small leaks. For present hydraulic systems this would amount to a quantity added of approximately one-fourth that produced by the gas power source.

Special Tool Hazards

In addition to the hazards listed above, these power sources offer certain special hazards.

The electric tool may offer the electric shock hazard. The past electric tools, which were operated under 15 volts, do not offer a practical electrical shock hazard. Deciding precisely at what voltage we get into this type potential hazard may be difficult, but if operated under 30 volts, it seems that no practical hazard exists.

The hydraulic tool offers a special hazard. The hydraulic fluid is operated at high pressures; if such a high-pressure line is broken, the high pressure spray of hydraulic fluid near the break is found to have a much lowered spontaneous ignition termperature. The effect is very similar to diesel injection ignition. This increased risk adds to the large fire hazard already discussed concerning this type power tool.

SUMMARY AND CONCLUSIONS

This study has shown that hydraulic, electric, and gas power sources offer safety hazards when used as space power tools.

The hydraulic power system must be driven by either an electric or gas power source and will therefore have the inherent disadvantages of the chosen prime power source. The hydraulic system offers a major fire hazard, especially should a failure occur in the hydraulic supply system. Because of this hazard it is recommended that hydraulic systems not be considered where there is any IVA requirements.

The gas power tool offers several hazards. Any tool housing manufactured of high-aluminum alloys will have a natural insulating, auto-oxidized surface or may have any extra thick anodized surface of aluminum oxide. Even if the aluminum is electrically grounded, this surface film will be potentially effective in allowing an electrostatic spark to build up.

There are two solid-state reactions which can occur to create incandescent hot particles. Both reactions involve aluminum metal. One reaction is aluminum with iron oxide (rust); in the other aluminum is reacting with nickel. Both will operate in either the presence (IVA) or absence (EVA) of gaseous oxygen. Both reactions may be initiated at lower temperatures and may react at faster rates in space because of the lack of free oxygen to form a protective film. Any drilling, cutting, or shearing of an aluminum/steel combination will involve this potential reaction.

These solid state reactions are not a hazard to the air tool but may occur with any tool system where the two reactants are brought into intimate contact.

The major disadvantage peculiar to the gas tool is found because of its high, continuous liquid lubrication requirement. This lubrication represents a fire hazard if accumulated into a small "pool" condition and offers a major hazard as a vaporized fuel expended into the 100 percent oxygen cockpit environment.

Dispersed lubrication accumulating within the environmental control system is also a toxicological hazard.

There are also several hazards peculiar to the electric power tool. Unless the electrical wiring is chosen for its heat-failure mode under 100 percent oxygen conditions, the wiring may offer both a small fuel supply hazard and a larger hazard in its emission of toxic pyrolysis products. The metal case may strike sparks unless "safe" metals are used in its manufacture. The on-off switch may serve as a spark ignition source, especially on the break-circuit condition.

Several hazards occur as a result of the requirement for carbon brushes carrying current through slip rings and commutators. The brushes may create the toxicant ozone; brush arcing and sparking will create radio magnetic interference (RMI or EMI). This arcing may also serve as an ignition source for gaseous combustibles or severely arcing brushes may eject incandescent carbon particles. The hot brush surfaces may also serve as ignition sources. Brushes running i·

vacuum will be generally unreliable and have high or catastrophic wear.[28]

Both the electric and gas power tools offer some hazard when used in space flight. Some research and development must be accomplished if these tools are to be considered as a completely safe power source.

This study has shown several hazards peculiar to tools, and also several that are important hazards not specifically limited to power tools. These hazards will be listed again since they may apply in many aspects of space flight.

1. Production of free metal powder and chips in an IVA environment, especially if the environment is 100 percent oxygen, produces a highly combustible mixture that can be ignited well below the bulk ignition temperature of the metal.

2. There are two dangerous heat-envolving solid-state reactions of aluminum with iron oxide (rust) or nickel metal. These reactions may be initiated at lower energies and have faster, more energetic reaction rates in the hard space vacuum.

3. Grounded aluminum structures may present an electrostatic spark hazard because of the ever present alumina or insulating film on the aluminum surface.

RECOMMENDATIONS

Neither the electric nor the gas power tool is sufficiently safe. In deciding which one of them will be improved there are three major considerations:

1. The development program must resolve the major safety hazards without sacrifice of tool performance or tradeoffs that will introduce other hazads.

2. There should be high confidence in the end result of the development program before it is undertaken.

3. There should be no other outstanding deficiencies in the power source chosen for development which would effectively prohibit its real value for use on active missions.

This study has been conducted within the considerations of safety and reliability. From a safety standpoint all the inherently extra hazardous high-energy fuel cycles have been eliminated. A previous study showed that even using maximum-energy cycles the gas power source was at a disadvantage when compared on a power versus fuel weight basis. With safety considerations limiting gas tools to only low energy pressure-work systems, this disadvantage is increased. From the standpoint of weight requirement, there is a practical question of whether such a power source could be accepted.

Positive predictions of the improvement of the gas tool for use in space flight are also difficult. Where we have extensive statistical background of use of electric motors in many protected and exposed locations on high altitude aircraft, we have no equivalent background on air motors other than terrestrial shop use. Where the difficulties with the electric motor have been extensively researched and basically well defined both practically and theoretically, the development program for that gas power tool would have neither large amounts of practical information nor be based on deep theoretical knowledge about the processes involved.

Suggestions to make use of some special advantage of the gas tool in order to increase its positive value are also difficult. It has been suggested that such a tool use oxygen as the gas, and to supply the vented gas to the cabin atmosphere as the human oxygen supply. The human oxygen requirement is much lower than the tool gas use requirement. On Gemini flights only 47.17 kg (104 lbs.) of oxygen was carried including reserves for 2 men for 14 days.[33] The long-term average use is expected to be 0.9 kg (2 lb.) of oxygen consumed per day for each man. Figure 9 shows typical performance curves of a gas power tool of hand tool size (approximately 186 W or 0.25 hp). The flow-rate requirements show that with 50 percent on-off duty cycle only a few minutes of tool use would produce enough oxygen to satisfy the daily requirements of the astronaut. A more economical gas use may be effected through a development program but it is doubtful whether a gas power source can be brought into energy balance with the gas useage requirements of the human body. Any such scheme of use of the gas effluent from the tool inside the cabin brings the tool directly into the environmental control system loop and will generate additional problems.

The electric power source offers a favorable contrast for possible improvement. Should the major safety hazards be eliminated the electric tool has other very favorable characteristics for use in space flight.

Two development options offer some confidence in being able to make the necessary improvements in the electric power tool.

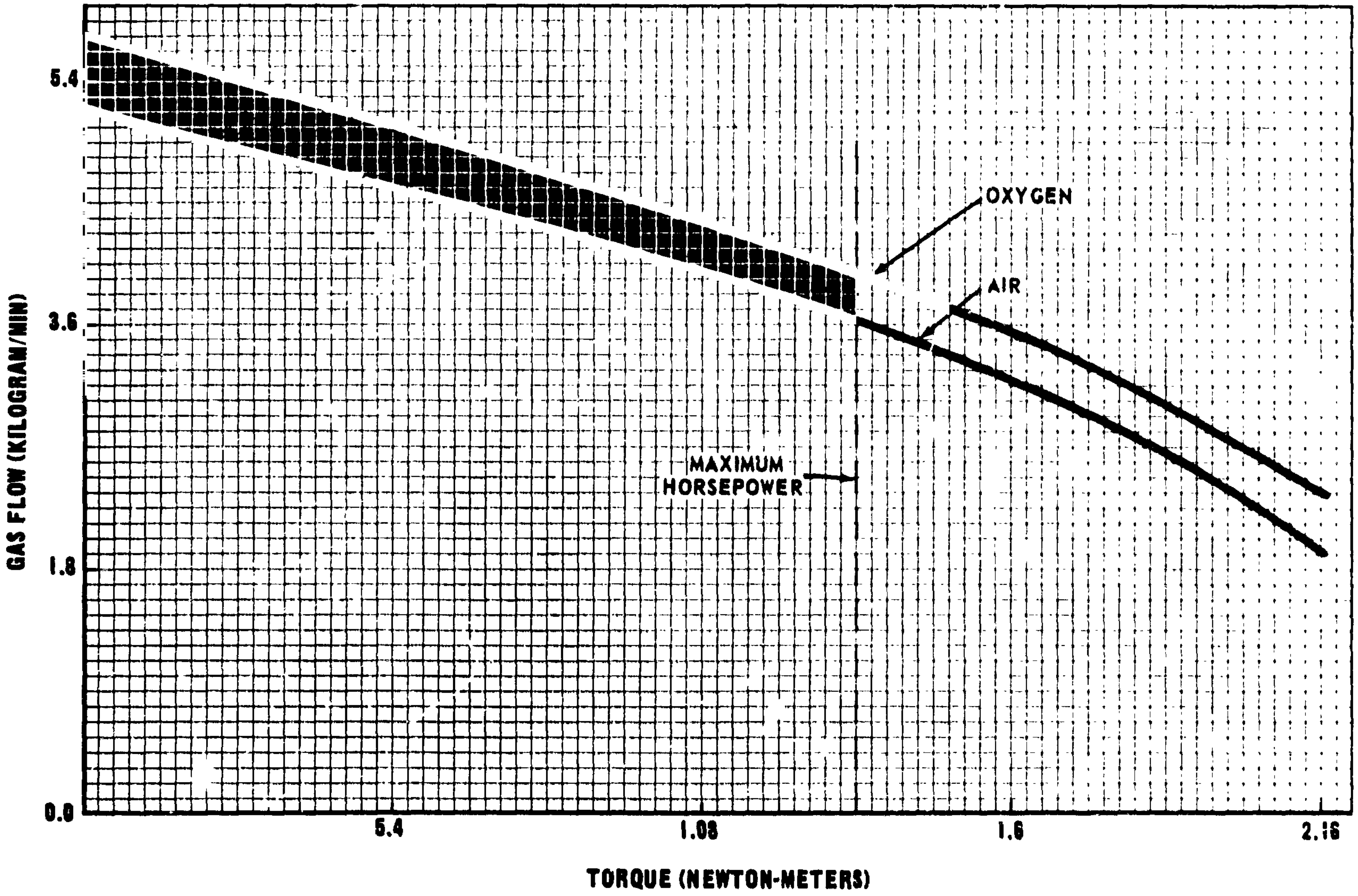
One method would be to adapt the higher frequency ac induction motors, for space tool power source. Such a motor would operate on 300 Hz or higher ac power. Its advantages are known in ordinary shop use to be sufficient to shift some shops to this type tool in spite of the requirement for a frequency changer.[34,35] It would remove most of the problems associated with a brush-commutated dc motor but would introduce the problem of obtaining and using a type of electric power not developed by the prime space-electrical systems. It would require a frequency changer and would also introduce the electrical shock hazard.

The other electrical development option would be to adapt a new type of brushless dc motor, previously developed by NASA for satellite and space power applications, to a configuration and size for use as a power tool drive.

This development was previously accomplished by NASA and the Sperry Farragut Company.[36,37] The objective on the original program was to develop a very small low-wattage motor for special use in hard vacuum space conditions. It is true dc motor and its unique properties and design are based on a photosensing solid-state commutator system. In this design the rotor is a permanent magnet while the stator contains the windings. A small light shield is attached to the rotor which rotates around a stationary lamp. The lamp is operated under derated conditions and a light beam passes out through the light shield. Photodiodes in the stationary commutator section sense the

FIGURE 9

TORQUE PRODUCED BY FLOW OR VARIOUS TOOL GASES



position of the rotor. Signals from the photodiodes control a solid-state amplification and power switching system which energizes the proper stationary armature coils. The input dc power is thus switched and commutated without transfer through a brush or sliding contact and without transferring to a rotating electrical component.

This system will eliminate the brush problems of arcing and sparking, the potential to produce ozone, the hot-brush surface ignition hazard, high friction and wear, and general low reliability of the brush commutator system.

Because of the unique design, other safety improvements can also be accomplished. Since the stationary armature windings do not rotate, it is practical to consider encasing these windings and pressurizing the case with an atmosphere safer than 100 percent oxygen. Such a hermetically sealed motor would have a much lower fire hazard from the electrical insulation but in case of fire could be built to contain the pyrolysis products. Thus even under failure-mode operation, the motor would fail-safe with respect to its potential toxic (fire) hazard.

Solid-state devices are used throughout the main power circuit. This allows the consideration in the design to use a low-power solid-state "gate" switching sub-circuit at the on-off motor switch. If accomplished, solid-state switching would allow the motor to be controlled by a power circuit where the order of magnitude for voltage and current would be 0.5 to 0.75 V and 200 to 300 mA. Accomplishing these low switching values and isolating this switching circuit from the main power circuit may allow the switch to be maintained at all times below both the critical minimum voltage and minimum current necessary to produce the "break" switch spark. This would completely eliminate the switch as an ignition source.

This motor to date has seen several specialized applications since the first low wattage (fractional horsepower) model was developed. There have been at least four different sizes designed for several different uses on satellite systems. Although no design suitable for use as a tool power drive has been produced, the original concept has been scaled up to a 746 W (1 hp) motor pump drive.

There should be no major limiting reason why this type motor cannot be produced in a size, power, and torque range suitable as a power tool drive. Since the solid-state commutator will be heavier and will occupy more volume than the brush commutator, design attention should be given to the possibility that some of the commutator system be placed at the source of power; i.e., in the tool battery housing for a portable system, or at the power plug in a ship-supplied system.

The photodiodes and the power diodes produce some heat and this must be conducted away since these devices do not operate properly at temperatures above 366°K (200°F). Keeping these solid state devices cool may be the only design difficulty, but it should not be a limiting design condition.

Several other features should be included in the motor development and design. The motor should have a simple system to indicate housing and internal temperatures.

This may be a self-powered circuit (thermocouples) with a gauge indicator built into the rear of the case, or temperature indicating paints may be used. Several temperatures points should be measured so that localized heating would be registered.

The power circuit should be protected with a fusing system so that sustained electrical overload would not be possible.

Collateral with safety improvements in the motor, it is recommended that studies be undertaken to:

1. Investigate the ability of the insulating alumina film, formed on grounded aluminum, to store an electrostatic charge. Auto-oxidized and various anodized aluminum alloys, including the heavy hard coat process, should be investigated. Should the practical hazard from such electrostatic sparks be proven through this investigation, then conducting coating of nonauto-oxidizing metal should be developed. This thin coating would modify the surface of the aluminum so that effective insulating films would not form and the surface could be electrostatically grounded. Such a coating would not appreciably alter the favorable properties of weight, strength, and safe strike-spark characteristics of the basic aluminum. It is suggested that the coating could be a 2 percent beryllium copper coating applied by flame spray or vacuum metalizing. In developing the coating a process must be used which will place it onto the conductive aluminum surface and effectively ground potential capacitative spark development.

2. Investigate the solid-state reactions of iron rust with aluminum and nickel metal with aluminum while under the conditions of hard space vacuum. Quantitative values can be placed on the temperature of initiation and the rate of reaction by a research method such as differential thermal analysis while under vacuum conditions. Whatever method of analysis is used it should be based on providing an atomically clean unoxidized aluminum reacting surface. It should provide practical and quantitative theoretical answers to the development of incandescent sparks when performing drilling, cutting, and other operations in space on aluminum with rusted steel tool surfaces.

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THE ROLE OF SPACE MANIPULATOR SYSTEMS FOR EXTRAVEHICULAR TASKS

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SUMMARY: Manipulators have application for tasks such as docking, tethering, grappling, mass transfer, refurbishment, and repair. Advantages of manipulator systems include astronaut effectiveness and safety, continuous EVA with operators working in shifts, and remote operation from space or ground stations. At present, no manipulators are space qualified; however, there is an adequate technology base to undertake their development.

INTRODUCTION

The potential capability of space manipulators was dramatically demonstrated in April, 1967 when the Moon-Digger on Surveyor III successfully touched, tapped, pressed and trenched the surface of the moon. This drag shovel is a simple mechanism having only four step-controlled motions. Crude as it is, it scooped lunar soil samples and deposited them within a quarter-inch accuracy.

The true value of space manipulators -- the utilization of man's adaptiveness and ingenuity -- was demonstrated in January, 1968 when the Surveyor VII Moon-Digger corrected an equipment failure and saved a valuable experiment. This manipulator application of the Moon-Digger was not conceived until after the failure had occurred on the moon.

Recent studies for NASA and the United States Air Force indicate that manipulators may effectively perform extravehicular tasks in space. Time and motion studies by the Argonne National Laboratories (ANL) (Reference 1) have shown that simulated space tasks can be accomplished by a shirt sleeve operator using a master/slave manipulator about equally well as an astronaut in a 3.5 psi space suit. In either approach, the time was about triple that required to do the task directly by hand.

The basic concept of space controlled and ground controlled manipulator systems is illustrated in Figure 1 (see next page). This shows a slave manipulator system on a remote maneuvering unit controlled from either a master station in an SIVB Workshop or a master station on the ground. In addition to these two concepts, this paper will also discuss electric manipulators positioned by a boom (e.g., Cherry Picker or Serpentuator*) and through-wall manipulators.

The Air Force Aero Propulsion Laboratory sponsored "Remote Manipulators and Mass Transfer Study" (Contract AF33615-67C-1322) by the General Electric Company (Reference 2) and arrived at the following conclusions:

1) Space or ground controlled manipulators for space missions are technically feasible within the present state-of-the-art
2) Electric master/slave manipulators based on designs now used in nuclear laboratories can be developed for flight on manned vehicles in five to ten years
3) Manipulator systems should be considered rather than astronaut EVA in space controlled manned missions when:
    a) Extra-hazardous environments or tasks are involved
    b) Endurance is required
    c) Cost, weight or probability of success advantage is demonstrated
4) Manipulator systems should be considered for ground controlled unmanned missions when:
    a) Cost-effectiveness is demonstrated relative to deploying a manned system or relative to total satellite replacement
    b) Hazardous missions are involved
5) Man-equivalent manipulator-TV systems similar to that shown in Figure 2 appear to be optimum for both manned and unmanned missions

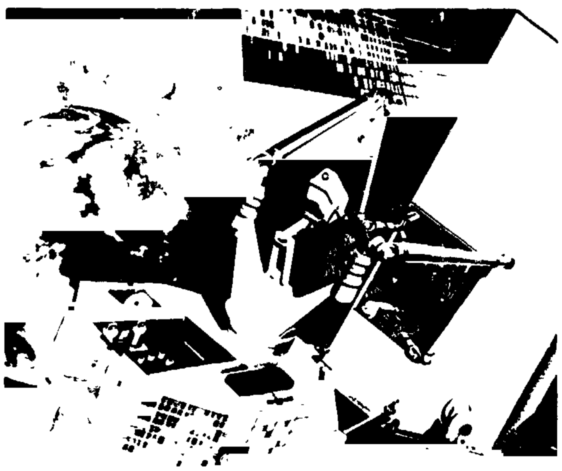


FIG. 2 - MANIPULATOR-TV SYSTEM ON REMOTE MANEUVERING SATELLITE

6) Immediate development is required for:
    a) Time, motion and energy versus task data for mission planning and preliminary system design

*Articulated boom being developed by NASA/MSFC

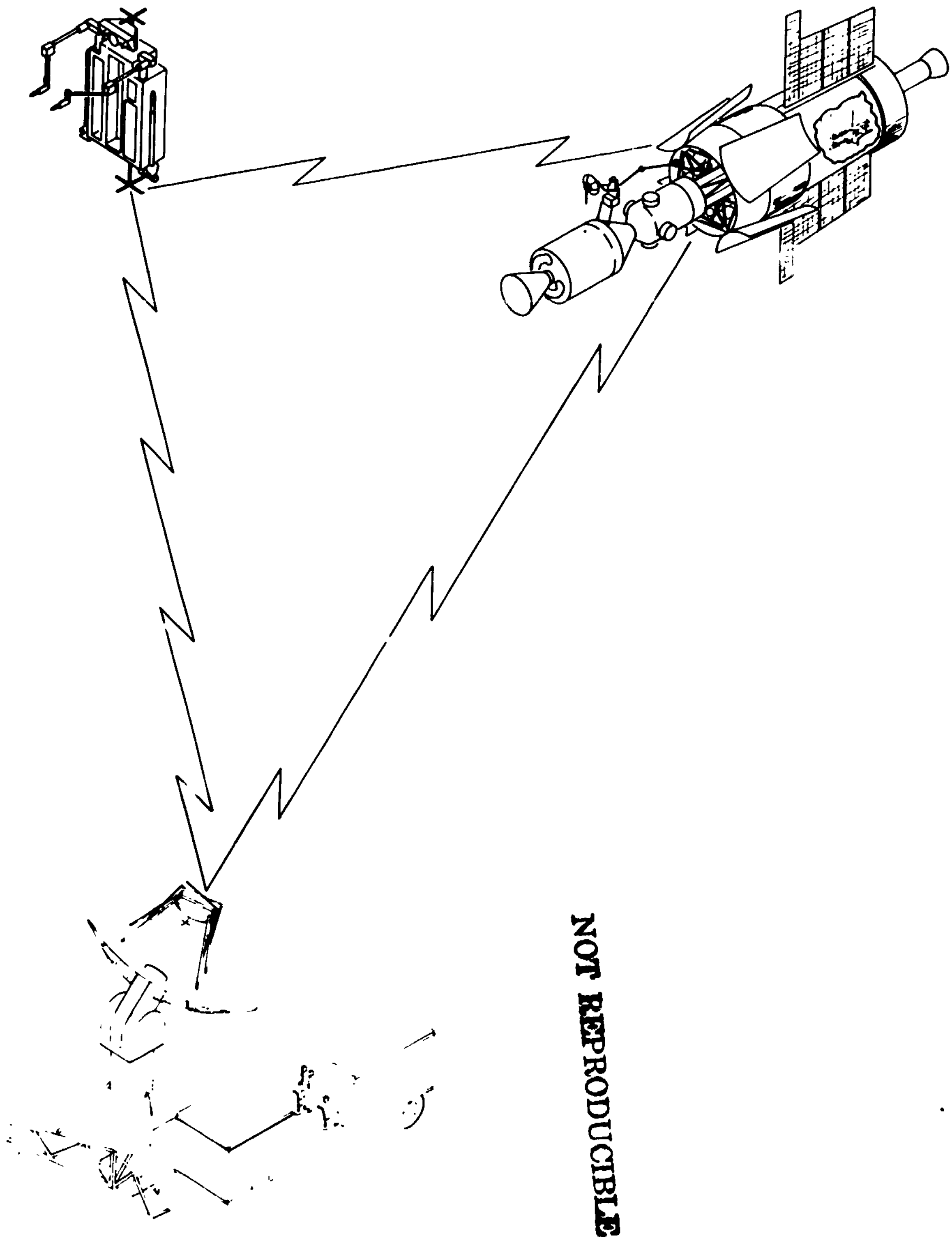


FIG. 1 - MANIPULATOR ON REMOTE MANEUVERING UNIT CONTROLLED FROM SPACE STATION OR EARTH STATION

b) Transmission time delay effects data for control system design and operator training

The above conclusions result from an analysis of work elements in earth orbit tasks and a conceptual design study of manipulator arms to accomplish these tasks. Weight, power and thermal analyses of the arm design shown in Figure 2 show that this simple, passively cooled design can meet the specifications established by the study for a synchronous satellite refurbishment mission. The feasibility of ground control for this system was strengthened by experimental transmission time delay work done in support of this study by Professors W. H. Ferrell and T. B. Sheridan of the Massachusetts Institute of Technology. Their previous work (Reference 3) had shown that two dimensional manipulation with transmission time delay between master and slave could be accomplished predictably with an open-loop (unilateral) move and wait strategy. During the recent subcontract, they demonstrated that their conclusions are valid for six degrees-of-freedom manipulations as well.

MANIPULATOR TECHNOLOGY

This paper is concerned with man-controlled manipulators as contrasted to programmed robots with artificial intelligence. The manipulator-TV system shown in Figure 2 is not a robot, but is a slave mechanism reproducing a human operator's hand and head motions. Forces and torques applied by the operator to the master arms are also reproduced at the slave, and vice versa. Only the operator's hands are involved. This is done by linking the mechanism's joints to be kinematically determinant within the manipulator's working area. The controls are deceptively simple. Corresponding master and slave joints are independently servoed together so the torque developed by their actuators is proportional to the position error between each master and slave joint. Hence, the linking structure is the only control interaction between the joints of the manipulator arms.

The terms "telefactor" and "teleoperator" have been proposed by Bradley (Reference 4) and Johnson (Reference 5) to describe remotely controlled master/slave mechanisms. Since the manipulator-TV system is Figure 2 has man-equivalent reach and working force, it can be described as an androidal teleoperator, or android for short. Thus, the term robot can be saved for its seemingly inevitable future use.

There are two basic types of manipulators:

- General Purpose Manipulator

  A man-controlled device with seven or more independent motions; one for grasping, three for translation motions, three for angular motions, and redundant motions as required for work site access or dexterity

- Special Purpose Manipulator

  A limited motion manipulator with less than six degrees of freedom at the terminal device

The general purpose manipulator, such as the E-2 type electric manipulator shown in Figure 3, is used for tasks that require adaptability and cannot be reasonably automated, i.e., for tasks requiring human judgment or dexterity. The special purpose manipulator is used for simple, potentially programmable tasks or when the transporter or vehicle carrying the manipulator can provide the missing degrees of freedom.

FIG. 3 - E-2 ELECTRIC MANIPULATORS USED BY GENERAL ELECTRIC IN SPACE SIMULATION STUDIES

There are many varieties of existing manipulators and potentially an infinite variation of new ones. However, they tend to separate into two groups:

a) Bilateral: Bilateral manipulators are reversible. A force or motion applied at the input (master handle) will produce through the control system and mechanisms of the manipulator, a force or motion at the output (slave hands). Similarly, if a force or motion is applied at the output, it will produce a force or motion at the input.

b) Unilateral: Unilateral manipulators are not reversible. Forces and motions at the slave are not transmitted to the master. There is no force or motion feedback between the output and input.

From system weight and reliability considerations, it is concluded that the manipulators for space applications will be limited to manual and/or all electric types for the foreseeable future. An example of a manual through-wall manipulator is shown in Figure 4 being used for docking and tethering experiments. This tape and cable connected master/slave manipulator is the most widely used model for "hot" laboratory work. The bulkhead sealing problem in space vehicle applications

FIG. 4 - M-8 MANUAL MANIPULATORS USED BY GENERAL ELECTRIC FOR DOCKING SIMULATION

may be easier to solve with manual manipulators using ball-joint bulkhead pivots. However, ball-joint type of manipulators have motion reversal and require larger working volume at the master.

Electric master/slave manipulators can be bilateral or unilateral. The bilateral M/S manipulator is the most versatile and best performing of all manipulator systems. The unilateral M/S is next and offers potential weight advantage for space controlled systems and stability advantages for ground controlled systems involving transmission time delay. The unilateral master station requires no force feedback actuators which account for nearly half of the weight of a bilateral M/S system. However, the unilateral M/S system energy consumption will be much higher because (1) force levels are indeterminant, (2) better TV is required and (3) more time is required to complete a task. Control station weight can be even further reduced by using panel control rather than master/slave. On-off or rate controls for each slave motion can be individual or coordinated with a joy-stick. Many tracking type tasks, such as docking, cannot be reliably performed with unilateral panel controls and those tasks that can be performed take typically ten times longer to accomplish. Although their performance is poor, panel controlled systems can have an overall advantage for simple missions. And if communication bandwidth is limited, they may be the only alternative.

The Surveyor Moon-Digger is a panel-controlled special purpose manipulator. Each of its four motions are individually controlled by step commands of 0.1 and 2.0 seconds which result in motion increments of about three inches. The only feedback to the operator are still pictures. These have good resolution, but require nearly one minute to process. Step control is unique to the Moon-Digger which is the first, and thus far only, space manipulator. The typical panel-controlled manipulator has individual rheostats to control the rate of each arm motion. Because their first cost is lower than an equivalent master/slave, they are the most commonly used electric manipulator in "hot" laboratory work. Another example of on-off control is the General Electric manipulators used on the Aluminaut submarine. These heavy duty, hydraulic powered manipulators are controlled by joy-stick actuated switches on the operator's console. The joy-sticks provide directional correspondence and are so arranged that the operator does not have to take his eyes off the slaves to see what he is doing at the console.

The bilateral electric master/slave manipulator clearly has the most significant role in future space manipulator applications. Because of its cost, it is not used extensively in nuclear laboratory work. However, its design technology is highly advanced and those units that have been put in service have demonstrated good reliability and life. All existing electric master/slave manipulators are based on designs developed by the Argonne National Laboratories. A version of the ANL Model E-2 shown in Figure 3 is now being used by General Electric for space task simulation, transmission time delay experiments, and data link development.

The E-2 manipulators have a 7.5 pound capacity and kinematics which appear to be nearly optimum for both manned and unmanned missions.

MANIPULATOR CAPABILITIES

The task performance of remote manipulators in space has to be extrapolated from ground experience. Extensive experience in nuclear "hot" laboratories during the past 20 years has ranged from very simple to very complex tasks. Task time with bilateral master/slave manipulators has been found to range from 3 to 10 times the direct-hand time. The time factor ranges from 30 to 100 with unilateral panel - controlled manipulators. These hindrance factors are for assembly and repair tasks. When used for transport tasks, such as cargo transfer, manipulators can actually reduce by-hand time if they have reach or force capability exceeding man's.

Rules-of-thumb, though convenient, must be used with caution. The wide range given for hindrance factors indicate the importance of the task itself. In this context, simple tasks do not necessarily yield a low hindrance factor (HF) and complex tasks do not necessarily yield a high HF. Much better correlation is obtained by classifying tasks as positioning (wrist and terminal motions) or transport (shoulder and elbow motions). Positioning tasks will tend to have a high HF; transport tasks low. The highest HF will be encountered in tasks requiring dexterity and tactile sensitivity.

Task board work by ANL (Reference 1) has been repeated and confirmed by General Electric as shown in Figure 5. The ANL results were:

Task Procedure	Time (Minutes)	Hindrance Factor
Directly in Shirt Sleeves	7	1.0
Directly in a Pressurized Apollo Soft Suit	20	2.9
With Model-8 Manipulators While in Shirt Sleeves	24	3.4

The same tools were used in all three tests.

FIG. 5 - TASK BOARD TIME AND MOTION STUDIES WITH M-8 MANUAL MANIPULATORS

Most existing terminal devices have only one degree of freedom and therefore cannot change the position of an object in their grasp without assistance from another arm, bench, fixed object, gravity, or inertia. Another degree of freedom in the tongs or a third finger would be required to increase the negligible dexterity in existing terminal devices. Also, tactile information is useful to detect slippage of objects in the manipulator's grasp. However, better dexterity and tactile feedback does not appear to be necessary, or even desirable, if the tools and the work are designed for manipulator operation.

COMPARISON OF POTENTIAL SPACE MANIPULATOR CONCEPTS

It was concluded from an analysis of space task work elements (Reference 2) that the maximum force capacity required for space manipulators will be about 15 pounds. This is about the normal maximum force utilized by machinists and aircraft assembly workers. When higher forces are required, a lever or powered tool is normally employed. Astronauts and androids will follow the same practice.

The ideal work volume* of a standing operator is about 20 cubic feet. With body motions and awkward overhead positions, the working volume of a one to one master/slave manipulator is extended to over 60 cubic feet. This will be permissible for space manipulators where the master station is on the ground. But if the master station is in a space vehicle, volume will be too precious and it will be necessary to utilize index motions to extend the reach of the slave. Argonne National Laboratories suggest that the minimum effective working volume of the master is about four cubic feet. If this is incompatible with the space vehicle, then a unilateral, panel controlled manipulator must be used. These require about one cubic foot for the controls and the operator's hands. They are also much lighter than bilateral masters but, as discussed later, they require as much as ten times the time and energy to perform a task and cannot perform many tracking and "blind" tasks.

*All work volumes refer to two arm volumes.

The optimum terminal velocity of master/slave manipulators, at no load, has been determined by ANL to be about 30 inches per second. The panel controlled manipulators have an optimum velocity of only four inches per second according to Seidenstein (Reference 6).

Using the above force, reach and response data, design estimates were made to compare the five classes of manual and electric space manipulators in Table 1. The lightest weight system is the manual through-wall manipulator which utilizes a ball-joint for a bulkhead pivot. This system is probably not acceptable for most manned missions because it is questionable that indexing can be used to reduce master working volume. The lightest weight electric manipulator is panel-controlled, but as already pointed out, it is task limited and very slow.

Elimination of force feedback motors in the master arms reduces the master weight of the unilateral M/S system by almost 100 pounds as compared to the bilateral M/S system. However, the weight advantage of the unilateral M/S system is countered by the lack of visual as well as force feedback when the slave is attached to a fixed object. In this situation, the force level and power consumption can range from zero to full stall without indication.

The heaviest, but most versatile, manipulator is the electric bilateral master/slave.

Table 1

COMPARISON OF MANIPULATORS

ORDER OF MAGNITUDE ESTIMATES FOR TWO ARMS OF MAN-RATED SPACE DESIGN

		MANUAL		ELECTRICAL			REMARKS
		Master-slave	Ball-joint	Panel-control	Unilateral M/S	Bilateral M/S	
Force Capacity, lbs	(a)	15	15	15	15	15	(a) In any direction at about 2 ft reach of hands
No Load Velocity, in/sec	(a)	Man's	Man's	4	30	30	
Master Working Volume, ft^3	(b)	4	60	1	4	4	(b) Minimum for two hands (20 ft^3 ideal)
Index Motions Possible	(c)	XYZ	XY	NA	XYZ	XYZ	(c) Required to extend working volume of slave
Slave Working Vol. with Index, ft^3	(d)	200	60	200	200	200	(d) 1:1 Correspondence with Master at given index position
Master Weight per Pair, lbs		-	-	2	5	100	(e) Includes servo amplifiers
Slave Weight per Pair, lbs		-	-	78	100	100	Excludes data link, power supply and heat rejection system
Total Weight, lbs	(e)	150	90	80	105	200	
Peak Power, M/S Watts		40 (Index)	0	Nil/40 (f)	Nil/800+(g)(h)	800/800 (g)	(f) On-off control, one motion at a time (g) Power Input Amplifiers (switching) (h) Power and forces indeterminant (without feedback) when unilateral M/S is "grounded"
Avg. Power, M/S Watts	(i)	Nil (Index)	0	Nil/20	Nil/40+	40/40	
Task Time Relative to Manual M/S	(i)	1	1.5	10	3	1	(i) Avg. Power Estimated from Base-line Mission

SPACE SYSTEMS

Based on the excellent performance record of manipulator systems for earth application, such as handling radioactive elements in hot labs, industrial mass transfer, and prosthetics, the possibility of extending their application to space missions looks quite promising.

Several concepts for space manipulator systems have been investigated. These include:

- Space Vehicle with Through-wall Manipulators
- Space Station with Boom Positioned Manipulator
- Remote Maneuvering Unit with Manipulator

Space Vehicle with Through-wall Manipulators

Through-wall manipulators have been proposed for extravehicular tasks from manned vehicles since the late 1950's. A survey of these concepts was published in 1962 by the Air Force Aeromedical Research Laboratories (Reference 8). The first comprehensive vehicle and mission study involving through-wall manipulators was made by Lockheed-California for the Air Force Aero Propulsion Laboratory in 1964 (Reference 9). The terms "Astrotug" and "Shuttle" were used in this study. In an independent study, Lockheed Missiles and Space Company coined the acronym SCHMOO for Space Cargo Handler and Manipulator for Orbital Operations (Reference 10). The term "Space Taxi" was used for a one-man vehicle concept studied by Ling-Tempco Vought (LTV) for NASA's Advanced Systems Office at the George C. Marshall Space Flight Center (Reference 7). Manipulator design studies for the "Space Taxi" were conducted for LTV by the Argonne National Laboratories (ANL), who in 1967 published a consultant support study "Manipulator Systems for Space Applications" for NASA (Reference 1).

The mechanically connected or manual manipulator offers the lightest weight and highest reliability for through-wall applications. However, it requires large working volume at the master, and the slave must be located in close proximity just outside the operator's compartment. Electrically connected and powered manipulators overcome these geometric restrictions and offer extended reach and force capability at the slave. Extended slave reach is obtained in electric master/slave manipulators by biasing (i.e., indexing) selected M/S motions or by adding index motions. By this approach, slave working volumes of over 200 cubic feet can be covered with only four cubic feet of working volume at the master. The weight and power consumption of electric M/S manipulators are considerably greater than for an equivalent manual system. Therefore, each mission must be analyzed to determine which system is optimum. Argonne National Laboratories estimated the weight for 25 pound force capacity manipulators with 96 inch reach for LTV's "Space Taxi." They concluded that two manual arms would weigh about 150 pounds whereas the equivalent electrical system would weight 535 pounds exclusive of power supply, cabling and heat rejection subsystems.

An equivalent switch controlled (unilateral) electric manipulator system would weigh only half as much as the bilateral master/slave. As already discussed, however, the task performance is so poor that it is not recommended for the Space Taxi-type of application.

Space Station with Boom Positioned Manipulators

In this concept, an articulated boom serves as a locator for the terminal device. The terminal device contains the two working manipulators, video, docking arms, decoder, processor, harnessing, and structure. Control for the boom and terminal device is from within the space station. Power and control would be via cables through the boom. Several boom concepts appear feasible for this application. One is the "Serpentuator" concept being developed by H. Wuenscher of the NASA Marshall Space Flight Center. Another is a three inch deployable boom being developed by the Space Systems Organization of the General Electric Company.

A summary of the system weight and power requirements are tabulated in Table 2.

Remote Maneuvering Units with Manipulators

This concept is also illustrated in Figure 2. It could be used to augment an astronaut's capabilities through control from within the space station. It could also be used in unmanned operations through ground control via a communications link.

The mission capability of the RMU-with-manipulator concept includes all those of the boom version plus operations on nearby satellites either already orbited or launched from the space station. These operations would include rendezvous, docking and stabilization of the refueling and component replacement.

Table 2

BOOM POSITIONED MANIPULATOR
SYSTEM WEIGHT AND POWER SUMMARY

Subsystem	Power Watts	Weight Lbs
Video	17.0	26.5
Manipulators (two)	40.0	46.0
Manipulator Amplifiers (switching type)	---	35.0
Docking Tethers (three)	nil	50.0
Communications	12.0	14.0
Harnessing	---	16.0
Structure	---	32.0
TOTAL	78.5	210.0

The advantages of this concept are that its sphere of operations and mission capabilities are much greater than the boom version. The system contains the following subsystems: two bilateral electric manipulators with passive docking tethers; head directed TV; propulsion to perform rendezvous and docking; attitude stabilization powerful enough to stabilize the RMU plus the satellite being worked on; communications relay; power; and thermal control.

Typical weights and power requirements for an RMU-with-manipulator system are shown in Table 3.

SPACE MANIPULATOR DESIGN

The five types of manipulators that are likely candidates for earth orbit space missions are:

- Manual Master/Slave
- Manual Ball-joint
- Electric Panel-controlled
- Electric Unilateral Master/Slave
- Electric Bilateral Master/Slave

None of these have been developed for earth orbit missions, although the panel-controlled Surveyor Moon-Digger has demonstrated the space worthiness of articulated mechanisms. All of these are presently used in nuclear and underwater applications and must meet stringent reliability and maintainability requirements.

The kinematic and human factor design requirements for all five types are well known and are directly applicable to space designs. In existing manual and electric master/slave systems, an overhead or inverted elbow configuration is used to minimize physical interference with the operator. This configuration also gives good access to restricted work sites (e.g., through hatches and in recesses) because the manipulator is in the cone of vision. The kinematic relationship

Table 3

RMU WITH MANIPULATOR SYSTEM
WEIGHT AND POWER SUMMARY

Subsystem	Power Watts	Weight Lbs
Video	33.0	52.0
Propulsion	25.0	22.0
Guidance and Control	83.9	56.8
Communications	21.8	28.9
Power	- - -	165.0
Thermal Control	- - -	5.0
Manipulators (two)	40.0	81.0
Docking Tethers (three)	nil	50.0
Structure	- - -	95.0
Harnessing	- - -	45.0
TOTAL	203.7	600.7

of links and pivots has evolved with experience to minimize parallel joint lock-up (gimbal lock) within the normal working positions. Force and torque capacity of existing master/slave units are usually man-equivalent, although some manual units are designed for heavy duty work where the operator uses two hands on one manipulator (usually because he has forgotten to bring in the proper tools).

Most manual and electric master/slave manipulators are of the "tendon" design wherein cables or tapes are used to transmit most of the motions. A variety of tape and cable anchor and pulley guide designs have been developed over the years. Cammed pivots have also been developed to maintain constant cable length and tension over wide ranges of pivot actions.

Because inertia and friction reduce manipulator effectiveness and cause operator fatigue, existing designs utilize aerospace design techniques to reduce weight and increase efficiency. As a result, much of the design practice for ground systems will be applicable to space. The most significant weight saving seen for space systems is the elimination of counter weight for weightless earth-orbit missions.

Preliminary Design Study Specifications

The USAF/GE study (Reference 2) concluded that a single general purpose electric manipulator can be developed to meet the requirements for most anticipated manned and unmanned missions. Tentative functional and environmental specifications for missions up to the stationary synchronous orbit have been established. These specifications are based on the following rationale:

For Manned Missions:

- The reach (without indexing) should be no greater than the reach of an EV astronaut. This is to insure that tasks are designed so that they can be accomplished by the EV astronaut in the event of manipulator failure.
- The force capability should be adequate to operate astronaut "hand" tools but should not exceed that of an EV astronaut. This force capability is estimated to be 15 pounds.
- Design should be for bilateral control with provision for unilateral position control and unilateral rate control.
- Response should be high enough to make the operator the limiting factor in determining task speed. This means no load velocities of 30 inches per second for bilateral control. Optimum unilateral (on-off) no load velocity is about four inches per second.
- Anthropometric relationship between "hands" and TV "eye" should be provided to utilize operator's natural and learned responses -- particularly his proprioception.
- Tools and procedures will be provided whenever above capabilities must be augmented.
- Control should be possible from both spacecraft and ground master stations.

For Unmanned Missions:

- Reach, force and response should be no greater than the operator's. Tools and procedures will be provided when these capabilities must be augmented.
- Manipulator system should be as small and light weight as possible to be compatible with low cost boosters and to simplify integration with transport vehicles.
- Generally, task speed can be sacrificed in unmanned missions to reduce system weight and complexity, particularly where solar recharging is practical.
- Interchangeability with manned systems should reduce development costs and improve emergency or backup position for user programs.

In summary, the universal general purpose manipulator system is essentially equivalent to an EV astronaut. It would utilize the same tools used by an astronaut and would follow the same work procedures with the exception of those dictated by astronaut endurance or exposure limitations.

A preliminary design study of manipulators for an unmanned ground controlled system was made as part of the USAF/GE study. This application was selected because it encompassed all conceivable hardware and operational problems for either manned or unmanned missions. The mission selected was the refurbishment of a communication satellite in synchronous orbit. The principle requirements established for this mission are:

Base-line Mission:

1. Dock and tether slave vehicle to cargo craft or refurbishment kit
2. Transfer and restrain refurbishment cargo to slave vehicle
3. Dock and tether slave vehicle to communication satellite
4. Open hatch and make visual and electrical inspection (electrical connector)
5. Replace electronic module:
    a. Six captured screws
    b. Two electrical connectors (pigtail)
6. Check out and release communication satellite

Duty Cycle:

1. The system should be capable of repeating this operation at least ten times over a two year period
2. In the M/S mode, each actuator must be capable of holding maximum forces for at least 30 seconds (after thermal equilibrium) without exceeding a motor rotor temperature of 220° F

Data Link:

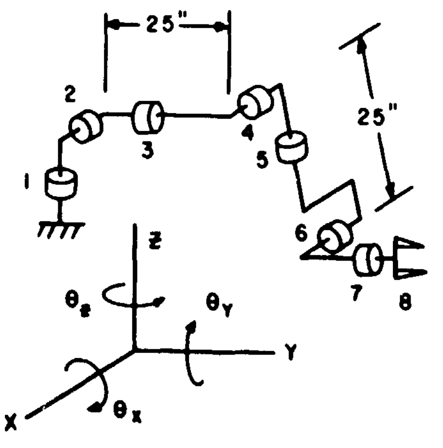
1. Transmission time delay will range from 0.24 to 1.0 seconds, depending on location of synchronous slave vehicle to ground master station. When not in line-of-sight, communi-

cation will be via one or two relay satellites.

Results of Preliminary Design Study

Figure 2 shows the arm configuration developed in the design study. The slave kinematics are shown in Figure 6. Master kinematics are identical with the exception of the first shoulder index joint which is used to extend the reach of the slave or to avoid obstacles. The second shoulder joint has an index bias at the slave for the same purposes. The shoulder joints are principally for transport motions; wrist joint for positioning motions. In Figure 2 an additional torso index motion is provided to allow the android to work on equipment in the RMU.

The main departure from convention in this design is the use of servo-actuators at the pivots rather than in the torso with tendon connections to the pivots. The pivot actuator approach provides significant weight and reliability advantages because of its simplicity. The approach is practical only because it was determined that the specified duty cycle could be met with passive heat rejection from the actuators without exceeding the temperature limits of the motors. If a circulatory coolant was required, the "tendon" design would simplify actuator cooling since the actuators could be grouped together on a common heat sink in the torso. The temperature distribution in a stalled shoulder joint actuator is shown in Figure 7. These temperatures were reached after more than one minute of stall, starting from a 32° F equilibrium condition. The calculations used the emissivity and absorptivity values (after over two years in orbit) of the silicone acrylic paint used on the Geos satellite. No-load equilibrium temperatures are 32° F in the sun, 172° F between sun and large reflecting surface, and -400° F in earth shadow. This latter temperature is, of course, intolerable. However, it can be kept to a reasonable -65° F by an isometric exercise producing 18 watts in the joint. For long periods in shadow,



Actuator No.	Function M/S	Function Index	Principal M/S motion
1 - Shoulder		x	-
2 - Shoulder	x	x	Z
3 - Shoulder	x		X
4 - Elbow	x		Y
5 - Wrist	x		θ_z
6 - Wrist	x		θ_x
7 - Wrist	x		θ_y
8 - Terminal	x		Terminal

M/S = Master/Slave Servoactuator

FIG. 6 - KINEMATICS FOR SPACE MANIPULATOR

it will probably be necessary to enclose the android.

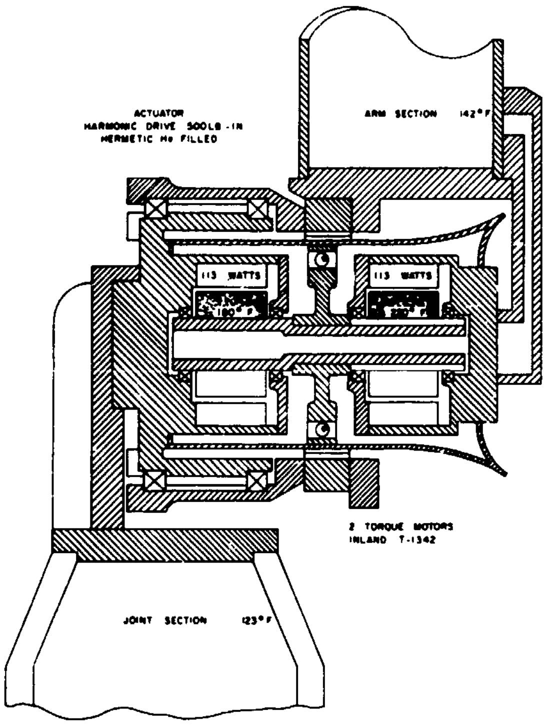


FIG. 7 - HARMONIC DRIVE ACTUATOR TEMPERATURE DISTRIBUTION

Another departure from existing M/S manipulator design practice is the use of DC servomotors. AC motors are used in the master and slaves of present M/S manipulators and they are driven by a common amplifier. Radio controlled manipulators will, of course, require separate master and slave power supplies and servoamplifiers. Position error is commonly used to provide the force feedback. However, the higher friction of harmonic drives (than spur gear drives) may dictate that slave motor current or actual output force be sensed and fed back for replication at the master. The actual selection and implementation of the specific bilateral control system is not critical and will have little effect on the total weight and power requirements of the slave manipulator tabulated in Table 4 below:

Table 4

WEIGHT, VOLUME, AND POWER OF ONE MANIPULATOR ARM FOR BASE-LINE MISSION

Weight	
Arm	23.0 lbs
Amplifier	17.5 lbs
Total	40.5 lbs
Volume	
Arm (stowed)	1680 in.^{3}
Amplifier	460 in.^{3}
Total	2140 in.^{3}
Power	
All Joints Stalled (unlikely)	1330 watts
Expected Peak (<0.1% of time)	400 watts
Average	20 watts

CONCLUSIONS

Existing manipulator concepts can meet most foreseeable space mission requirements. The major technology problems in the space application of manipulators are in the control link. On-off controlled manipulators have been radio controlled, but thus far, this has not been tried with master/slave manipulators. Existing telemetry equipment is more than adequate on a bandwidth basis for both command

and feedback signals for two or more arms. However, line-of-sight limitations and multipath interference problems have not yet been defined.

Transmission time delay will not effect the data link, but will have an effect on stability and performance of bilateral manipulator systems. For distances from 1000 to 20,000 miles, transmission time delay is primarily a servo stability problem. However, for greater distances, it becomes a human factor problem as well. Additional research and development is required to learn how to stabilize ground controlled bilateral manipulators in orbit without degrading their performance.

DC servos offer lighter weight and better efficiency than the AC servos used in most master/slave manipulators to date. The DC servo also avoids synchronization and phasing problems that are inherent with transmission time delay. There appears to be an adequate technology base to develop space worthy DC servo actuators and servoamplifiers. However, a development program is indicated from both a systems and environmental standpoint to fully assess the problems.

More ground and space task time and motion and energy data are required for mission and system planning purposes. Useful base-line data can be obtained with manual through-wall manipulators; however, energy and time delay effects data can be obtained most readily with electric master/slave manipulators. These data are particularly important for cost-effectiveness trade-off studies of missions where there are alternatives to the manipulator approach.

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QUICK-RELEASE FASTENERS FOR SPACE APPLICATIONS

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SUMMARY: Human performance requirements for eleven hand-operated fasteners were determined experimentally. Two situations were investigated: (1) simulated zero-gravity and (2) normal gravity conditions. A six degree-of-freedom air-bearing device was used for zero-gravity simulation. Data analyses showed effects of fastener design and gravity condition to be significant in terms of five dependent measures. There was also a significant interaction between the fastener design and gravity condition effects.

INTRODUCTION

Accessibility is essential to maintainability. If you can't get to it, you can't fix it. It's almost as simple as that. The matter is complicated somewhat by the fact that certain individuals are less apt to acquiesce to "environmental barriers" than others. Among the more extreme examples is that of the anonymous technician who removed and replaced B-52 gunsight components without "noting" the damage to metal webbing which he deliberately smashed to gain the required access.[3] Less determined individuals, on the other hand, tend to ignore or avoid maintenance tasks which present formidable access problems. Believing that the more effective solution to such problems is usually found in improved design rather than modified personnel selection, incentive, or training programs, we have given our attention, in this instance, to alternative design configurations for quick-release, hand-operated access panel fasteners for space system applications.

Hand-operated, quick-release fasteners for access panels are preferred to other types under normal maintenance conditions. The advantages they afford should be even more important in the extra-terrestrial environment. For example, hand-operated fasteners require no tools, so the astronaut's tool kit may be reduced in size and weight. Quick-release fasteners also simplify the task—even simple tools are difficult to grasp and manipulate under pressurized suit conditions. Reduced fastener operation times also would help minimize exposure of the astronaut to extra-vehicular risks.

Because the advantages are

appa.ent, this investigation was not designed to prove the desirability of quick-opening, hand-operated fasteners for use in space. Instead, the primary objective was to develop and apply engineering and human performance criteria for selecting from a variety of designs for such fasteners. Independent variables of primary interest were: (1) fastener design and (2) gravity condition (simulated zero-gravity vs. one gravity). The principal dependent variables were: (1) time required to perform tasks involving different fastener designs, (2) forces and torques exerted during these tasks, and (3) user opinions with respect to ease of fastener operation.

APPARATUS

The apparatus shown in Figure 1 was used to simulate maintenance requirements under zero-gravity conditions. The support cradle into which the subject is secured can be adjusted to accommodate a relatively wide range of anthropometric dimensions. It also affords six degrees of freedom in motion under near-frictionless conditions. Gimbals rotating on ball bearings provide rotational movement in the roll and pitch axes. Translation in the yaw axis is facilitated by air bearings. The static balance required for the vertical axis is maintained by means of a piston and cylinder mechanism supplied with appropriately regulated air pressure. Free movement in the horizontal plane is provided for by the air bearing pads at the base of the tripod supporting the simulator. When supplied with air pressure, these pads ride approximately 0.002 inches above the epoxy floor. The floor is level to within 0.0005 inches. Floor dimensions are 26 x 30 feet.

The work panel, in front of the subject in Figure 1, is suspended from a 20-foot span of bridge crane. The work panel assembly is instrumented to permit simulation of the reactions of a free-floating mass to forces applied to it; however, the assembly was maintained in a stable position throughout this investigation. Stabilization of the work panel was based on the assumption that the mass of the space vehicle involved would be large relative to that of man; hence, reactions of the vehicle itself to forces applied by the worker during the task would not be great enough to significantly affect his performance. (More detailed descriptions of the simulation facility are included in references 1 and 2.)

The work panel structure was equipped with 6 strain gages which sensed forces and torques exerted against it in the 3 principal axes (x, y, and z). Strain gage displacements were transformed into electrical signals by means of bridge circuits. Outputs of the bridge circuits were transmitted to an 8-channel Beckman strip chart recorder so that the forces and torques in the 3 principal axes were recorded independently and simultaneously.

The bridge balancing network was mounted at the rear of the work panel structure. The force and torque measurement circuitry was calibrated periodically by imposing known forces and torques to the work panel. During the initial phases of the experiment these forces and torques were mea-.red by means of a linear coil spring mechanism. Later, the

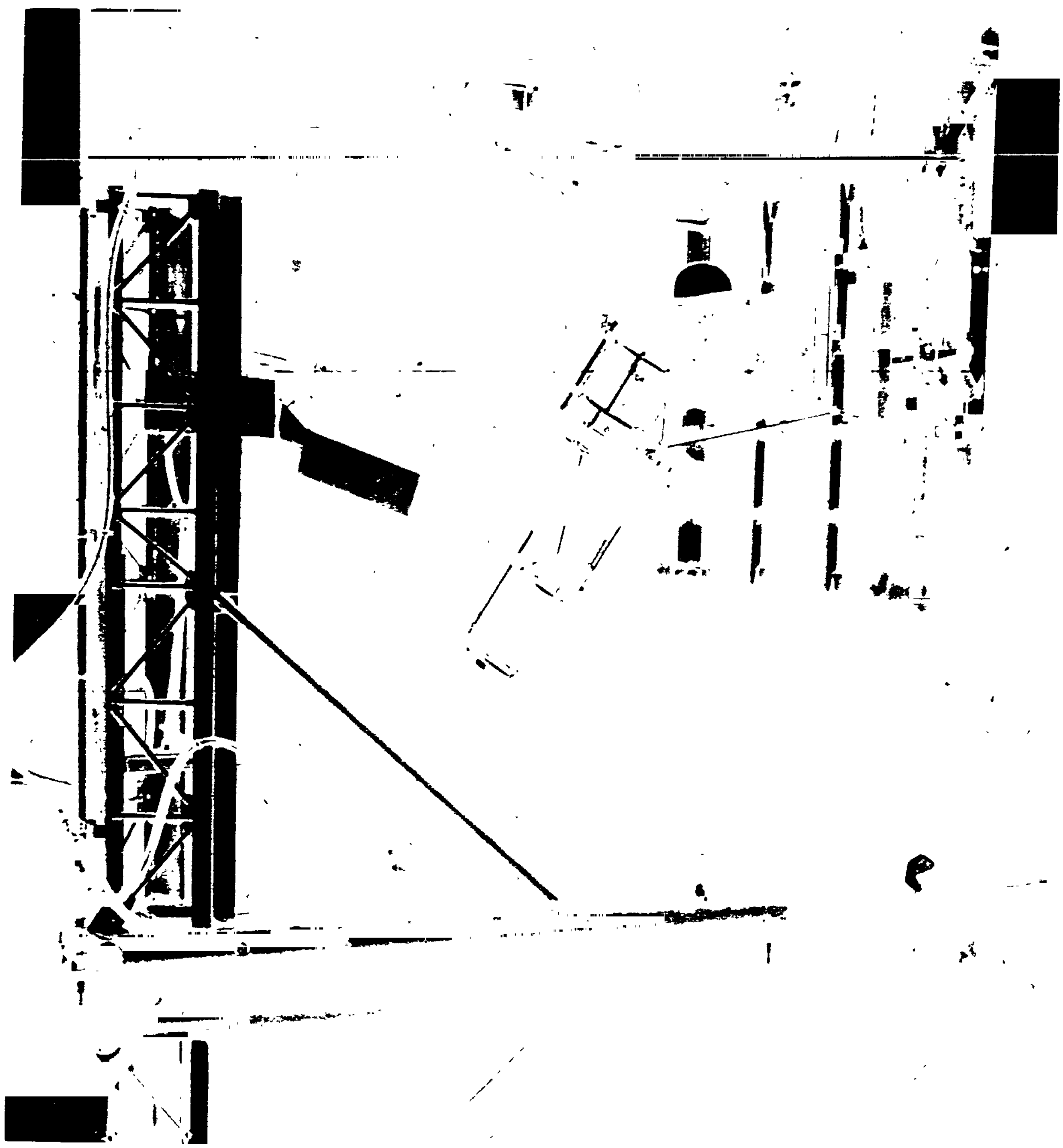

FIGURE 1

ZERO GRAVITY SIMULATION FACILITY INCLUDING WORK PANEL calibration technique was improved upon by substituting dead weights and pulleys for the spring mechanism.

Although the subject's body and legs were secured to the simulator support cradle, he was free to move his head and work with his hands and arms with relatively unrestricted freedom. However, two 28-inch tethers kept the subject from drifting beyond reach of the work panel while performing his task. The tethers were attached to a safety belt at the subject's waist and snap-hooked to eyelets on the work panel superstructure. Anchor points were separated by approximately 12 inches at the subject's waist and 24 inches at the work panel. The tethering provisions were similar to those used by Seeman et al in studying space maintenance problems.[4]

Eleven different fasteners were selected for study in the following manner. Approximately 100 manufacturers of hand-operated fastening mechanisms were surveyed to obtain samples of applicable, off-the-shelf test items. Subsequently, 13 manufacturers submitted a total of 21 items for evaluation. Ten of the 21 fasteners were eliminated prior to the collection of experimental data. Some were eliminated because they were essentially duplicates of others. Others were rejected because of repeated mechanical failures during preliminary tests. Still others were not subjected to experimental evaluation because they could not be mounted in a manner that would permit a valid comparison of performance data.

Figure 2 shows the disassembled components of each of the 11 fasteners for which experimental data were obtained. Operational requirements associated with each fastener are summarized briefly in Table 1. Although complete detailed specifications were available in most cases, they will not be cited here. In general, the specifications complied with standard military requirements.

The work panel includes a 10-inch x 10-inch access opening equipped with a 12-inch x 12-inch cover panel. The cover panel was secured in place by two identical hand-operated fasteners. The two fasteners were separated by 11 inches in all but two cases. Fasteners 2E and 2A were 12-3/8 inches and 13 inches apart, respectively. Separate cover panels were provided for each of the 11 different fastener types investigated.

SUBJECTS AND PROCEDURE

Ten paid volunteers from a local university served as subjects for the investigation. They were all males and averaged 21 years of age.

Each subject was required to operate all 11 fasteners in performing a representative access task under two conditions: (1) "simulated zero gravity" and (2) "one gravity". The one-gravity situation, alternatively referred to as the "static condition," served as an experimental control with which performance in the "simulated zero gravity" condition could be compared.

One half (5) of the subjects began testing with the zero-gravity condition and the other half began with the static

TABLE 1

SUMMARY OF OPERATIONAL REQUIREMENTS OF HAND-OPERATED FASTENERS

FASTENER	OPERATIONAL REQUIREMENTS
1B	Push in wing head to overcome spring tension and turn cw $\frac{1}{4}$ to lock. Turn $\frac{1}{4}$ ccw to unlock.
1F	Turn wing head $\frac{1}{2}$ cw to lock. No detent. Turn $\frac{1}{2}$ cw to unlock.
2A	Adjustable pawl positioned by lever. Lever is cammed so that depression brings pawl in contact with back edge of access opening.
2C	Cam lock. Turn wing head $\frac{1}{4}$ cw to lock. Turn $\frac{1}{4}$ ccw to unlock.
2D	Captive screw with pointed tip inserted into tapped hole. $4\frac{1}{2}$ turns required to fasten as well as unfasten.
2E	Adjustable pawl, positioned by pawl positioning key. Knob adjusts pawl for various panel thicknesses and compresses pawl to back edge of access opening.
2H	Push button releases friction lock.
4B	Operationally the same as 2C.
5B	Operationally the same as 2C.
6A	Captive screw with blunt end inserted into tapped hole. 3 turns required to fasten as well as unfasten.
7A	Snap-slide type. Slide thrown outboard and snaps under stud.

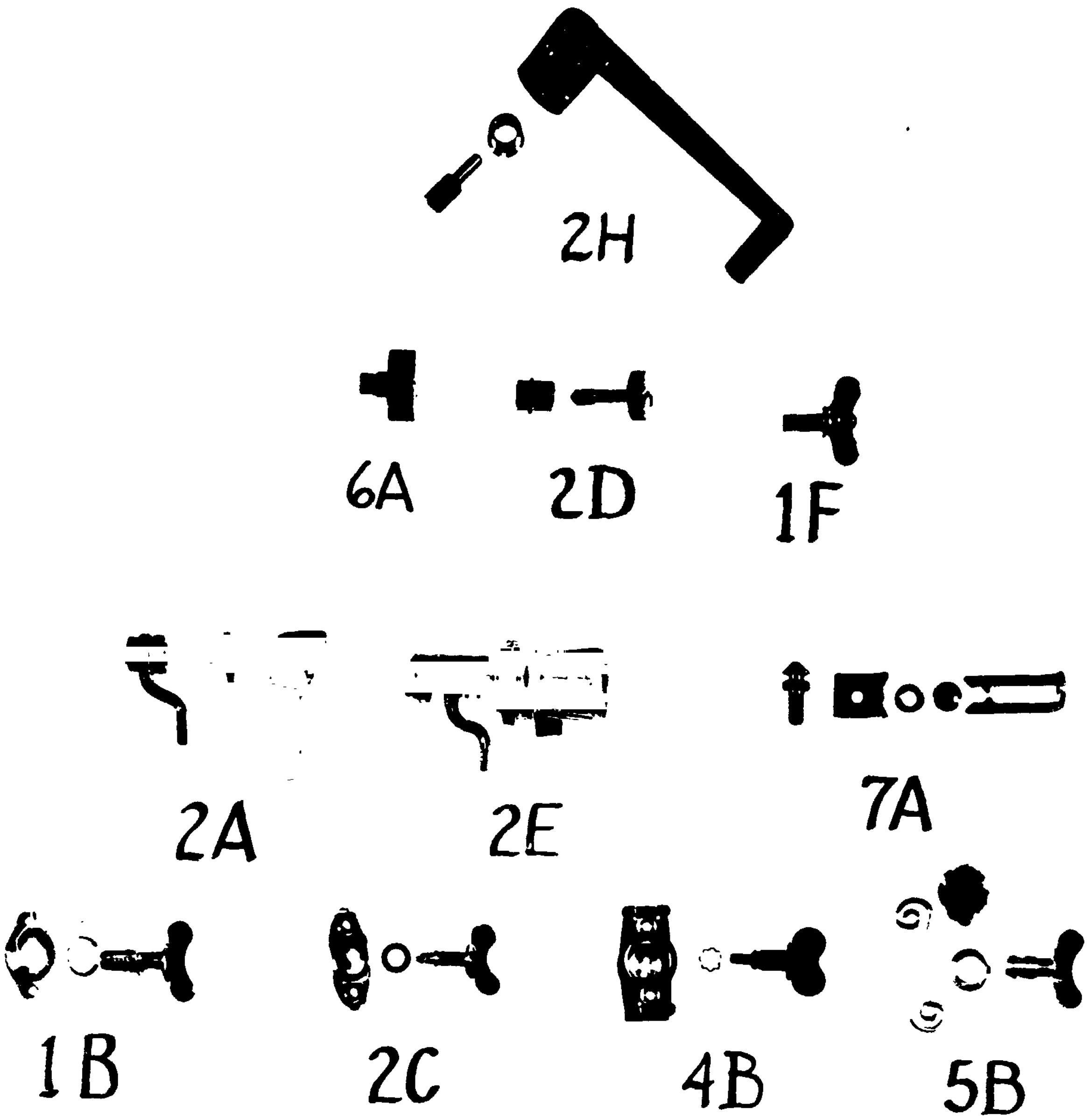

FIGURE 2

HAND-OPERATED FASTENERS USED IN THIS STUDY condition. These two sequences of environmental conditions will be referred to hereafter as the "simulator-static" and "static-simulator" sequences, respectively.

Tests conducted under the first condition of each sequence were separated from those of the second condition by an interval of approximately 3 days. The 3-day interval was necessary because subjects were available only for a limited period of time each day, e.g., one or two hours, which was not sufficient for completion of the entire experiment. The delay also assured each subject adequate rest time between test conditions.

The task used in this study involved installation and removal of the access cover panel. Under test conditions, each subject performed the complete task three consecutive times with each fastener type. The order in which the fastener types were presented was randomized for each subject.

Prior to testing under the initial condition, each subject was instructed as to the purpose of the experiment and provided the opportunity to become thoroughly familiar with the design and operation of each fastener. All the cover panels were laid out before the subject so that he could operate each fastener as many times as he desired. This was done to minimize the progressive effects of learning during the test phase of the experiment. Testing did not begin until the subject indicated that he understood the operational characteristics of each fastener.

Prior to testing under simulated zero-gravity, the subject was fitted into the support cradle and familiarized with the functional characteristics and purpose of the simulator. This required about 15 minutes. During this familiarization period, the subject gained experience with respect to the instability of the simulator primarily through two types of activity: (1) pushing and pulling with his hands against the work panel and (2) shifting his center-of-gravity through arm and hand movements.

For the initial condition only (whether static or simulator), there was additional practice in fastener operation just prior to the first recorded trial for each fastener. At this time, the subject was asked to practice removing and installing the panel until he was sure that he could operate the fastener properly. On the average, this practice constituted 2 to 3 complete remove-install operations per fastener.

Times were recorded separately for the "install" and "remove" phases of each trial and later combined for purposes of analysis.

In the standard starting position before each trial, the subject held the panel cover, by means of the attached fasteners, just in front of the access opening. When the subject signaled that he was ready to begin, the experimenter started a stop watch. The watch was stopped by the experimenter when the panel was in place with both fasteners secured. Before removing the panel, the subject held his hands just in front of the fasteners, indicated that he was ready, and began the removal operation. The experimenter started the watch again at the subject's ready signal and stopped it when the cover had been removed.

After each subject had completed all the trials under both experimental conditions, he was asked to rank the fasteners from 1 to 11 with respect to ease of operation.

RESULTS

Analyses of variance were applied to the human performance data to identify significant effects. For analysis purposes, the fastener design and gravity conditions were treated as fixed factors since there was no intent in either case to generalize beyond those levels for which data were obtained. Subjects, on the other hand, were treated as a random variable to permit generalization on the basis of the data sample. Preliminary tests showed no systematic trials effect or gravity condition sequence (i.e., "simulator-static" vs. static-simulator") effect; hence, procedural measures taken to minimize practice effects were regarded as effective.

Task Times

The analysis of variance summary for task times is shown in Table 2. Note that all effects except the fastener X gravity condition interaction were statistically significant. Figure 3 depicts graphically the effects of fastener design and gravity condition in terms of operation time.

Force and Torque Measures

Because the effort required for a more rigorous treatment of the force and torque data seemed unwarranted, these measures were combined to derive estimates of related physical quantities in the following manner. First, the peak (maximum) force (F) and torque (T) exerted during each trial were determined for each axis. Estimates of "resultant force" (F_r) and "resultant torque" (T_r) were then computed as follows:

$$F_r = \sqrt{F_x^2 + F_y^2 + F_z^2} \qquad (1)$$

$$T_r = \sqrt{T_x^2 + T_y^2 + T_z^2} \qquad (2)$$

In turn, the resultant force and torque estimates for each trial were multiplied by the associated trial time (t), in seconds. These two products were termed, respectively, "resultant momentum" (M_r) and "resultant angular momentum" (M_{ra}). Thus:

$$M_r = (F_r)(t) \qquad (3)$$

$$M_{ra} = (T_r)(t) \qquad (4)$$

Despite physical "impurities" the two momentum values seemed to be reasonable indexes of the amount of effort associated with fastener operation in this investigation.

The summary analysis of variance for the resultant momentum data is shown in Table 3. Results of the other three analyses involving force or torque measures were almost identical in terms of significant effects.

Figures 4 through 7 show the effects of gravity condition and fastener design for dependent measures involving force or torque, i.e., F_r, T_r, M_r, and M_{ra}.

TABLE 2

SUMMARY ANALYSIS OF VARIANCE FOR FASTENER OPERATION (INSTALL-REMOVE) TIME DATA

Source of Variation	Degrees of Freedom	Sums of Squares	Mean Squares	Significance F(df)	p <
Subjects (Ss)	9	4298.14	477.57	10.85(9,440)	0.01
Gravity Condition (A)	1	1262.43	1262.43	13.66(1,9)	0.01
Fasteners (B)	10	21630.57	2163.06	21.03(10,90)	0.01
Ss x A	9	831.94	92.44	2.10(9,440)	0.05
Ss x B	90	9256.01	102.84	2.33(90,440)	0.01
A x B	10	886.50	88.65	1.10(10,90)	N.S.*
Ss x A x B	90	7195.67	79.95	1.82(90,440)	0.01
Within Replicates (Trials)	440	19349.98	43.98		
TOTAL:	659	64711.24			

*Not Significant

TABLE 3

SUMMARY ANALYSIS OF VARIANCE FOR RESULTANT MOMENTUM DATA (DIVIDED BY 1000)

Source of Variation	Degrees of Freedom	Sums of Squares	Mean Squares	Significance F(df)	p <
Subjects (Ss)	9	4.66	0.52	4.63(9,440)	0.01
Gravity Condition (A)	1	3.59	3.59	13.82(1,9)	0.01
Fasteners (B)	10	10.14	1.01	5.86(10,90)	0.01
Ss x A	9	2.34	0.26	2.33(9,440)	0.05
Ss x B	90	15.57	0.17	1.55(90,440)	0.01
A x B	10	3.23	0.32	2.28(10,90)	0.05
Ss x A x B	90	12.71	0.14	1.26(90,440)	N.S.*
Within Replicates (Trials)	440	49.71	0.11		
TOTAL:	659	101.41			

*Not Significant

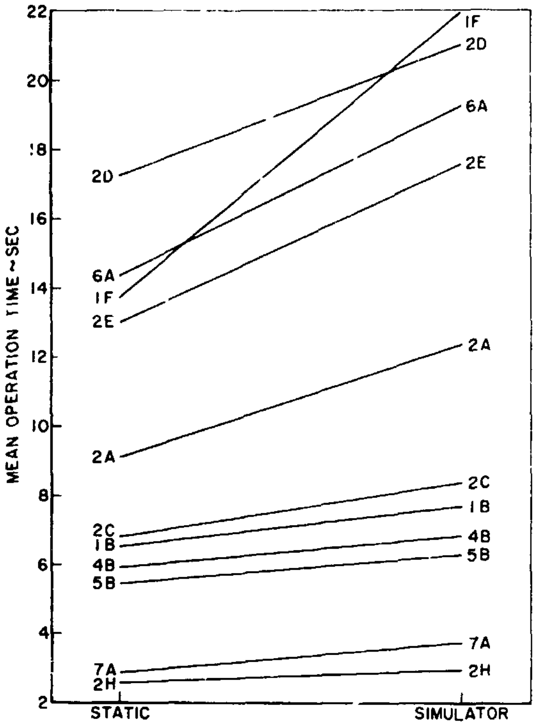

Figure 3. Effects of Fastener Type and Gravity Condition on Operation Time.

Subjective Estimates

Mean ranks, based on subjects' judgments of ease of fastener operation, are shown in Table 4.

Fastener	Mean Rank
2H (Push button with handle)	1.3
7A (Snap slide)	1.7
2A (Pawl with lever handle)	5.1
5B (Winghead cam lock)	5.8
2E (Pawl with knob)	5.8
4B (Winghead cam lock)	6.2
1B (Winghead cam lock)	6.7
2C (Winghead cam lock)	7.2
6A (Captive screw)	8.0
2D (Captive screw)	8.3
1F (Winghead cam lock)	9.9

Table 4. Mean Ranks Assigned by Subjects to Indicate Relative Ease of Use for Each Fastener.

Lower ranks indicate greater ease of operation. A Spearman rank-difference correlation coefficient of 0.75 ($p<.01$) was obtained between the subjects' judgments and operation time averaged across both gravity conditions. Spearman rank-difference correlation coefficients were also computed between the subjects' rankings of the fasteners and each of the four indexes involving force and torque measures. Coefficients obtained for resultant force and torque were not statistically significant. They were -0.38 and -0.18 for force and torque, respectively. For resultant momentum and resultant angular momentum, the coefficients were 0.63 ($p<.05$) and 0.69 ($p<.05$), respectively. A complete set of intercorrelations between the various dependent measures is shown in Table 5.

Experimenter Observations

The experimenters also made certain specific observations during the experiment which are not directly reflected in the recorded data. A compilation of these observations, by category, is shown in Table 6. The fact that fastener 2H failed twice may be of particular interest since it tends to be one of the better fasteners in terms of other criteria. Both failures were due to shearing of the stud. Fastener 2H was also subject to mechanical bind problems.

Note that operation of the pawl-type (2A and 2E) fasteners frequently involved alignment and positioning problems. Fastener 2E also produced more "false positive reports" than any other fastener. The term "false positive reports" refers to instances when the subject indicated that installation had been effected

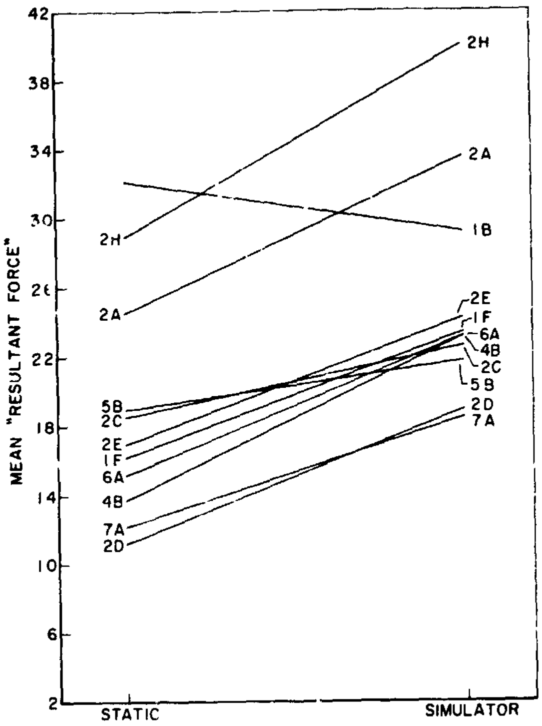

Figure 4. Effects of Fastener Type and Gravity Condition On Resultant Force.

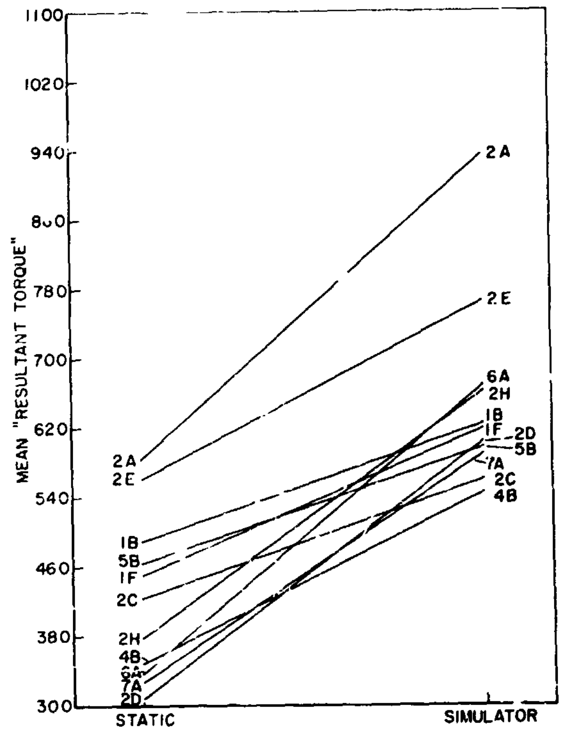

Figure 5. Effects of Fastener Type and Gravity Condition On Resultant Torque.

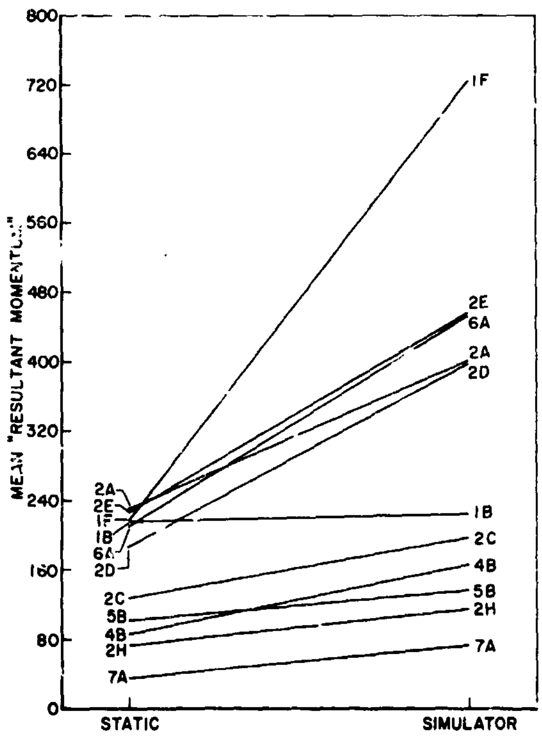

Figure 6. Effects of Fasterer Type and Gravity Condition On Resultant Momentum.

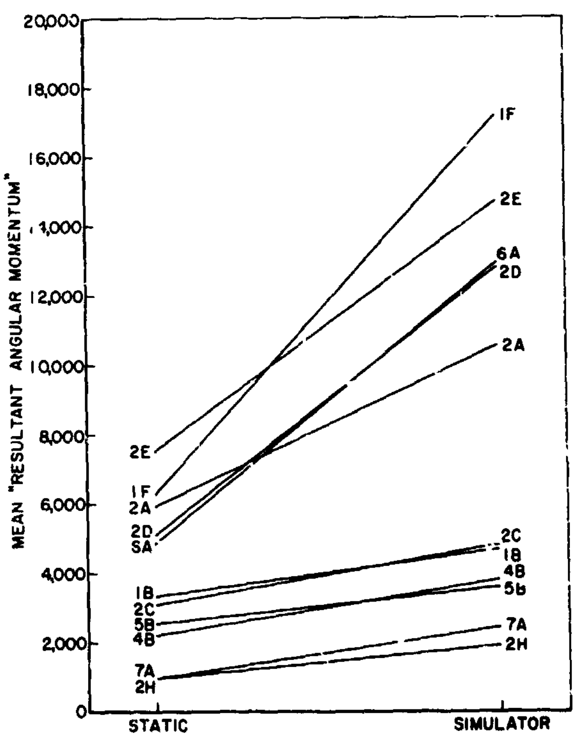

Figure 7. Effects of Fastener Type and Gravity Condition On Resultant Angular Momentum.

TABLE 5

Spearman Rank-Difference Correlation Coefficients Showing Inter-Correlations Among Dependent Variables Within Each Gravity Condition (O-G and 1-G) and Across Both Gravity Conditions (Comb.).

	Operation Time			Resultant Force			Resultant Torque			Resultant Momentum			Resultant Angular Momentum		
	O-G	1-G	Comb	O-G	1-G	Comb	O-G	1-G	Comb	O-G	1-G	Comb	O-G	1-G	Comb
Subjects' Judgments	81**	70**	75**	-25	09	-38	-12	-22	-18	62*	37	63*	71*	57	69*
Operation Time				00	-38	-34	25	-05	13	92**	69*	91**	96**	84**	94**
Resultant Force							76**	75**	67*	32	25	-02	11	-06	-17
Resultant Torque										49	66*	48	34	45	40
Resultant Momentum													97**	93**	95**

NOTES: 1. There were two sets of measures, one for O-G and one for 1-G, for every dependent variable except "subjects' judgments.")
2. All decimal points have been omitted.

* $p < .05$
** $p < .01$

TABLE 6

COMPILATION OF SUBJECTIVE REMARKS BY FASTENER, BY CONDITION

Fastener	Mechanical Failure (Fastener)			Mechanical Bind (Panel)			Alignment and Positioning Problems			False Positive Report			Σ
	Static	Sim	Tot	Static	Sim	Tot	Static	Sim	Tot	Static	Sim	Tot	
1B	1	0	1	0	0	0	0	0	0	1	1	2	3
1F	0	1	1	4	4	8	0	1	1	0	1	1	11
2A	0	0	0	0	0	0	6	6	12	0	0	0	12
2C	0	1	1	1	0	1	0	0	0	1	1	2	4
2D	0	0	0	0	0	0	2	1	3	0	0	0	3
2E	0	1	1	0	0	0	2	1	3	5	3	8	12
2H	1	1	2	1	2	3	0	0	0	0	0	0	5
4B	0	0	0	1	0	1	0	0	0	2	1	3	4
5B	1	0	1	0	0	0	1	0	1	0	1	1	3
6A	0	0	0	0	0	0	3	4	7	0	0	0	7
7A	0	0	0	0	1	1	1	1	2	0	0	0	3
Σ	3	4	7	7	7	14	15	14	29	9	8	17	

although it had not.

DISCUSSION

Of primary concern for the purposes of this study are the main effects of gravity condition and fastener type and perhaps, the fastener x gravity condition interaction. The two main effects were statistically significant for every dependent measure. The fastener x gravity condition interaction was statistically significant in terms of the force, torque and momentum measures but not in terms of operation time.

Gravity Condition Main Effects

Mean operation time was approximately 30% greater for the simulated zero-gravity condition, i.e., 11.64 seconds for zero-gravity vs. 8.88 seconds for normal gravity. However, the momentum data, which reflect force and torque as well as time, suggest that amount of effort required for fastener operation may be almost twice as great for zero-gravity as for normal gravity. Mean resultant momentum measures were 155.57 and 303.14 for the normal gravity and zero-gravity conditions respectively. Corresponding values for resultant angular momentum were 3888.40 and 7936.22.

Fastener Design Main Effects

Fastener 2H, the push-button friction lock, and fastener 7A, the snap slide device, proved superior to all others in terms of the time and momentum data. The lowest mean operation time was obtained for 2H; the second lowest, for 7A. The lowest resultant momentum score was for 7A; the second lowest, for 2H. For resultant angular momentum, scores for 2H and 7A ranked first and second, respectively.

For the resultant force measures, 7A retained its high ranking, second, but 2H dropped to last; hence, 2H's good standing on resultant momentum was due solely to low operation time. The lowest resultant force score was obtained for fastener 2D, one of the screw type fasteners.

In terms of resultant torque, the two best scores were obtained for fasteners 4B and 2D, in that order, while fasteners 7A and 2H dropped to third and sixth respectively. It may be important to note that although fastener 4B, a cam lock mechanism requiring one-quarter turn to operate, was not especially favored by the subjects (mean rank = 6.2), it ranked no lower than fourth on any of the other criterion data.

Fastener 2D, on the other hand, did not show such consistency. Fastener 2D, a screw type device, ranked above average on force and torque measures but fell to last place on the basis of time scores and subjects' rankings. This pattern, which is also characteristic of rankings obtained for fastener 6A, a second screw type fastener included in the sample, is consistent with the subjects' tendency to place a premium on short operation time rather than low force requirements.

With the subjects' apparent emphasis on time, the negative correlation between force requirements and subjects' rankings may be traced to the fact that some fasteners (notably 2H, 7A and 2A) were designed to be operated by impulsive forces, while others were designed for operation by repetitive, low-level forces, e.g., 6A, 2D and 2E. The Spearman rank-difference correlation between time and resultant force was, in fact, -0.34.

Fastener 7A is exceptional in that its design allows a pair of such fasteners to be mounted so that simultaneous operation acts to cancel transmitted forces.

The foregoing considerations, in combination, account for the fact that time was found to be a better predictor of subject preferences than the resultant momentum index which reflects both time and force.

Another general observation of the experimenters was that several fasteners fared poorly because their design features did not include positive visual cues as to functional status, i.e., locked or unlocked.

Fastener Design-Gravity Condition Interaction

The significant interaction between fastener design and gravity condition supports a basic notion with respect to the relationship between task difficulty level and zero-gravity effects, i.e., the more difficult the task under normal gravity conditions, the greater the performance decrement under zero-gravity. (See Figures 3, 6 and 7 especially.) It also should be noted that weightlessness is only a small portion of the stimulus complex encountered by the astronaut in the actual task of space maintenance. The increased performance requirements identified in this study may very well interact with the remaining stimulus environment to increase demands on the astronaut even further.

CONCLUSIONS

Results of this investigation support the following conclusions.

1. Fastener operations in removing and installing access cover panels required, on the average, 30% more time under simulated zero-gravity conditions than under normal gravity.

2. Torque and force requirements, combined with operation time as an index of level of effort, suggest that performance requirements for a fastener operation task are almost twice as great under simulated zero-gravity as under normal gravity.

3. Two fasteners, one equipped with a push-button friction lock and the other a snap-slide mechanism, showed definite advantages over the nine other fasteners included in this investigation in terms of operation time and subject preferences. Although operation time was slightly lower for the push-button device, resultant force and torque measures cast the snap-slide fastener in a more favorable light.

4. Operation time was more highly correlated with subjects' fastener preferences than were indexes of the forces, torques and momentum associated with fastener operation.

RECOMMENDED RESEARCH

Additional research on hand-operated fastener design is needed to evaluate the effects of encumbrances imposed by pressure suits such as are required for extra-vehicular space operations.

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EXPERIMENTAL AND ENGINEERING RESEARCH STUDIES ON TOOLS FOR EXTRAVEHICULAR MAINTENANCE IN SPACE

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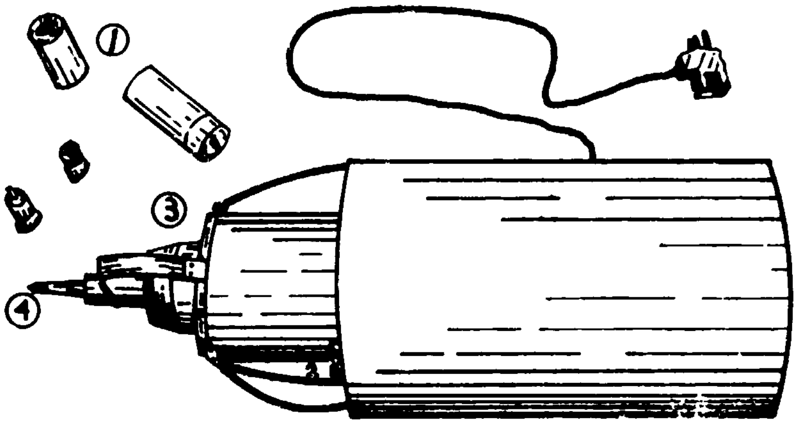
SUMMARY: The joint engineering/experimental program objective is to evaluate space-tool mitten concepts by the development of breadboard models. These models were used by maintenance operators in a series of small experiments that were designed to offer inputs to the evolving tool design.

INTRODUCTION

Raff Analytic Study Associates has conducted a research program on space maintenance concepts and tools for the National Aeronautics and Space Agency since December of 1966.[1,2,3] The major purpose of this multiphased research program was to develop tool mitten concepts for possible use in extravehicular activities (EVA) and to review existing literature on maintenance performance decrements resulting from weightlessness and pressure suited conditions. I will only discuss the tool mitten development phase of our work today.

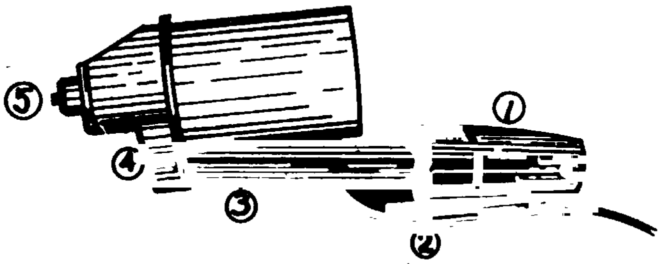
PURPOSE

The purpose of the initial experimental and engineering effort was to develop two operating, bread board models of space-tool mitten concepts. The designation "space-tool mitten" has been selected to include both developed tools, the space mitten and the tool mitten. Each of these tools is a multipurpose power tool that was designed to function as a wrench or screwdriver. The tool mitten is an impact tool (Figure 1) that was designed as a cylindrical metal structure having storage sites for tool attachments emplaced within annular wells toward the face of the tool. Tool attachments can be exchanged by merely pulling a particular attachment out of its well and then mating it to the chuck of the tool mitten. Each attachment is restrained by a flexible metal clockspring and slip ring. The space mitten (Figure 2) originally planned for bare-handed operation, has its motor near the maintenance worker's elbow and



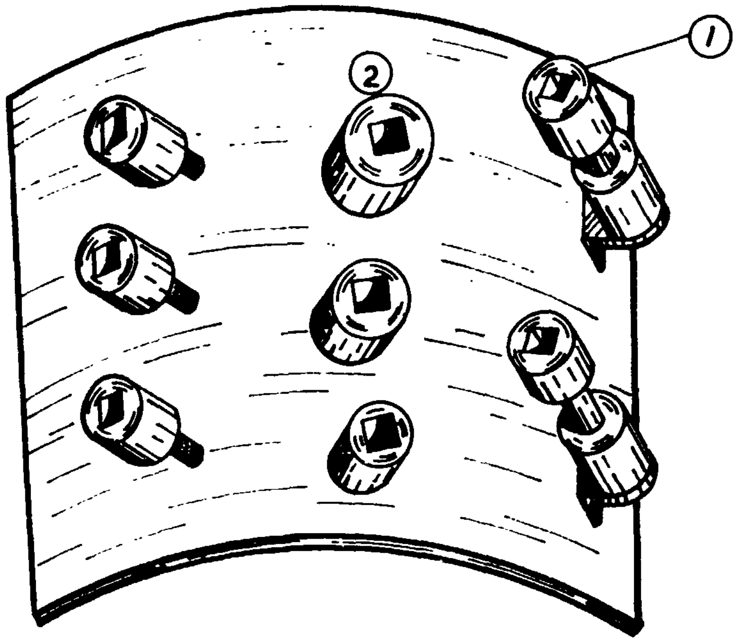
1. TOOL ATTACHMENTS
2. TOOL WELL(STORAGE SITES)
3. FLEXIBLE METAL CLOCKSPRING
4. PHILLIPS HEAD SCREWDRIVER

FIGURE 1 -- TOOL MITTEN (Breadboard Model)



1. MOTOR
2. SWITCH
3. DRIVE TRAIN
4. GEARS
5. CHUCK

FIGURE 2 -- SPACE MITTEN
(Breadboard Model)



1. SCREWDRIVER ATTACHMENT
2. HEX SOCKET ATTACHMENT

FIGURE 3 -- TOOL CUFF
(Breadboard Model)

was not developed as an impact tool. It was designed for manual and power applications. A tool attachment storage site called a tool cuff (Figure 3) is used as an accessory to the space mitten for the storage of hex sockets and screwdriver attachments.

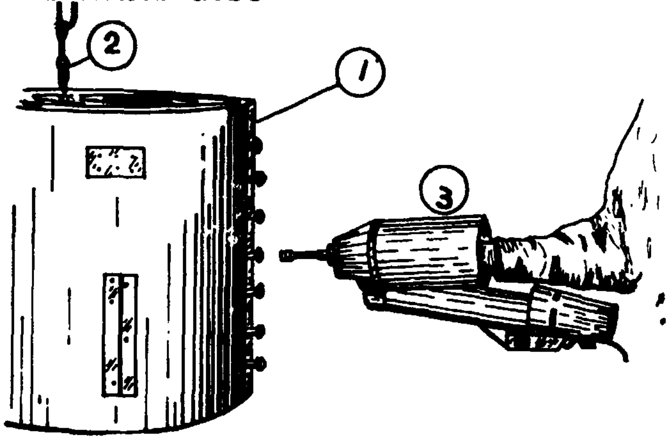
The tool cuff operates on the principle that the tool attachments will be stored separately from the tool. However, the storage and removal of tool attachments will require the use of the power tool. The storage sites for the impact wrench sockets are tapered studs containing split captured nuts. Each tapered stud retains a single tool socket; as the socket is emplaced and torqued, the split nut is driven down over the stud, expands and binds the socket. To remove the socket, the operator backs off on the nut, thereby allowing the nut to release the engaged socket. Allen head, slotted, and Phillips screwdrivers are held in place by the squeeze action of the inner walls of small chambers that bind the screwdriver attachments. The removal of the screwdriver merely allows the internal walls of the cylinder to relax, thereby releasing the tool.

Two maintenance task assemblies, a maintenance panel and a maintenance task cylinder, were developed for use in five of the six experimental evaluations. Both the maintenance panel and the maintenance task cylinder were employed in the static (stationary) mode during testing. However, the maintenance task cylinder (Figure 4) was also suspended from a spring (dynamic mode) that permitted restricted five degrees of freedom. Further, the maintenance task cylinder was modified during Phase II to accomodate a simulated spacecraft hatch that had thirty-six 1/2-inch hex bolts placed 2-1/2 inches apart around the periphery of the hatch. The task cylinder was also designed to be used to evaluate the ease with which operators could work within confined spaces. Figure 5 shows one side of the task cylinder and the egg crate dividers that permitted depths of 6, 9, and 12 inches and overall square dimensions of 8 x 8, 12 x 12, and 16 x 16 inches.

STUDY GOALS AND DESIGN OBJECTIVES

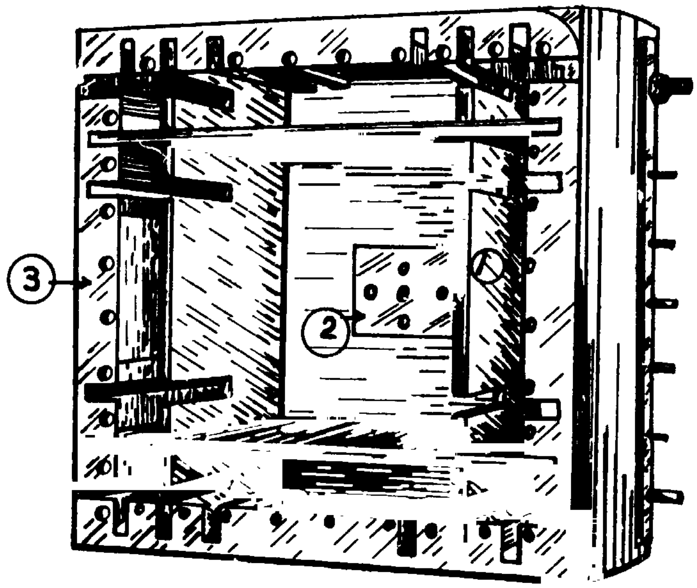
It is apparent that the goal for developmental EVA tools must be to alleviate performance decrements associated with pressure suit wear and weightlessness. Since the space worker will be somewhat restricted in mobility and dexterity, one solution to efficient tool design is to combine several operations into one, thereby relieving the worker for critical activities yet retaining the one-handed operation concept. An example of this goal is the design of a space-tool mitten that not only provides the space worker with suit-glove protection, but also allows for rapid tool attachment exchanges. Further, these tools are designed to allow the operator greater access to work areas by the relocation of the tool motor away from the working end of the tool, thereby achieving less bulk in the forward end of the tool.

The plan of the research program, conducted by Raff Associates under Contracts NASW 1537 and NASW 1590, was to evaluate the space-tool mitten concept by performing



1. METAL BAR
2. SPRING
3. SPACE MITTEN

FIGURE 4 -- MAINTENANCE TASK CYLINDER



1. EGG CRATE DIVIDERS
2. SCREWDRIVING PLATE
3. HATCH COVER INTERFACE

FIGURE 5 -- "EGG CRATE" DIVIDERS ON MAINTENANCE TASK CYLINDER

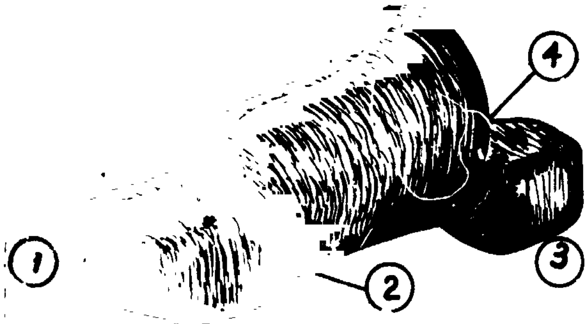
a variety of human factors experiments. The experiments were aimed at gaining an insight into the design of tools for a space maintenance worker performing extravehicular maintenance and assembly tasks in a zero g environment. The design objectives for this program were: (1) to develop a space tool for bare-handed operation; (2) to develop a multipurpose power tool; (3) to allow for rapid and efficient tool exchange procedures; (4) to offer suit-glove protection; and (5) to permit one-handed tool operation. In the course of our design and evaluation program, Raff Associates personnel added a number of other objectives or concepts as follows: (1) locate the center of mass at the operator's hand; (2) utilize a self-contained power source; (3) design for less tool bulk in the front end of the tool; and (4) develop a fiberglass shell to made the tool lighter in weight.

The basic philosophy of the evaluation program was to identify and correct design deficiencies in the initial breadboard models by using human factors experiments and engineering analyses. These research studies were performed in order to develop firm requirements for an operational prototype space tool.

Even though engineering developmental and experimental efforts are generally performed sequentially, experience has shown that the two efforts complement each other. However, it was not always possible to separate the rationale for re-tooling on the basis of mechanical or electrical deficiencies, or on the requirements of the human operator for comfort, safety, and visibility. Therefore, some compromises in rigorous experimental control were made. Sometimes compromises were made because of emergency tool and maintenance task assembly repair.

The experimental program consisted of five small experiments identified by the following designations: (1) Space Mitten and Tool Mitten Evaluation; (2) Manual Tool Evaluation; (3) Hatch Test; (4) Screwdriving Test; (5) Access Area Study. In addition, a qualitative critical evaluation was made of the performance of maintenance workers using the space mitten with the tool cuff and using the tool attachments of the tool mitten.

The most recent effort, Phase III, of the research and development program was a basic engineering development of a prototype, impactor type, multi-purpose space tool. The device features a fiberglass shell and human engineered internal controls. An artist's conception of this device is shown in Figure 6. Figure 7 depicts the space-tool mitten about to retrieve a tool attachment from the tool cuff.



1. CHUCK
2. FIBERGLASS SHELL
3. MOTOR
4. BATTERY (Not Shown)

FIGURE 6 -- SPACE-TOOL MITTEN (PROTOTYPE)

EXPERIMENT 1

This experiment was conducted to evaluate operator performance when using either the tool mitten or space mitten to perform a number of torqueing tasks. The maintenance task was to tighten five hex bolts of the same size into a metal bar containing imbedded welded nuts. There were five hex bolt sizes ranging from 3/4 to 7/16 inches. Further, the subject either performed this task while the maintenance task cylinder was stationary (static mode) or when the cylinder was suspended from a triangular support (dynamic mode).

Performance was measured in terms of the time taken to complete the tightening task. The results indicated that there was no significant difference between the performance of the two tools when the maintenance cylinder was in the static mode. In fact there was only a slight three percent superiority in the use of the space-tool as compared with that of the tool mitten in the static mode. On the other hand, the performance with the tool mitten took 37 percent longer than the task performance with the space-tool in the dynamic mode.

In reviewing the data for a particular tool under static and dynamic modes, it was noted that operator's performance with the space mitten increased from 13.9 seconds to 24.1 seconds (approximately a

1. SPACE-TOOL MITTEN
2. TOOL ATTACHMENT STORAGE SITES
3. FIBERGLASS CASE

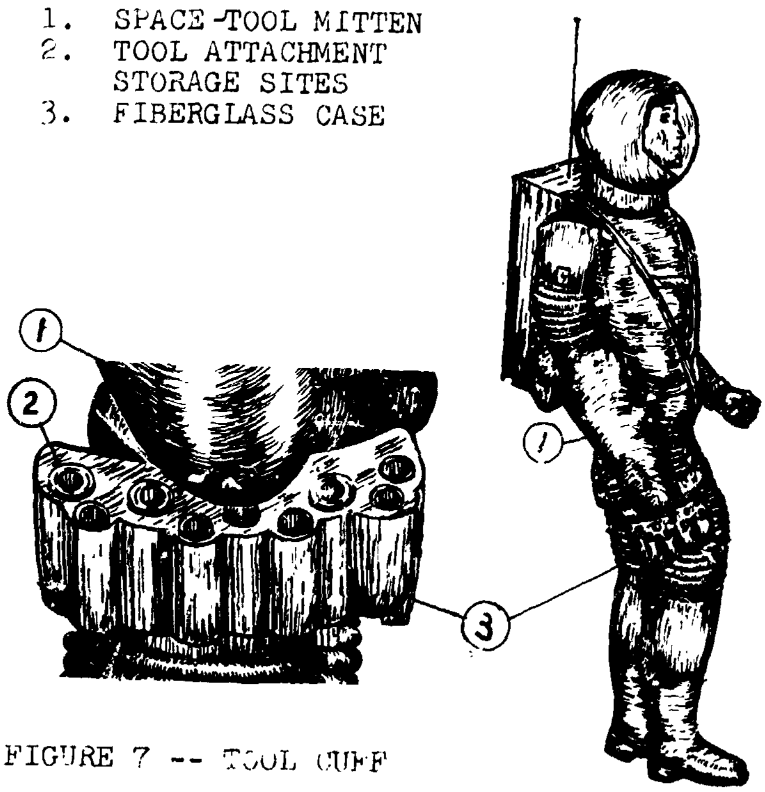


FIGURE 7 -- TOOL CUFF

73 percent increase). On the other hand, when one considers the performance time under static and dynamic modes for operators using the tool mitten the increase from 14.3 to 33.1 seconds was an approximate 132 percent increase!

These results clearly indicate the superiority of the space mitten under these tasks conditions when the worksite is unstable. No significant learning effects were observed during the three trials that were required in either tightening or loosening of the five hex bolts.

EXPERIMENT 2

This experiment was basically a validation of the previously described experiment. For this particular experiment, two subjects with maintenance experience were used. The task for these subjects was to tighten 32 of the 36 1/2-inch hex bolts distributed around the periphery of the hatch cover that was attached to the maintenance task cylinder. The task cylinder was either in a static or a dynamic mode during this experiment. The experiment was planned as a 2 x 2 x 2 factorial design in which the variables were: (1) task cylinder in either the dynamic or static mode, (2) space mitten versus tool mitten, and (3) subject 1 versus subject 2. The study was conducted as a complete factorial with the assignment of the experimental conditions by Latin square randomization. Each subject participated in nine blocks of the experiment; each block consisted of four trials of two tools with each tool used in the dynamic and static modes. Unfortunately, some of the bolts became cross threaded or damaged during the experimental trials. However, due to the randomization procedure these problems were rather constant for all experimental treatments.

The results of this experiment indicated essentially equivalent performance with the space mitten and tool mitten for the hatch fastening operation while the cylinder was in the static mode. Although, the analysis of variance did not reveal a statistically significant difference between the performance of the two tools in the dynamic mode, the data indicates that the tool mitten's performance time was 19 percent longer than the space mitten's performance time. It is interesting to note that the decrement in performance resulted in a 241-percent increase in time. On the other hand, performance with the tool mitten resulted in a tremendous 321-percent time increase. These results tend to substantiate those findings noted in the earlier experiments. It appears that the dynamic cylinder mode as a simulation of a frictionless work environment offers a reasonable fascimile.

EXPERIMENT 3

As part of the overall research plan to evaluate the multipurpose features of the space-tool mitten, a brief experiment was conducted to investigate the capability of the space mitten and tool mitten to operate as a screwdriver. In addition, a speeder handle or crank was used in the investigation.

The same two experienced subjects were used in this experiment. The task of the subjects was to tighten five recessed Allen screws with screwdriver bit sizes 3/16, 1/4 and 5/16 inches. For this particular experiment, the excessive torque of the tool mitten was reduced by an autotransformer that lowered the voltage from 120 to 70 volts. This procedure prevented the tool from damaging the nuts and bolts as it had in previous experiments. This experiment was conducted as a 3 x 3 factorial design with Latin square randomization of experimental conditions.

The results indicated that the space mitten was significantly more efficient than the tool mitten or hand operated crank. The two operators took approximately 61-percent longer to complete the task with the tool mitten than with the space mitten. Further, the use of the hand operated crank resulted in a 141-percent time increase for the performance of these screwdriving task, as compared with the space mitten.

It appears from this experiment that such multipurpose functions are well within the realm of capabilities for the next generation of space-tool mittens.

EXPERIMENT 4

This experiment was conducted to investigate the ease in which various tools could reach simple worksites areas when accessibility to areas becomes a problem. The subjects' task was to use a tool (space mitten, tool mitten, Allen key, or crank) to mate with the five Allen screws that were located on a metal plate within the access opening. The subject was not required to complete the screwdriving operation but merely to be able to mate the tool with the five fasteners. The "egg crate" dividers allowed the subjects a variety of access apertures. The particular pairings of access depths and access areas were randomly assigned to the nine experimental blocks. Each experimental block consisted of four trials with each of the four tools. The subjects gave their preferential rankings of the four tools during each of the nine experimental blocks. The results of this preferential ranking indicated the following: The Allen head wrench was preferred over the other three tools followed by the crank, space mitten and tool mitten. These rankings are based upon median rankings over all conditions.

This experiment suggests that for screwdriving operations in which recessed hex head tools are used, conventional tools, that is, Allen head wrenches appear to be more suitable. The choice of a power tool for this function must necessarily depend upon the frequency and number of these operations that are required.

ENGINEERING EVALUATION

The tool mitten previously illustrated is quite bulky and unwieldy. The motor location in front of the operator's hand and the location of the tool attachment around the periphery of the tool cause visibility and control difficulties.

We further noted that tool attachment exchange was extremely difficult. For example, the retaining ring on the chuck remained quite stiff during the evaluation and caused operators to use a great deal of force in compressing the retaining ring while placing tool attachments on the chuck. Further, the tool attachment exchange is made even more difficult because the length of the tool is such that the operator is unable to effectively use his opposite hand to make the tool exchanges. One problem was related to the tremendous output torque that was quite excessive for the maintenance tasks required in the program. Because of the high torque, nuts, and bolts often became cross threaded during the experimental series.

The space mitten was much smaller than the tool mitten since its overall diameter was reduced by having the motor toward the end of the tool. Further, the introduction of a canted handle that still allowed the operator positive control of the tool resulted in some reduction of the overall diameter. The canted handle was dis placed about 15° from the vertical and had a cross sectioned T-shaped handle. The narrow end of the T fitted easily into the V of the grasping hand. It is thought that in later designs of the space mitten, the front structure should be reduced in size to further improve visibility and control for the operator. It is also felt that further reductions in weight and size can be accomplished while still allowing the operator adequate freedom of motion during tool use.

SPACE-TOOL MITTEN

Following the experimental/research program just discussed, Raff Associates developed the tool shown earlier in Figures 6 and 7 called the space-tool mitten. This device embodies many of the findings and concepts that were developed during the experimental program. One innovation was a tool that was developed with a fiberglass shell in order to reduce the weight. Further, the space-tool mitten is powered by a silver cadmium battery that runs an approximate 1/4-horsepower electric motor. The silver cadmium rechargeable battery is housed on the side of the tool and is in a position to allow quick forward loading battery exchange. The tool has a 3/8-inch chuck and will accept a variety of hex sockets and screwdriver attachments. The tool cuff associated with the space-tool mitten is also made of fiberglass and operates on the same essential principles described earlier. Raff Associates feel that at the present time we are at an interim step in the development of the space-tool mitten. We have not formally evaluated this second generation space tool, but we see the need for an evaluation program under simulated zero-g and pressure suited conditions.

It has not been determined whether we will continue to develop the current space-tool mitten for rigorous simulation and actual flight tests or develop an entirely new mitten concept. We feel at this juncture that we have demonstrated a valid concept in this space-tool mitten.

The potential advantages of our concepts are as follows: (1) potentially a bare-handed tool operation, (2) multipurpose power tool, (3) rapid tool attachment exchange, (4) suit-glove protection, (5) one-handed operation, (6) center of mass at the operator's hand, (7) less tool bulk in the front end of the tool and (8) fiberglass housings for space tools. Further, we believe that our methodology of mating engineering and experimental testing is also quite valid for contributing to overall design improvements.

We believe that we are moving in the right direction for better space tool designs.

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COLD CATHODE WELDING FOR SPACE APPLICATIONS

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SUMMARY: A program was performed involving design, fabrication and test of two manually-operated cold cathode electron beam welding tools. One tool produced a disc-like electron beam for welding tubing, and the other, a planar beam for welding sheet metal. Both tools fit on a single handle.

INTRODUCTION

Recent advancements in space technology require methods to be developed for sound, leakproof joints in extraterrestrial structures. Anticipating these needs for "space fabrication" can provide an insight into the growing importance of this area of fabrication technology. For example, large boosters become inefficient with increasing booster size, requiring the weight and/or volume of the payload to be limited. The effects of this limitation can be bypassed by sending the payload aloft in segments and assembling it in space. Smaller and less powerful boosters also could be used to power large payloads over very great distances if the payloads originated from sources having less gravity than earth (e.g., from an orbiting space station or from the moon).

Complex space or moon stations will have to be manned, and be of high structural integrity so that the safety of the station-keeping crew never is imperiled. Such structures must be capable of containing a life-sustaining environment. Unless a "shirt-sleeve" environment is maintained within the structure, many complex situations will result. "Shirt-sleeve" environments in space or on the moon must also be enclosed within a hermetically-sealed container to ensure retention of air.

Although many mechanical methods exist for the preparation of "leak-tight" joints, not one of them can equal a metallurgical bond heremetically. The inadequacy of mechanical joints is related to the leakage which remains after completion of even the best joint. When a large structure is fabricated using mechanical joints, the total leakage of all the joints can result in a significant loss of atmosphere. This atmosphere must be replaced to ensure the safety of the inhabitants. If the loss, due to structural defects, becomes sufficiently large to affect the "normal" resupply schedule, then the effective cost of the structure increases, and can easily become greater than the first manufacturing cost. Thus, a method must be provided for in situ fabrication by welding of such structures.

In addition to the need for a welding method for primary fabrication are the requirements for repair and maintenance welding. In both cases the structural integrity of the assembly and the retention of a habitable environment are of paramount importance to crew safety. Therefore, any fabrication repair or normal maintenance procedure must include this capability for fusion welding. Thus, it will be necessary to provide metals joining equipment for the use of astronauts during extended missions.

Normally, welding is a highly skilled trade that frequently requires many years for development of high-caliber proficiency. Therefore, relating an astronaut's skill to this requirement results in three possible alternatives:

a. Train the astronaut to perform the welding task,

b. Train skilled welders to become astronauts, or

c. Devise a welding process that is manual, yet sufficiently automatic so that little training beyond the most rudimentary indoctrination will be required.

Obviously, the third alternative is most desirable because it permits every astronaut to function as a skilled welder whenever the need arises. This is the primary philosophy behind the use of an electron beam produced by a cold cathode as the heat source for extraterrestrial welding.

The electron beam produced by a cold cathode device can be shaped in a pattern fixed by the cathode configuration. This unique characteristic provides the basis for the inherent simplicity of the tool whereby heat energy is transferred to the workpiece in a pattern fixed by the cathode configuration. This feature of the electron beams produced by cold cathode discharges was developed privately by United Aircraft Corporation. The advantage of this "extended geometry" electron beam for extraterrestrial joining is obvious: a few simple cathode configurations can provide a great degree of joining versatility without requiring significant operator skill.

In addition to welding, the materials joining tool has been used to prepare a brazed lap joint. Other programs now underway at United Aircraft have shown that cold cathode electron beams can be applied successfully to heat treating and also to float-zone growing of single crystals.

ELECTRON BEAM GENERATION

Electron beams in cold cathode devices are the result of abnormal flow discharges in low pressure gaseous regions. The cathodes are at high negative potentials while the remainder of the system is maintained at ground potential. The low pressure (approximately 0.1 to 1 Torr, depending upon the gas, is required to provide gas atoms for plasma formation. If the quantity of gas atoms (e.g., the gas pressure) is too high or low, the abnormal discharge producing the directed electron beam will not form.

During operation, the ambient gas is transformed into a plasma by the potential difference between the cathode and ground.

The positive particles in the plasma are attracted to the cathode and strike it causing secondary emission of electrons from its surface. Also, some neutral particles and photons will strike the cathode causing additional secondary emission electrons to be formed. The secondary emission electrons, and also the electrons formed as a result of the plasma formation, are directed away from the cathode with a velocity related to the potential of the cathode with respect to ground.

CATHODE TECHNOLOGY

The basis of the beam shaping characteristics of the cathode is related to the formation of equipotential surfaces about the cathode, approximately parallel to the cathode surface. The electrons in the beam are accelerated in a direction normal to the equipotential surfaces; therefore, electron beams will come to focus at the geometric center of curvature of the emitting surfaces. This focusing characteristic is relatively insensitive to changes in voltage and pressure so long as the discharge operation is in the high-voltage abnormal glow mode.

The beam shape can be altered to meet the exact requirements of an application by the use of cathode shields. This technique has been used in the linear tool to control the length of the beam. The critical factors in the use of shields are their size, shape and placement. Shields placed close to the cathode cast a shadow upon it, preventing its bombardment by the plasma, thereby precluding beam formation. These shields prevent cathode emission and are not heated by the beam. A second method of effecting changes in the beam is to physically block the beam. In this method, the shields blocking the beam are heated by it, and as a result, this method of beam operation is undesirable.

EQUIPMENT DESIGN

The design of the material joining tools was based upon basic operational and physical requirements. Essentially these were:

a. Operating potentials should be 5.0 kv or less to assure elimination of X-radiation.

Annular butt welds shall be made in 3/8-inch diameter tubing.

c. Full penetration is required in 0.060-inch wall, 3/8-inch diameter stainless steel tubing.

d. Windows shall be provided to permit observation of the welding.

e. The cathode and shield of the annular tool will be split to provide maximum usefulness of the tool.

f. Fixturing will be provided in the annular tool to hold the tube halves in the beam and to adjust the joint into the beam focus.

g. Workpiece seals will be made of metal to preclude vulnerability to degradation by high temperatures.

h. An adjustable cathode shield will be provided on the linear tool to produce linear beams no longer than six inches, nor shorter than 0.030 inch.

i. The ambient gas will be admitted to the cathode area through the high-voltage potting.

j. Interlocks will be provided to prevent activation of a properly wired power supply * until the tools are safely closed, or are in full contact with the work.

k. The handle will contain the gas supply, the gas supply valve and regulator, and the high voltage feedthrough.

l. The handle will be oriented at 30° to the plane of the work to provide a comfortable operating position.

Annular Tool Design

The stainless chamber, shown in figure 1, is 6 inches in diameter, and 5 1/2-inches long; a 90° segment of the chamber forward of the high-voltage feedthrough opens to permit placing the tube segments within the cathode. The window location provides an unobstructed view of the cathode and work. A simple, four-leaf shutter assembly is used at the workpiece access hole in the chamber to reduce the flow of gas from the chamber to the low-pressure surroundings. In this design, two leaves of the shutter are in the basic chamber, and two on the movable "door" segment. Two fixture adjustments are provided, the first tightens the fixture about the tube, and the second moves the fixture to align the joint to the beam focus circle. The weight of the annular tool, including the fixture but excluding the handle, is 9.5 pounds.

* The power supply used during this program was not designed for on/off control from the trigger of the tool. A power supply designed specifically for the Materials Joining Tool should have this circuitry.

Linear Tool Design

The linear tool shown in figure 2 is considerably simpler than the annular tool. It consists of a chamber with an open bottom; the cathode is at the top of the chamber to produce a beam whose focal line is in the plane of the bottom of the chamber. In addition, an adjustable shield system provides variations in beam length. The weight of the linear tool is 4.5 pounds.

Two large windows are located on the opposing long faces of the tool chamber. Originally, Lexan * was considered for the windows because indications were that it would withstand the operating conditions of the tool and not lose transparency. However, the Lexan softened and deformed under the temperatures reached during sustained operation. Although it probably could be used for many welding tasks, Lexan was not suitable for welding heavy gage or high melting-point materials, nor could it be used for brazing or general heating. A clear grade of mica was obtained; it was not adversely affected by cathode operation, and performed satisfactorily.

Handle Design

The handle design was based upon previous work at Hamilton Standard. Its size and shape in effect had been tested previously in Hamilton Standard's Man-Rated altitude chamber by a pressure-suited test subject using a tool with a similar handle at an ambient pressure of 2.6×10^{-5} Torr.

A low-pressure, gaseous environment is required in the region of the cathode

* Lexan is a proprietary plastic manufactured by the General Electric Company.

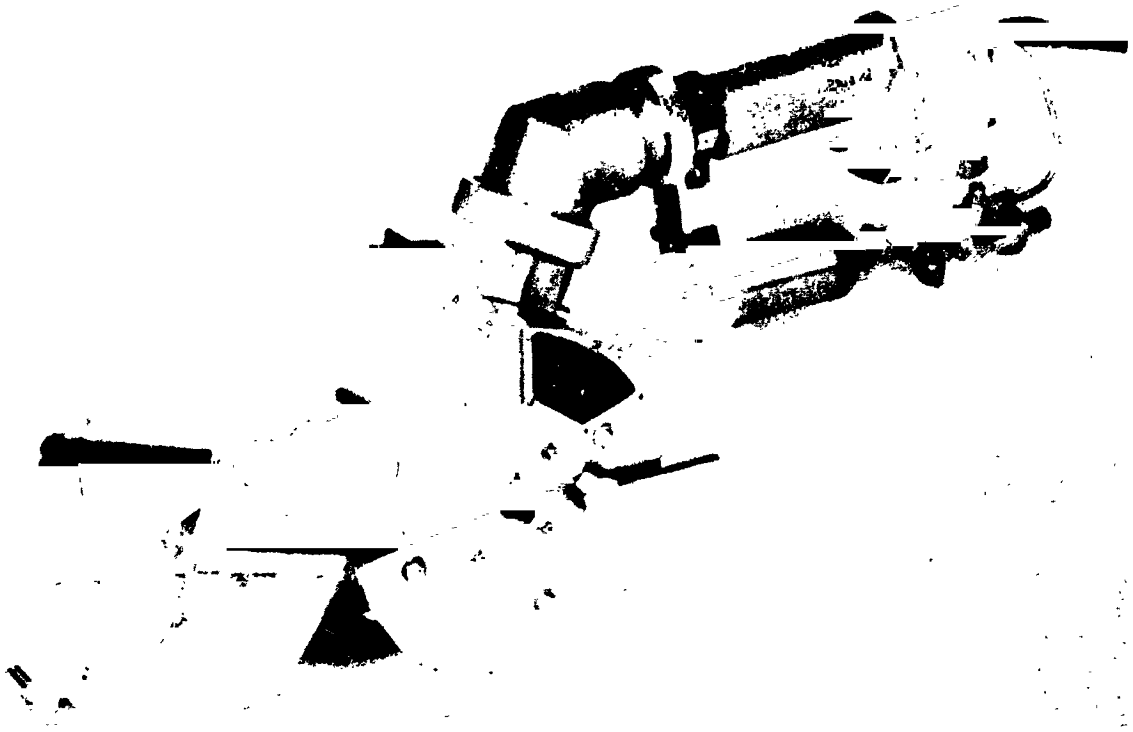


FIGURE 1. ANNULAR TOOL, OPENED, SHOWING CATHODE AND TUBE IN PLACE

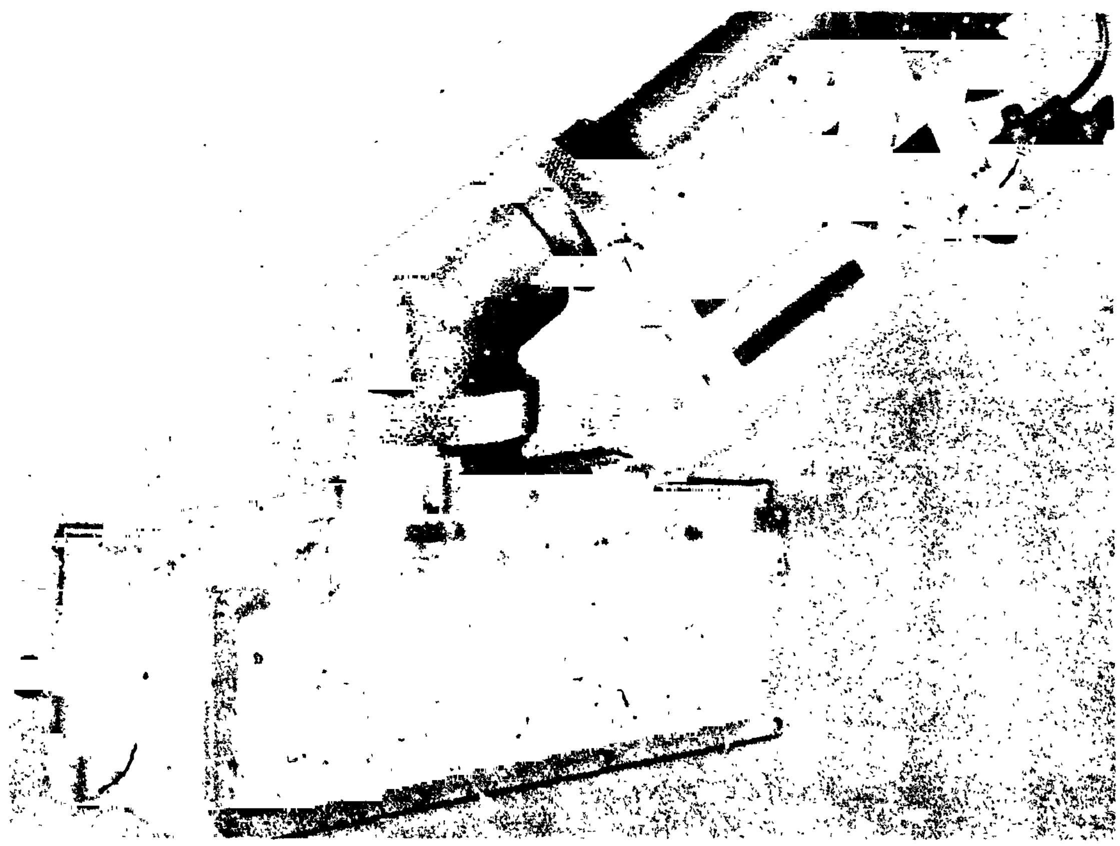


FIGURE 2. LINEAR TOOL WITH HANDLE ATTACHED

to supply atoms for ionization and subsequent beam formation. Terrestrial cathode devices use a bleed valve to introduce a gas to the cathode area while the system is being evacuated to produce a dynamically balanced pressure. The tool will operate in a similar manner: the space environment will act as the pump while the seals, using simple metal-to-metal contact, will provide the fixed leak. A small gas cylinder, a valve, a pressure regulator, and a bleed valve (all included in the handle assembly) comprise the gas supply system.

The gas system shown with the handle in figures 1 and 2 consists of a small, 75 ml., gas cylinder which can be pressurized to 400 psig. Downstream of the cylinder a toggle valve is used for on-off control; the toggle action facilitates easy operation for an astronaut wearing space gloves. The remainder of the pressure system, in order, consists of a needle valve, a pressure regulator and a second needle valve. The first needle valve acts as an anti-surge valve protecting the regulator from suddenly receiving the full impact of the gas pressure within the bottle. Without this, the regulator oscillates, and the entire contents of the gas cylinder are lost through the regulator relief port. The second needle valve is used in conjunction with the pressure regulator to provide gas pressure and flow control. Operation of the system will be limited to a 200-psig charge within the gas cylinder to ensure that the system is operated safely under the upper operating level of each of the components.

WELDING TESTS

A comprehensive series of welding tests were performed using the annular tool. Stainless steel, aluminum, copper, and titanium tubing, and also copper, stainless steel, titanium and Hastelloy W rod and wire were used for welding tests as shown in table 1.

Annular Tool

Initial tests were performed to evaluate the device as a welding tool. These tests verified the speed and ease of making butt-weld joints in tubing, rod and wire. All material was cut square and, except when prohibited by test requirements, was cleaned prior to welding.

Weld Capability

The first test sequence was to study the weldability of the four different tubing materials indicated in table 1. All welding was performed using argon as the shielding gas. The arrangement of the annular tool in the bell jar for these tests is as shown in figure 3.

Satisfactory welds were made in stainless steel, copper and titanium; whereas, aluminum could not be joined satisfactorily. Typical welds in these materials are shown in figures 4 and 5. The samples shown in figure 4, titanium, aluminum, and copper, were all made in an argon atmosphere; the stainless steel samples shown in figure 5 were made in argon and air as the ambient atmospheres.

Additional attempts to weld aluminum were performed using nitrogen, helium, and air atmospheres. These were not successful. In addition to normal cleaning procedures, the aluminum samples were degreased in methylethyl ketone (MEK), abraded using emery cloth at and near the faying surfaces, and then washed a second time in MEK. However, this treat-

TABLE I. TEST MATERIALS

ALLOY AND DESIGNATION	SIZE, INCH O.D.	WALL	THICKNESS
TUBING			
STAINLESS STEEL, AISI 304	3/8	0.065	—
COPPER, OFHC	3/8	0.065	—
TITANIUM, C.P.	3/8	0.049	—
ALUMINUM, 5052–0	3/8	0.049	—
WIRE AND ROD			
STAINLESS STEEL, AISI 304	0.125	—	—
COPPER, OFHC	0.125	—	—
TITANIUM, C.P.	0.044	—	—
HASTELLOY W	0.035	—	—
SHEET			
STAINLESS STEEL, AISI 304	—	—	0.060
STAINLESS STEEL, AISI 304	—	—	0.030

ment did not provide any improvement in weldability of the aluminum. It has been hypothesized that the difficulty was because the beam did not break up the surface aluminum oxide. When aluminum samples were heated, the aluminum metal (encased in an oxide sheath) was heated to greater than its 1, 100 - 1, 200F melting range while the aluminum oxide on the surface with a melting point of 3, 722F, remained solid. However, the oxide layer was sufficiently thin to be easily deformed while the aluminum was molten, explaining the puckered surface of the weld zone seen in figure 4.

The formation of aluminum oxide proceeds so rapidly that it is impossible for an aluminum surface to remain free from oxide unless the surface is mechanically abraded in an oxygen-free, inert atmosphere or in a vacuum. Thus, welding aluminum with this tool should be reevaluated during manned testing by a pressure-suited test subject in a man-rated altitude chamber.

Welded tubes were sectioned, mounted, and polished to provide an examination of joint quality. Photomacrographs of the weld joints in copper, stainless steel and titanium are shown in figure 6. The photos verify the presence of full penetration weld joints. More detailed examinations are shown in figure 7. The base metal microstructure of stainless steel, shown in figure 7a, is typical of stainless steel; the elongated voids are related to the tube-drawing operation. Figure 7b shows the weld metal with the outer surface of the tube at the top of the photograph. Large dendritic grains are apparent only at the outside surface indicating that that surface cooled slower than

FIGURE 3. ANNULAR TOOL IN PLACE FOR WELDING TESTS

TITANIUM ALUMINUM COPPER

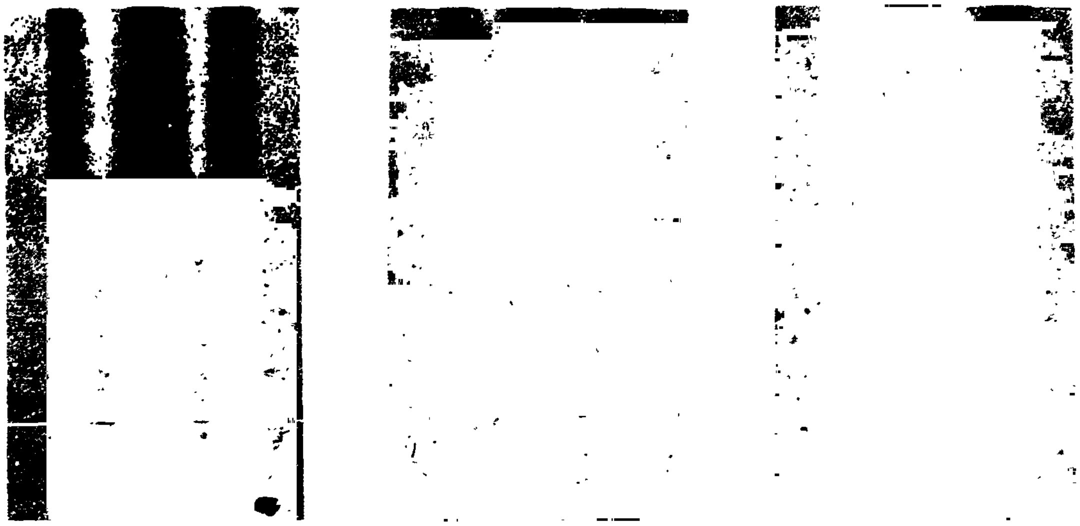


FIGURE 4. WELDS IN TUBING MADE WITH THE ANNULAR TOOL

ARGON ATMOSPHERE AIR ATMOSPHERE

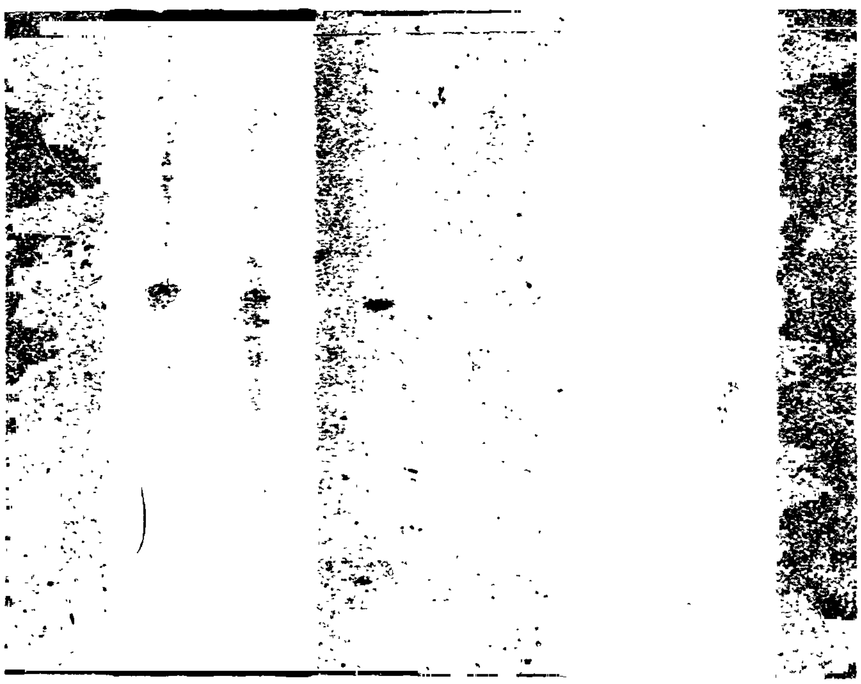
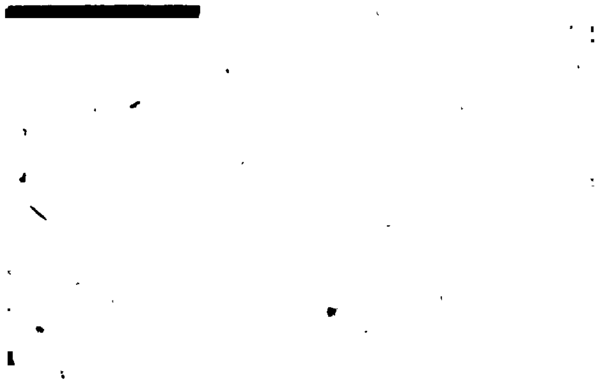
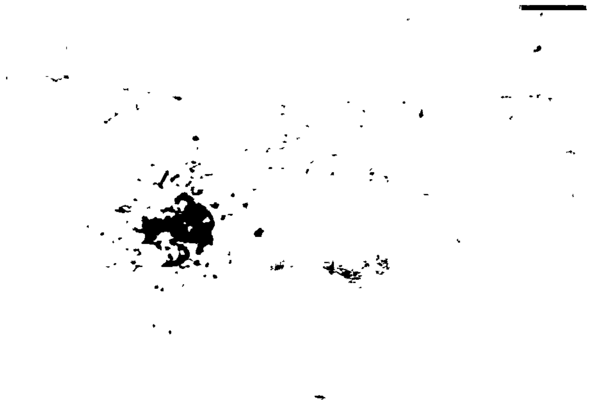


FIGURE 5. WELDS MADE IN STAINLESS STEEL TUBING MADE WITH THE ANNULAR TOOL



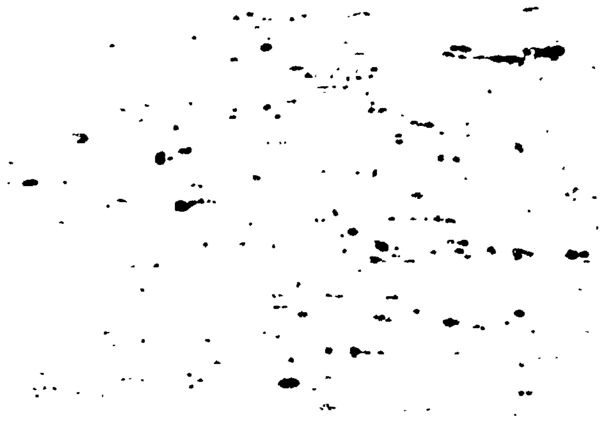
A. OFHC COPPER TUBING



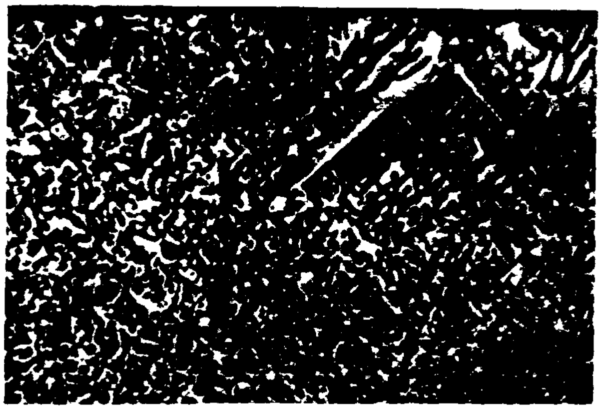
B. STAINLESS STEEL TUBING

C. TITANIUM TUBING

FIGURE 6. PHOTOMACROGRAPHS OF WELDS IN COPPER, STAINLESS STEEL, AND TITANIUM TUBING, 15X MAGNIFICATION



A. STAINLESS STEEL BASE—METAL



B. STAINLESS STEEL WELD METAL

C. TITANIUM WELD METAL

FIGURE 7. PHOTOMICROGRAPHS OF STAINLESS STEEL AND TITANIUM 250X MAGNIFICATION

the inner portions of the tube wall. This was the result of the heat sink effect of the tube material. The titanium weld metal shown in figure 7c is acicular alpha, a structure which commonly occurs in titanium weld metal.

Effect of Atmosphere

The second evaluation was to determine the effect of each of four different atmospheres on welds in stainless steel tubing. Of the four gases tested, air, argon, helium, and nitrogen, only air did not produce good welds. A tendency toward incomplete penetration in the welds was encountered when welding in air. Furthermore, the air caused oxidation of the weld which undoubtedly had a deleterious effect on weld properties.

The average energy required for welding with each of the four gases was determined; these data are reported in table II. The data for air are suspect because of the inability to completely penetrate through the sample. The incomplete penetration of the welds made in air are shown in figure 8a. Note especially the broad, shallow melted region. This can be seen at higher magnification in figure 8b.

In tests to determine operating pressures and gas flow, a solid 3/8-inch diameter stainless steel rod was inserted in the cathode. The tool was operated at five different air pressures and three different helium pressures; the beam current and voltages were measured and the results plotted as shown in figure 9. In figure 9a, the highest pressure reported for compressed air is 0.148 Torr, approximately 0.050 Torr less than the lowest helium pressure. The power at the upper end of each of the curves was approximately 1,000 watts where melting of the rod occurred. This test indicated that three to four times more helium than air will be required to produce a given beam power. Furthermore, tests performed previously have shown that air, argon, and nitrogen operate at approximately the same pressure level.

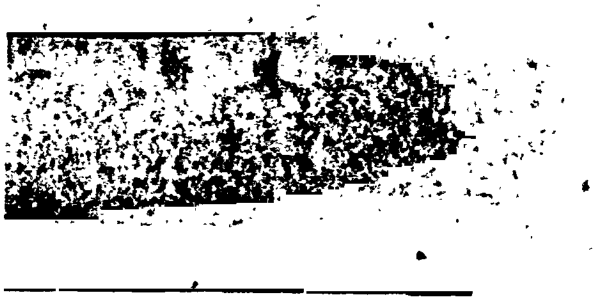
Nitrogen is interstitially dissolved into titanium causing embrittlement of the weld.

TABLE II. EFFECT OF ATMOSPHERE

GAS	NUMBER OF SAMPLES	AVERAGE ENERGY INPUT, KILOJOULES
ARGON	7	21.54
AIR	6	14.42
HELIUM	7	13.39
NITROGEN	4	15.83

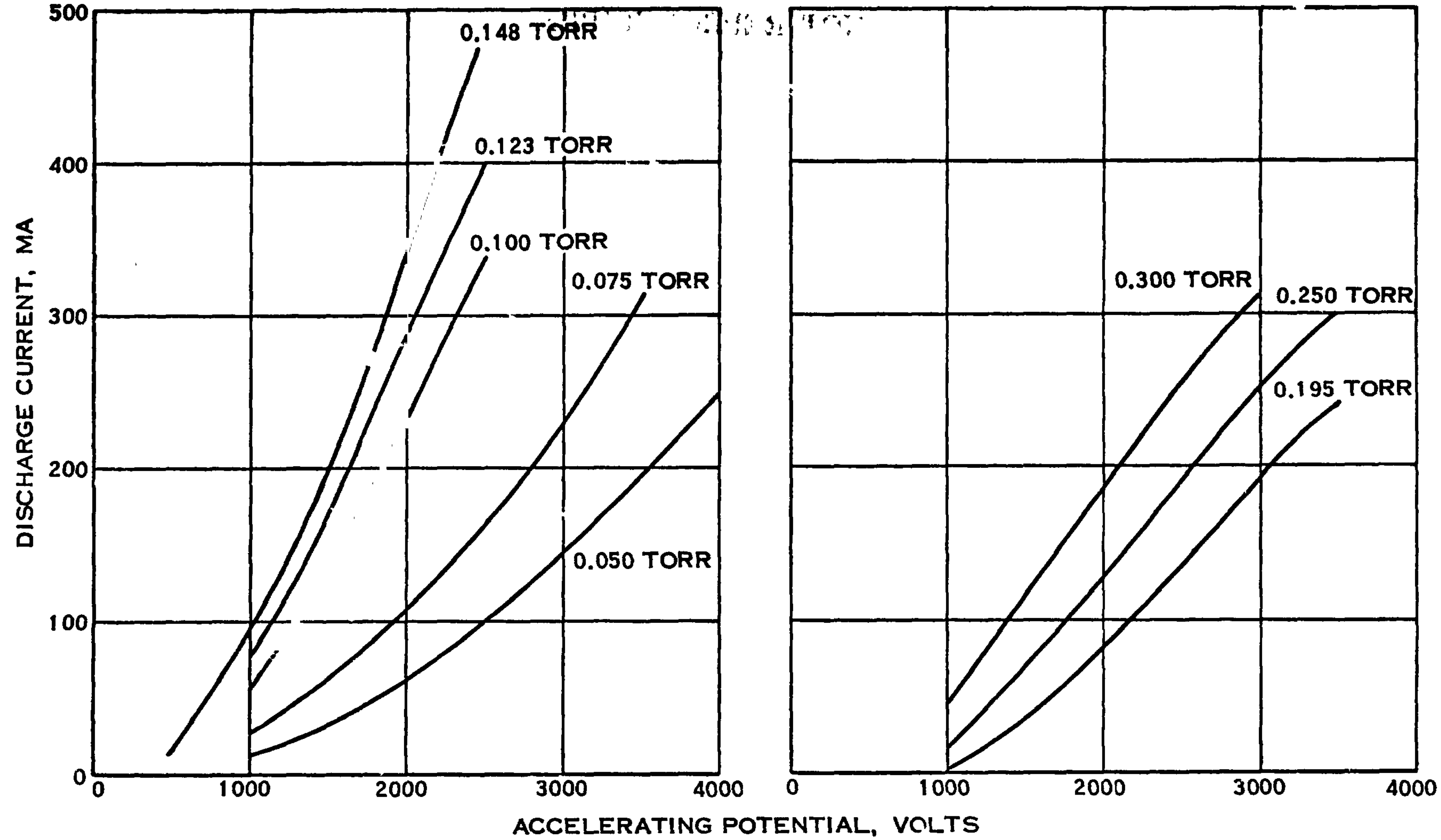
A. WELDED IN AIR. NOTE INCOMPLETE JOINT AT RIGHT SIDE 15 X

B. WELDED IN AIR. NOTE SHALLOW MELTED REGION 250 X



C. WELDED IN HELIUM. NOTE JOINT TOWARD LEFT SIDE 15 X

FIGURE 8. EFFECT OF DIFFERENT GASES ON WELDS IN STAINLESS STEEL TUBING



A. COMPRESSED AIR AMBIENT GAS   B. HELIUM AMBIENT GAS

FIGURE 9. EFFECT OF GAS SPECIES AND PRESSURE ON OPERATING CHARACTERISTICS OF THE ANNULAR TOOL

Thus, neither air nor nitrogen are satisfactory welding atmospheres. Helium is less satisfactory than argon because of the high pressure required for proper beam formation. Therefore, argon was chosen as the preferred environment.

Low Power, High Speed

A series of samples of titanium and stainless steel were welded using minimum power and minimum time to determine the effect of power on welding speed. These data were compared with other samples, and the power was plotted against the time to make the weld, as shown in figure 10. These data are represented by a scatter band because of the variables affecting precise measurement of welding time:

a. The time at which welding was deemed complete was based solely upon operator judgement.

b. The time required to raise the power to the desired weld level and to turn it off was controlled manually, and therefore was not reproducible.

Referring to figure 10 indicates that, to some limiting value, the time required to make a weld is inversely proportional to the power. It is apparent that a wide range of power and time can be used to make satisfactory weld joints. If the data used to plot figure 10 are examined in detail it can be seen that the energy (the product of power times time) to make a weld reduces as time diminishes. In a space flight when power is at a premium, it is desirable to use high power to minimize welding time.

Effect of Material Condition

The fourth evaluation involved making welds in stainless steel and titanium tubing in conditions which are representative of those found in actual structures. The two alloys were welded in three different conditions: oxidized, painted, and greased. All welds appeared to be sound; however, detailed tests were not performed to examine for brittleness, formation of non-metallic inclusions, or for the presence of excessive interstitial alloying elements.

The surface contaminants were accomplished by:

a. Heating, in air, to 750F for 1 hour, or

b. Painting one-half of the tube surface with black acrylic spray paint, or

c. Coating one-half of the tube surface with silicone vacuum grease.

Welds in titanium and stainless steel tubes oxidized prior to welding are shown in figure 11. The outward appearance of these welds was similar to those welds made in tubing that had not been oxidized. The surface of the titanium tubing had a tannish-brown oxide which was caused by heating; this appeared to have been removed by welding.

Samples which were painted and greased before welding are shown in figure 12. The greased titanium sample on the left of figure 12 had the grease burned off the weld zone and an area adjacent to it. It is assumed that this occurred before fusion. The grease seems to have been completely removed from the surface of the tube over the entire region of beam impingement, and not only at the region of melting. The paint shown on the sample at the right in

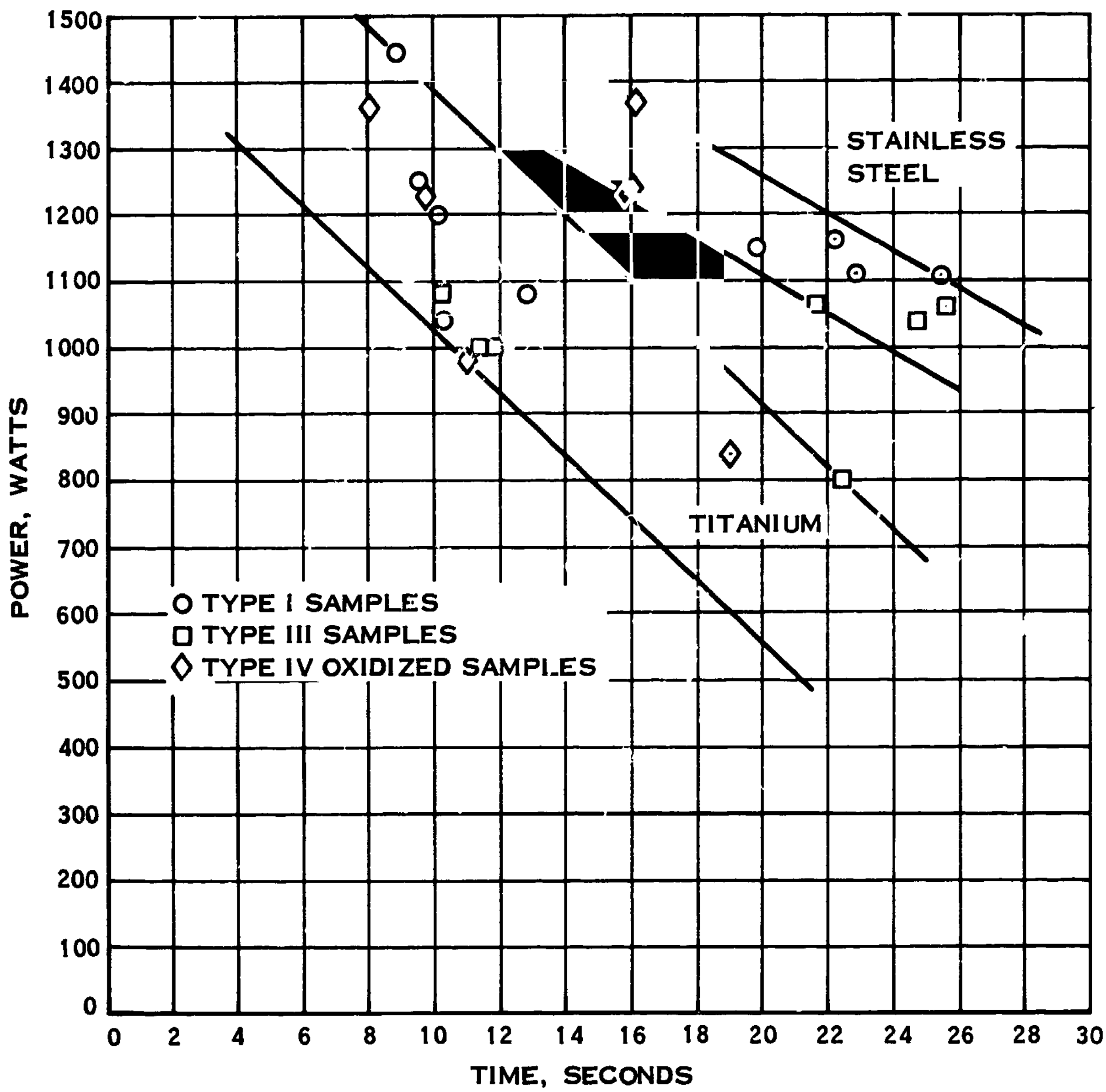


FIGURE 10. EFFECT OF POWER ON TIME TO MAKE ACCEPTABLE WELDS IN TITANIUM AND STAINLESS STEEL

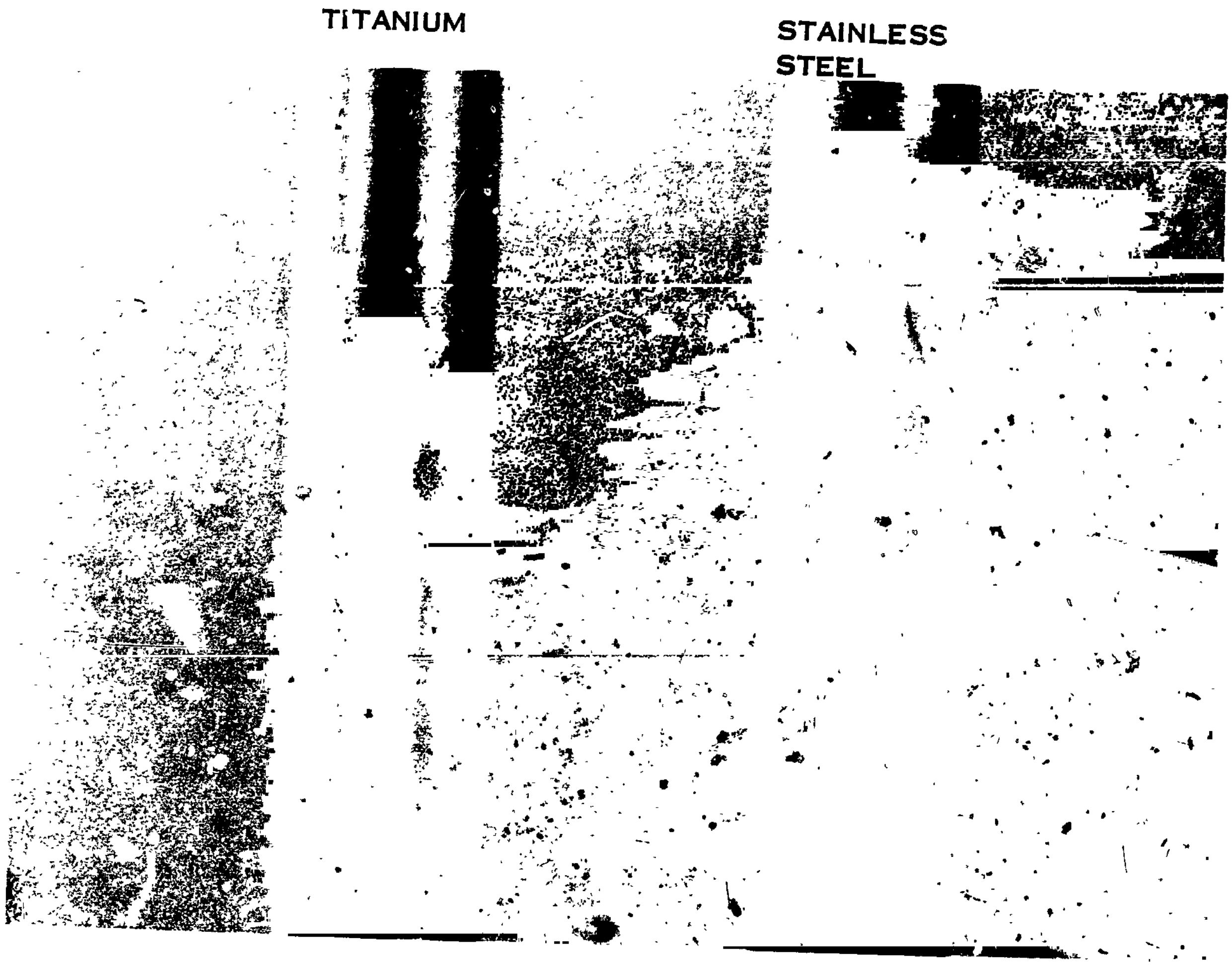


FIGURE 11. STAINLESS STEEL AND TITANIUM SAMPLES OXIDIZED IN AIR AT 750F PRIOR TO WELDING

FIGURE 12. TITANIUM AND STAINLESS STEEL TUBES, GREASED AND PAINTED PRIOR TO WELDING

figure 12 responded in a manner similar to the grease.

Linear Tool

The weld tests with the linear tool were performed to study the usefulness of the tool for linear welding. Weld tests on 0.031-inch and 0.062-inch stainless steel in butt weld configuration were undertaken.

Butt Welds

The technique employed in making welds with the linear cathode, exposing an extended length of weld joint to the beam, could not be used to produce acceptable welds without further development. This probably was the result of small differences in power across the active length of the cathode. One part of the joint started to melt first, and by the time the entire joint had been heated to fusion temperature, the molten metal in the first region to melt had balled up, producing a gap at the butt. This gap prevented the formation of a good weld over the full joint length.

This hypothesis was proved by using the cathode to make a very short weld. The length of the weld, diminished by closing the shields, was small enough to reduce the beam to almost a point impingement, but large enough so that a short line of weld would be discernible. Such welds are shown in figure 13. The upper, 0.031-inch thick, specimen was welded with a cathode having an effective length of 0.300 inch; the lower specimen, 0.062-inch thick, required a larger, 1.250 inch, effective cathode length. The thicker material required approximately 2kw of beam power to produce the full penetration.

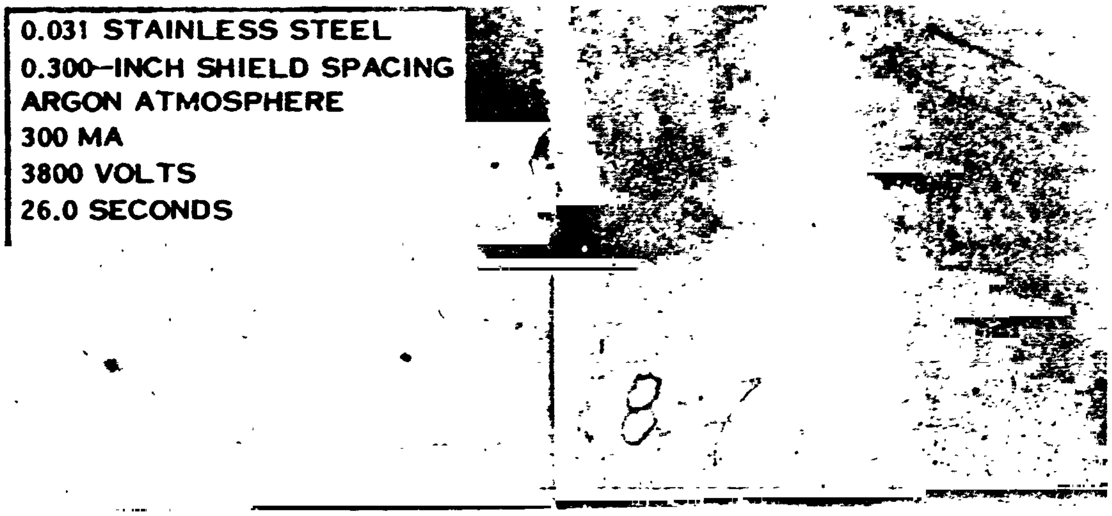
Brazing

The scope of the program limited the metals joining tests to welding; however, a test was performed early in the tool evaluation to determine its applicability to brazing. The tool, using a linear cathode, was used to prepare a lap braze joint. Two sheets of copper were lapped over each other with an AWS type BCuP brazing alloy sandwiched between them. The assembly was heated to the liquation point and an apparently good brazed joint was produced.

CONCLUSIONS

The annular materials joining tool can be used to butt weld tubing to 3/8-inch O.D., 0.060-inch wall, and wire and rod from 0.030 to 0.125-inch diameter. Effective use of the linear materials joining tool was limited to spot welding in 0.030- and 0.060-inch stainless steel. Stainless steel, copper and titanium were successfully welded; however difficulties were encountered in welding aluminum. A hypothesis has been presented explaining that this difficulty results from the rapid rate at which the aluminum surface is oxidized. As a consequence, it is anticipated that aluminum is weldable by the materials joining tool in the "high-purity" vacuum of space.

The tools require a low pressure gaseous atmosphere for beam formation. Of the four gases tested during this program, argon is preferred for welding, however, any of the gases tested can be used to produce an electron beam by this method.



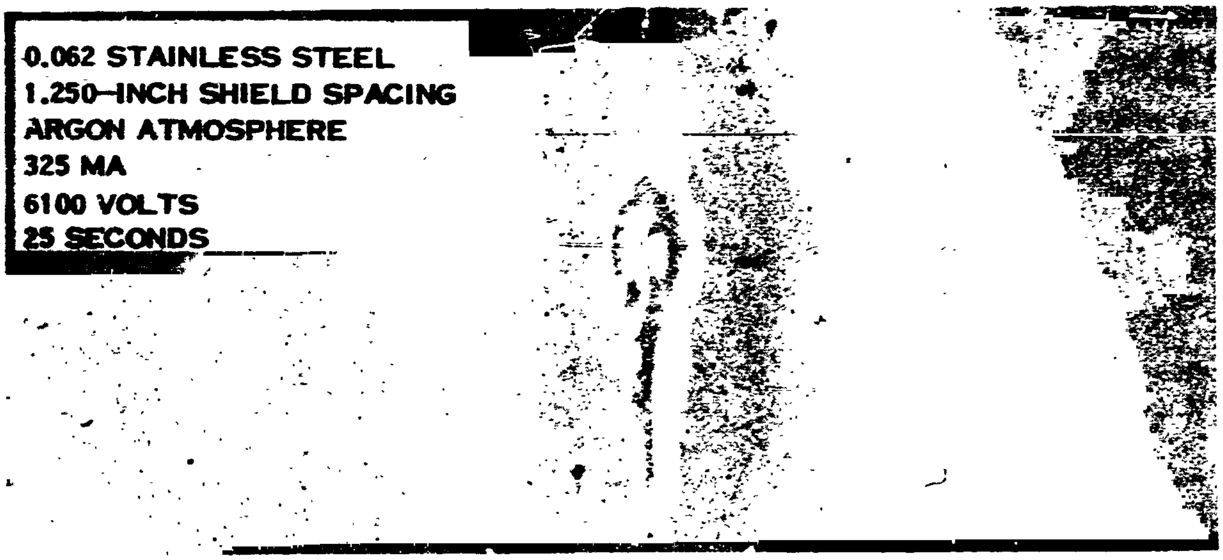


FIGURE 13. TACK WELDS MADE WITH THE LINEAR TOOL

EVA FORCE EMISSION CAPABILITY IN SIMULATED ZERO GRAVITY*

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SUMMARY: This paper describes the results of an experiment designed to evaluate and quantify man's ability to generate impulsive and sustained forces in an extravehicular, zero-gravity environment. The work was performed by the General Electric Company as part of a study for the collection of human engineering data for the maintenance and repair of advanced space systems which was sponsored by OART at NASA Headquarters and directed by the P&VE Laboratory at the Marshall Space Flight Center.

INTRODUCTION

One of the most basic demands to be made on man by space systems, present and future, is the requirement to exert forces of various types and directions. The need to remove and stow, assemble and disassemble, and install various structural components, as well as the need to move himself will require the applications of forces by the space-suited astronaut. The experiment described here is designed to evaluate and quantify man's ability to generate impulsive and sustained forces under a variety of conditions which simulate various modes of restraint and accessibility. The resultant data is of special importance to spacecraft designers, since it provides the answers to two essential questions:

a. Given a force generation requirement, what are the accessibility and restraint criteria which must be imposed on the workspace envelope?

b. Given a specific workspace envelope and restraint, what are the force generation capabilities of a space-suited astronaut?

OBJECTIVES

The systematic variation of restraint conditions while measuring maximum impulse and sustained force generation capability will provide the spacecraft designer with comparative data on the relative values of specific types of restraint systems. Varying the orientation and location of the force receiver will also provide

*The research reported in this paper was carried out under Contract No. NAS 8-18117, for the NASA Marshall Space Flight Center.

comparative data to evaluate the relative effects of accessibility and variations of the work envelope on man's force application capabilities. Although many of the restraint conditions to be tested would appear unreasonable in certain situations, this experiment is designed to generate sufficient information to assist the designer in specifying and designing new and better restraint systems than those presently available. The design of an optimum restraint when a desired force emission capability is required will be possible on a quantitative basis if the appropriate data is available. Also, since the astronaut will be provided with a restraint system which controls and limits his movements, the availability of force emission capability data as a function of force receiver location and orientation will assist the designer in the solution of the man/machine interface problems. Therefore, the major objectives of this experiment were to:

a. Measure and evaluate the effects of restraint system on impulse and sustained force-producing capability in zero-g

b. Measure and evaluate the effects of force receiver orientation on impulse and sustained force-producing capability in zero-g

c. Measure and evaluate the effects of force receiver location on impulse and sustained force-producing capability in zero-g

EXPERIMENT DESCRIPTION

This experiment was concerned with determining the effects of zero-gravity on the force-producing capabilities of subjects as a function of the type of restraint and simulated conditions of accessibility. In this study, the restraints were varied in the number of energy sinks provided to the subject and the location of these energy sinks. Additionally, the accessibility conditions were evaluated by changing the location and orientation of the force receiver relative to the subject. The subjects performed all tasks wearing an Apollo State-of-the-Art spacesuit pressurized to 3.7 psig. Zero-gravity was simulated by the technique of neutral buoyancy submergence described in a paper entitled A Method for Obtaining High Fidelity Underwater Simulation of Man Space Activities by Goldstein and Alvarado and presented at AIAA 4th Annual Meeting and Technical Display, Anaheim, California, October 23-27, 1967. The experimental apparatus was designed and constructed to provide efficient selection of the experimental condition combinations by an underwater technician. The experimental condition combinations consisted of eight types of restraint (including no restraint), three force receiver distances, three force receiver angles, and two handle orientations. Maximum impulse and sustained forces were obtained from each of four subjects for each experimental condition.

Impulsive forces were defined as the peak forces exerted during a 1.0-second interval, while sustained forces are defined as the minimum force maintained over a 4-second interval. The required forces were applied in push, pull, left, right, up, and down directions at all force receiver locations.

Prior to the initiation of each experimental sequence, the subject was attached to one of the restraint systems and stabilized in front of the force receiver handle. The handle had been previously set at one of the experimental distances and angles and at a selected orientation. When all personnel were ready, the experimenter initiated signals on a test director's panel which displayed on the subject's cue panel the required direction and type of force to be exerted. After a 2-second cue time a "go" signal was displayed to the subject, who was instructed to exert the appropriate force until the "go" signal extinguished. After a suitable rest period, new cue signals were displayed to the subject and the above procedure repeated. After performing 12 trials or required force exertions (sustained and impulse forces in all six directions), the handle orientation and/or distance was changed and a new sequence of 12 trials begun. An experimental session consisted of 96 trials, and the experiment required 192 sessions to complete the data collection across all experimental conditions.

EXPERIMENTAL VARIABLES

Man's ability to emit forces in a zero gravity environment is influenced by several variables. Some of the most important are: (a) type of restraint system; (b) force profile required; (c) position and location of the body relative to the force receiver; (d) orientation of force receiver; and (e) type of space suit worn and pressurization conditions. Variations and combinations of the above have been included in the experiment protocol to the extent limited by practical, budgetary, and equipment considerations. The range of each variable is briefly discussed below.

Selection of restraints to be used in this experimental program was made based on the feasibility and probability of being available for future manned space flights. Consideration was given to factors which influence the crew performance profile, such as the number and location of attachment points, rigidity of energy sinks, and freedom of movement. Selected restraints include:

a. None

b. Handhold

c. Two-point waist strap

d. Gemini-type Dutch shoes

e. Handhold and waist

f. Handhold and shoes

g. Waist and shoes

h. Handhold and waist and shoes

The type and direction of the applied forces were chosen to obtain data covering all directions of force application. Directions of application include: (a) push; (b) pull; (c) left; (d) right; (e) up; and (f) down. The left, right, up, and down directions also have direct applicability to torque generation. Both 1-second impulsive and 4-second sustained forces are included for each direction.

Although an almost infinite number of force receiver locations are possible in the volume enclosed by the subject's reach envelope, it was necessary to limit this variable to the plane described by the horizontal sweep of the man's arm and his reach. Utilizing a line drawn perpendicular to the right side of the chest as the 0-degree reference point, two additional locations at +45 degrees (right) and -15 degrees (left) were selected. At each of these locations, 3 distances forward of the sagittal axis were chosen to sample the range in variations in force-producing capabilities and include:

a. Near (elbow angle approximately 90 degrees)

b. Medium (elbow angle approximately 135 degrees)

c. Far (elbow angle approximately 180 degrees)

The handhold of the force receiver was oriented either in the horizontal plane (0 degrees) or vertical plane (90 degrees). As previously stated, Apollo State-of-the-Art suits are utilized in the conduct of this experiment.

Six subjects were used in this experiment, four extensively and two as alternates. Three of the subjects were selected to represent the 50th percentile and three the 90th percentile in height from the <u>Anthropometry of Flying Personnel</u> published in 1950 by Hertzberg, Daniels and Churchill. The actual subject descriptive data is presented in Table I. They ranged from 25 to 29 years in age (mean of 31), from 140 to 178 pounds in weight (mean of 163), and from 5'10" to 6'0" in height (mean of 5'11"). All subjects had high school diplomas or equivalent, one had 2 years of college and three had college degrees in engineering. All subjects were experienced SCUBA divers and had been pressure suit indoctrinated. All subjects had passed the Air Force Category III flight physical and had normal vision in both eyes.

TABLE 1 SUBJECT DATA

Sub. No.	Age	Wt. (Lbs.)	Ht. (In.)	Educ. (Yrs.)
1	37	170	72.55	16
2	33	178	71.50	14
3	39	150	70	12
4	30	140	69	16
5	25	165	72	12
6	25	178	70	16

Pressurization Method

Two methods of providing the required 3.7 psig pressure inside the space suits were utilized. The first was provided by an air pressurization system attached to an underwater backpack. Inlet and exhaust hoses were attached to the suit to provide air flow for cooling and CO_2 removal. The second was provided by a water pressurization system located in the underwater backpack. This configuration provided continuously pumped water into the suit through a single umbilical and dumped it through preset, parallel dump valves. The two pressurization methods were included as an experimental variable to provide an initial determination of the differential effects, if any, of the two pressurization modes.

EXPERIMENT APPARATUS

The underwater experiment apparatus (Figure 1) consisted of a force receiver that converted the forces applied by the subject into electrical output signals, a framework to support the force receiver and provide the proper restraints to the test subject, and a panel to display to the test subject the desired force direction and type. In addition to this equipment, a panel was provided to enable the test director to give instruction to the test subject in the water.

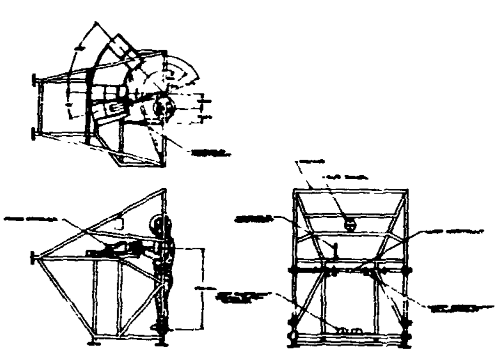

FIG. 1 - EXPT. 84A UNDERWATER APPARATUS

The force receiver hardware consists of a cylindrical shaft, anchored at one end with the handle affixed to the other end. Force application is measured by bending deflections in the cantilevered shaft in the Y and Z axes and deflections of a calibrated spring in the X axis. Deflection is measured by three orthogonally placed differential transformers and is relatively small (approximately 0.200 in./100 lb in the X axis and 0.250 in./100 lb in the Y and Z axes). The force receiver is mounted on a carriage which may be adjusted in the Y-Z and X-Z planes to vary elbow angle and horizontal locations, respectively. The carriage is mounted on a rigid frame which also provides attachment points for the subject to the various restraints.

During the conduct of the experiment, the test subject is attached to one of the eight restraints, and the appropriate experimental conditions are set up by a technician who remains in the water with the test subject. Instructions as to force direction and type are programmed by the test director through his control panel and received by the underwater test subject on his cue panel for a 2-second cue period prior to the force application command. The type of force to be applied is denoted by appropriate illumination of either the "impulse" or "sustain" legends, while the direction of force application is shown by illumination of the appropriate arrow (left, right, up, or down) or legend (push or pull). At the end of the cue interval, a "go" light is illuminated for a period of 4 seconds for the sustained force command and 1 second for an impulsive force. The test subject applies maximum force for the duration of the "go" signal.

EXPERIMENT SCHEDULING

The magnitude of the number of experimental conditions selected for investigation in this experiment required that great care be exercised in the scheduling and organization of the test sequences in order to minimize the possibility of systematically biasing the resultant data. The experiment was originally designed for 36 operational days with 768 trials on each day. The four subjects were to be tested in each of the 3456 experimental condition combinations twice, making a total of 27,648 data points or trials. The randomization of the variables and the required schedule revisions are discussed below.

In order to preclude the occurrence of such extraneous or systematic biases as transfer-of-training and order-of-presentation effects on the reliability of the data, it would be desirable to randomize the sequence of all the experimental condition combinations. Numerous practical considerations, however, made complete randomization impractical, as a relatively indeterminant number of changes in subject, suit pressurization, restraint type, etc., would extend the schedule beyond all reasonable bounds.

In actuality the condition combinations were arranged into sessions with 96 force applications or trials each, resulting in a total of 288 sessions. A working schedule of eight sessions a day, with each subject participating in two sessions each day, was planned. The following were the constraints placed on the randomizing, or scheduling, of the experiment:

a. Pressurization method remained constant throughout a day and was alternated each working day.

b. The order of the four subjects was random for the first four sessions and repeated for the second four sessions each day.

c. The receiver angle remained constant within each session, but was random across sessions.

d. Restraints remained constant within each half session (48 trials), but were randomly assigned across half sessions.

e. Receiver orientation and receiver distance remained constant within each block of 12 trials, but were randomly assigned across blocks of 12 trials.

f. Within each block of 12 trials, every combination of force type and force direction occurred. The order of presentation was random within each block of 12 with the restriction that within each four trials (i.e., 1 through 4, 5 through 8, and 9 through 12), two sustained and two impulse trials occurred.

Rest periods were systematically distributed throughout the session to minimize fatigue effects.

The original experimental schedule was revised in a number of ways due to operational constraints and problems. These primarily resulted in a change in the running order of subjects and the number of replications of each trial point. The completely random order of test subjects was restricted such that two subjects (1 and 2) were scheduled first during each day because of their availability only from 7:00 A.M. to 3:30 P.M. The other two subjects (3 and 4) always ran last because of their availability from 9:30 A.M. to 6:00 P.M. In addition, the limited availability of Apollo State-of-the-Art pressure suits required that the two 90th percentile subjects (1 and 3) alternate with the two 50th percentile subjects (2 and 4) to reduce session changeover time. These revisions resulted in only two possible subject running orders - 1,2,3,4, or 2,1,4,3. It was decided to use one running order for 2 days, then alternate with the other running order for two days. This schedule was generally kept for the first four sessions of each day.

The original protocol design called for two replications of each experimental condition combination for a total 288 sessions. Due to the considerable number of delays resulting from equipment failures and personnel problems, it was decided to terminate the experiment after obtaining only one replication of each condition combination and evaluate the feasibility of pooling some of the experimental conditions, specifically pressurization method and subjects.

Combining the data collected under the conditions of water and air pressurization ...s appeared to be justified on the basis of face validity from the test directors constant visual monitoring of the oscillographic tapes. Additionally, it was felt that the subjects percentile groupings could be combined. This pooling of the subject factor also appeared, from "eye-balling" the data, to be justified on the basis of the subjects not representing a 50th and 90th percentile grouping on the basis of force emission capability although they did represent the 50th (or 65th) and 90th percentile in height. Subsequent statistical analysis, presented in the Results and Conclusions Section, confirmed the validity of these decisions.

When the above decision to combine experimental conditions was made, it was found that 84 percent of the data for the first replication had been collected. In addition, approximately 14 percent of the data for the second replication had been collected. However, for the remaining portion of the data, a revised session format would be required. In the new session format, restraint would remain constant for only 24 trials as compared to 48 new trials in the original sessions. Otherwise the session format remained unchanged. Approximately 14 percent of the data was collected under this new session format.

Finally, when the computer reduction of the data was completed, it was found that some data was missing due to instrumentation problems or excessive noise on the analog tape. This missing data, approximately 2 percent, was collected during makeup sessions using approximately the same session format with more frequent changes of the experimental variables. The deletion of individual trials, for one reason or another, combined with the variation in the number of replications discussed above, resulted in a variation from 6 to 12 in the number of trials for each experimental condition combination in the data printout.

INSTRUMENTATION

The instrumentation utilized in this experiment was designed to provide for both computerized data reduction as well as the capability for real-time monitoring of the data by the test director. These capabilities were provided by the use of magnetic tape and oscillographic recordings of both the instructions to the test subject and the outputs of the force receiver transducers. In addition to recording the applied force in the command direction, the forces in the other two axes were also recorded to provide a measure of the error forces. Figure 2 is a block diagram of the instrumentation used.

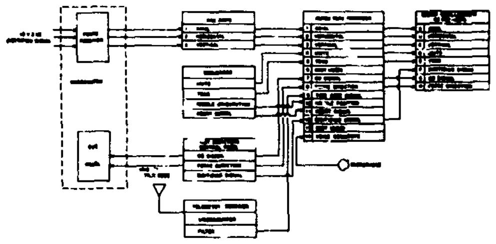

FIG. 2 - INSTRUMENTATION BLOCK DIAGRAM

The outputs of the force receiver transducers were recorded on Channels 1 through 3 of an Ampex Model CP-100 recorder, with the remaining channels used to record identifying information for the computer and verbal comments made by the test director. Channels 4 and 5 contain the trial identification numbers as listed on the session format sheet. The "go" signal, which was recorded on Channel 7, was a 4-second or 1-second full-scale deflection that served to indicate a commanded force as sustained or impulsive, respectively. Channel 8 identified the command force direction by coding as one of six discrete voltage levels, ranging from zero to full scale. Channels 9 and 10 recorded the IRIG "B" time code and the force receiver handle orientation.

The abort signal recorded on Channel 11 was used to indicate to the computer that a particular trial should be discarded from the data processing. This was done to prevent erroneous data from being incorporated in the data output and to save unnecessary computation time. A digitizing signal (Channel 12) served as a command signal for the A-D conversion beginning 2 seconds before the "go" signal and ending 1 second after the termination of the "go" signal. The data was later digitized during this 7-second (for sustained forces) or 4-second (for impulsive forces) time period at a rate of 100 samples per second. Channel 14 was used to record verbal comments by the test director. Channels 6 and 13 were unused spares.

Real-time viewing of the data was provided by an eight-channel oscillographic recorder. The inputs to this recorder were obtained from selected playback heads of the analog tape recorder so that the test director could continuously monitor the status of pertinent recorded data. This was especially important in order to detect zero-level shifts in recording channels.

A key element in the instrumentation system was the force transducers. These were the linear motion differential transformers, electromechanically proportional to the displacement of a movable core. The output of the transducer had infinite resolution over its specified range as displacement of the movable core on either side of a null point produced an increasing voltage directly proportional to the distance moved.

Calibration of the instrumentation system was performed each day prior to the start of the first experiment session. This consisted of attaching accurate weights to a holder which was attached to the force receiver handle. Weights were added in 5-pound increments to 20 pounds, and then 10-pound increments to 80 pounds. The calibration was performed for all three axes. During the calibration, the excitation voltage of each transducer was adjusted so that, after amplification, the output at full scale deflection was 1.0 volts and was linear over the full range.

DATA REDUCTION

The data collection and recording system was described in the previous section. The important output of this system is the analog tape which contains all force data as well as identifiers of particular session conditions. These tapes formed the input to the data reduction system, as illustrated in Figure 3.

In the analog-digital conversion process, the following channels were digitized:

a. Axial force data

b. Horizontal/vertical* force data

c. Vertical/horizontal* force data

d. Trial number, units

e. Trial number, tens

f. Go signal

g. Force direction

h. Force receiver orientation

i. Abort signal

Digitizing was performed by command of the 7-second (for sustained forces) and 4-second (for impulsive forces) digitizing pulse on Channel 12. A sampling rate of 400 samples per second was used; however, since the analog tapes were played at a speed 4 times greater than the

*Depends on force receiver handle orientation.

record speed, the effective sampling rate was 100 samples per

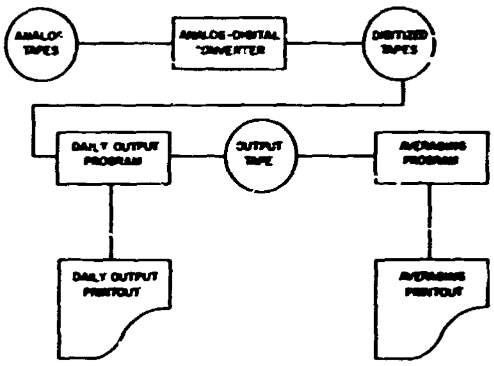

FIG. 3 - DATA REDUCTION BLOCK DIAGRAM

second. Low-pass filters were applied to all channels, except Channel 10, to eliminate as much noise as possible from the analog tapes. The filters had a cutoff frequency of 100 cycles per second, but the effective cutoff was at 25 cycles per second due to the 4-to-1 ratio of playback speed to record speed. Neither the sampling rate nor the filter cutoff frequency created any problem with respect to loss of data.

The digitized tapes then formed the input to the first of two computer programs, known as the Daily Output Program. The objective of this program was to provide a printout of the force data recorded for every trial in the experiment. The program consisted of the following sequential operations:

a. The digitized data was numerically converted into forces in pounds.

b. "Header cards", containing identification data not included on the tapes, (day, session, subject, pressurization method, receiver angle, and receiver distance), were read into the computer and combined with the proper data;

c. Data output was in blocks of 12 trials, with a complete set of identifiers.

Possible error messages were also printed out where applicable, and an editing feature was available for use where necessary to correct for errors. The identifiers and appropriate definitions used in the program are presented below:

For sustained forces:

a. In the correct (commanded) force direction:

   1. MAX is the largest force during the entire 4-second period the GO light is on.

   2. MIN is the smallest force encountered during the last 3 seconds of the GO-signal <u>unless</u> the force changes sign

(i.e., goes from positive to negative or vice-versa), in which case the minimum force is defined as zero.

3. FIN is the force at GO-signal cutoff.

b. In the error axes:

1. MAX is the largest force in either direction of an axis during the middle 3 seconds of GO-light.

2. MIN is the smallest force in either direction of an axis during the middle 3 seconds of GO-light.

For impulsive forces:

a. In the correct (commanded) force direction:

1. MAX is the largest force that occurs for the 1 second that the GO-light is on and 1 second afterwards; i.e., for a 2-second period.

2. MIN is the smallest force which occurs during this period.

3. FIN is the force at GO-signal cutoff.

b. In the error axes:

1. MAX is the largest force in either direction of an axis while the GO-light is on.

2. MIN is the smallest force in either direction of an axis while the GO-light is on.

After all the data from the experiment had been processed by this program, a composite output tape was prepared. This tape consisted of all the information necessary to identify the conditions for all experimental trials and all the data associated with these trials. This tape then formed the input to the second computer program, the Averaging Program. This program provided the capability to average various combinations of experimental conditions. In particular, the following conditions may be averaged either separately or in any combination:

a. Subject

b. Pressurization method

c. Receiver angle

d. Receiver distance

e. Receiver orientation

f. Restraint

A sample of the printout from the Averaging Program is shown in Figure 4. The asterisks (*) next to Subject and Pressurization Method indicate that these were the common factors for which all data points were considered; i.e., the data listed is an average of all the points which exist for all four subjects utilizing both the water and air suit pressurization methods. The numbers in parentheses to the right of the data indicate the number of trials contained in the average for the particular combination of force type and force direction and of the other specified conditions.

The meaning of "Range" in this data is best illustrated by example. In the first column of data in the lower table on Figure 4, it can be seen that a Sustained-Push force occurred six times under the conditions listed. For each of these trials, the Daily Output Program noted the minimum force which occurred, as previously defined. The Range shows that the minimum force for this condition ranged from 20.22 to 56.09 pounds, with a mean of 39.64 pounds. The interpretation of the remaining columns is similar.

Although the composite tape contains error forces, only the command direction forces are presented here. Furthermore, only the maximum forces in the impulse mode are presented. A graphic presentation of the mean and ranges of all experimental data is given in Appendix A.

RESULTS

The primary results of this experimental program are the means and ranges of the forces exerted under specific combinations of the experimental conditions. These means and ranges were derived from the Averaging Program described in the previous section and are provided in a tabular listing in the final technical report provided under Contract NAS8-18117. This listing, consisting of approximately 100 pages of tabulated information, provides all the design data collected in this study. However, the tabular listing is not the most efficient mode of data presentation for use by designers. The value of the data collected can only be realized by presentation in as efficient and utilitarian manner as possible. The resulting graphical presentation summarized the total data into 12 charts with 6 graphs on a page. These are presented in Figures A-1 through A-12 in Appendix A.

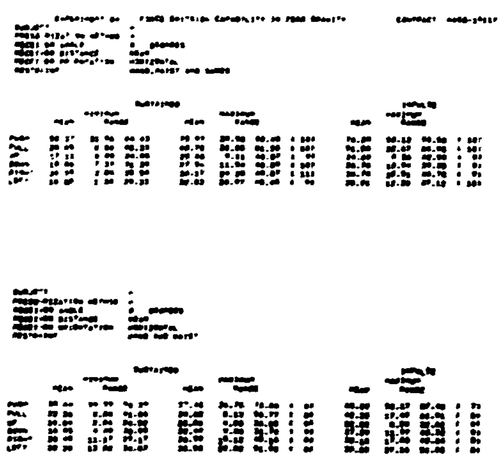

FIG. 4 - AVERAGING PROGRAM SAMPLE PRINTOUT

In addition to the primary design data discussed above, statistical comparisons were made across parameters of the experimental variables to determine the existence and direction of significant relationships. The data used for the statistical comparisons were the overall mean forces determined across experimental conditions and are presented in Table 2.

Nonparametric statistical analyses were considered appropriate for the analysis of these data because of the inability to meet the assumptions concerning the underlying distribution of the population of variables required by parametric analysis. Certain assumptions are also associated with most nonparametric statistical tests; i.e., that the observations are independent and that the variables under study have some underlying continuity, but these assumptions are fewer and more easily met than those for parametric tests. Moreover, most nonparametric tests apply to data in an ordinal scale, and some even apply to data in a nominal scale. The primary advantage of the nonparametric tests is that they can be used when the sample size is small.

Two nonparametric statistical analysis methods were selected for the data analysis. In situations where matched pairs of measures occur in two groups and the measures are in an ordinal scale, Siegel, 1956 recommends the use of the Wilconon Matched-Pairs Signed Ranks Test. This method was utilized to compare the following parameters: overall means across force types (sustain and impulse), mean forces across pressurization methods (air and water), and mean forces across receiver orientations (horizontal and vertical). In situations where K related samples of basically nonparametric measures on at least an ordinal scale are taken, Siegel recommends the Freedman Two-Way Analysis of Variance. This method was utilized to compare the following parameters: mean forces across subjects, mean forces across receiver angles, mean forces across receiver distances, and mean forces across restraints.

TABLE 2 SUMMARY DATA - MEANS ACROSS ALL VARIABLES (IN POUNDS)

Sustained Versus Impulsive Forces

The results of the Wilcoxon Matched Pairs Signed Ranks Test indicate that the sustained mean forces were significantly different from the impulsive mean forces at the 0.01 level of significance. Sustained forces were those force magnitudes that could be maintained over a 4-second interval. Impulsive forces were the peak magnitudes that could be exerted in a 1-second interval.

In general, it can be seen that Push/Pull impulsive force emission capability is approximately 2½ times as great as sustained Push/Pull force capability. Secondly, impulsive force capability in the Up/Down, Right/Left directions is approximately twice as great as sustained force capability in the corresponding directions. Finally, Push/Pull impulsive force emission capability appears to be about twice as great as the Up/Down, Right/Left force capability.

Air Versus Water Pressurization

The results of the Wilcoxon Matched Pairs Signed Ranks Test indicate that the sustained mean forces for air and water pressurization modes did not differ significantly. The data indicate that the impulsive mean forces for air and water pressurization modes also did not differ significantly.

The general conclusions regarding Push/Pull and impulsive over sustained force advantages presented above also apply here.

Horizontal Versus Vertical Handle Orientation

The results of the Wilcoxon Matched Pairs Signed Ranks Test indicate that the sustained mean force for horizontal and vertical handle orientations did not differ significantly.

In general, it appears that handle orientation has little effect on force emission capability in the Push/Pull and Left directions. Also, it appears that a vertical handle orientation increases the capability to exert Right direction forces.

Mean Forces Across Subjects

The results of the Friedman Two-Way Analysis of Variance Test indicate that the sustained mean force emission capability across subjects differed significantly at the 0.05 level. The data indicate that the impulsive mean force emission capability across subjects also differed significantly, but at the 0.01 level. Subjects 1 and 3 corresponded to the 90th percentile grouping and Subjects 2 and 4 corresponded to the 50th percentile groupings on the basis of stature.

In general, however, the force emission capability of the subjects did not follow these percentile groupings. Subjects 2 and 3 generally exerted the greatest mean forces, which indicate a differential force capability within subjects for sustained and impulsive forces. Subject 3 exerted the greatest impulsive forces, but Subject 2 generally exerted the greatest sustained forces.

Mean Forces Across Receiver Angles

The results of the Friedman Two-Way Analysis of Variance Test indicate that the sustained mean force emission capability across receiver angles did not differ significantly. The data indicate that the impulsive mean force emission capability across receiver angles also did not differ significantly.

It appears, however, for both sustained and impulsive forces, that the capability to exert both sustained and impulsive Push/Pull forces increases as the location of the force receiver is moved away from directly in front of the subject. However, this tendency appears to reverse for the other directions. That is, the Up/Down and Right/Left sustained and impulsive force emission capability tends to decrease as the force receiver is moved laterally from in front of the subject.

Mean Forces Across Receiver Distances

The three receiver distances were Near (15 inches), Medium (19 inches) and Far (24 inches) and roughly corresponded to the elbow angles of 90 degrees, 135 degrees, and 180 degrees respectively. The results of the Friedman Two-Way Analysis of Variance Test indicate that the sustained mean force emission capability across receiver distances did not differ significantly. The data also indicate that the impulsive mean force emission capability across receiver distances did not differ significantly.

It appears from the data that as the distance between the subject and the force receiver increases, the ability to exert Push forces decreases. Conversely, as the distance between the subject and the force receiver increases, the ability to exert Pull forces increases. Additionally, there appears to be a lesser tendency for the Up/Down and Right/Left force emission capability to increase as the distance between the subject and force receiver decreases.

Mean Forces Across Restraints

The eight restraint conditions were none (no restraint); handhold; rigid waist; Gemini dutch shoes; the combinations of handhold and waist; handhold and shoes; waist and shoes; and handhold, waist, and shoes. The first four, excluding the no-restraint case, were single-point restraints. The last four were considered as multiple point restraints. The results of the Friedman Two-Way Analysis of Variance Test indicate that the sustained mean force emission capability across restraints differed significantly at the 0.001 level. The data indicate that the impulsive mean force emission capability across restraints also differs significantly at 0.001 level.

In addition to the statistical analysis, the data also appears to indicate the following design implications. It appears that a force cannot be sustained in a no restraint condition. Secondly, the single-point restraints have differential value for difference force directions. For sustained forces, the waist restraint is best for Push/Pull, the Gemini Dutch shoes are best for Up/Down, and the handhold is best for Left directions. In addition, all single-point restraints are about equal in their inability to provide an assist for Right direction forces.

The handhold, waist, and shoes restraint combination resulted in the greatest Push/Pull forces with the waist and shoes combination very close behind. The handhold and shoes restraint combinations resulted in the largest mean sustained for the Up/Down and Right/Left directions. Finally, the data would indicate that Right direction sustained forces should be avoided whenever possible.

For impulsive force emissions, there is very little difference between the means for the single-point restraints, including the no-restraint case. Also, all the multiple restraint conditions are better than the single-point restraints. The handhold and shoes combination permits the greatest impulsive mean force emissions in all six directions. Finally, the handhold and waist is generally the poorest of the multiple point restraint conditions for impulsive force emissions.

CONCLUSIONS

The conclusions resulting from this experimental program are divided into two general groups. In the first are those conclusions that can be drawn from the data analysis and results. The second group contains those conclusions that resulted from the operational experience of conducting an underwater experimental program of such a large magnitude as Experiment 84A.

Data Conclusions

The following major conclusions are summarized from the findings reported above in the analysis and results section.

a. The statistical analyses were performed on means derived across experimental conditions and should not be used to generalize to the individual case. The reader should go directly to the specific condition combination presented in the graphical or tabular format to obtain the pertinent design data.

b. The handhold and shoes restraint combination resulted in the greatest Up/Down and Left/Right sustained and impulsive force generating capability.

c. The handhold, waist, and shoes restraint combination and the waist and shoes restraint combination resulted in the greatest Push/Pull sustained forces.

d. The waist restraint was the only single-point restraint in which a significant sustained (above 10 pounds) mean Push/Pull force could be exerted.

e. The handhold restraint provided the capability for sustaining significant (above 10 pounds) mean forces in only the Left/Right directions.

f. The shoes restraint provided the capability for sustaining significant (above 10 pounds) mean forces in only the Up/Down directions.

g. The mean capability to exert impulsive forces in a no-restraint condition did not differ greatly (4 to 14 pounds differential range) from the capability provided by the single-point restraints (handhold, waist, and shoes restraints).

h. The mean capability to exert impulsive forces did not differ greatly (5 to 12 pounds differential range) across the multiple-restraint conditions (handhold and waist; handhold and shoes; waist and shoes; and handhold, waist, and shoes).

i. The space suit pressurization mode did not differentially affect the ability of subjects to exert forces.

Operational Conclusions

The following operational conclusions were drawn from the considerable number of experiences and observations noted during the conduct of this experimental program:

a. Planning of extensive underwater pressure-suited operations should include a 100 percent contingency time factor.

b. Extreme care should be exercised to insure the cleanliness of the neutral buoyancy facility, especially to minimize the frequency of ear infections.

c. Neutrally buoying space-suited subjects for an upright, nontranslational operation is a relatively simple and easy task.

d. The water pressurization mode was more efficient, from a subject preparation and experimental session changeover time-saving standpoint, than the air pressurization mode.

e. Future water pressurized suit operations should include a face mask that can accommodate a communication system.

f. The possible hazard resulting from the physical reaction of a pressure-suited subject exerting forces under minimal restraint conditions should be carefully considered when selecting restraints for space operations.

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APPENDIX A

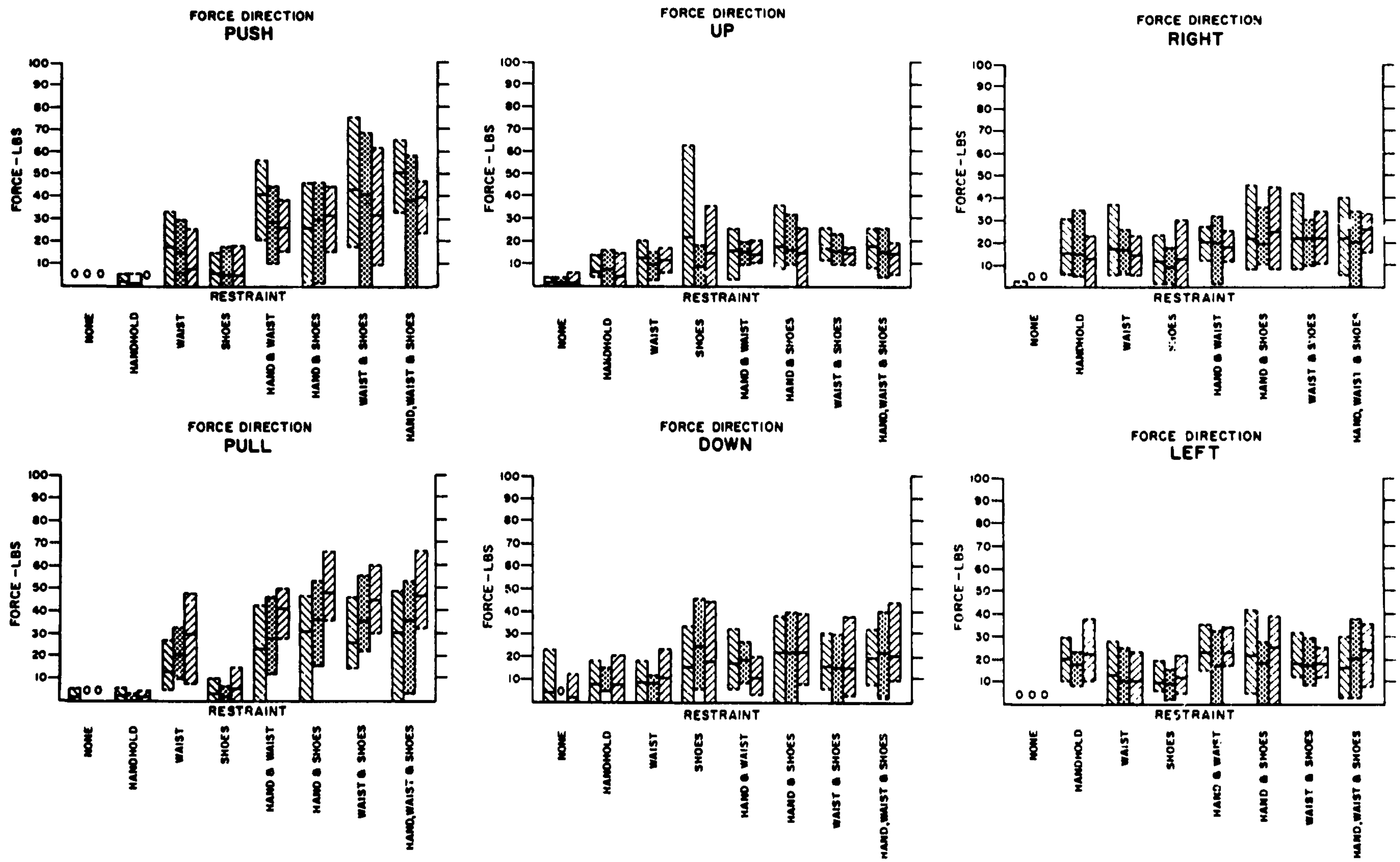
EXPERIMENT 84A - FORCE EMISSION
GRAPHICAL DATA PRESENTATION

In this Appendix are presented the summarized data on astronaut force emission capabilities under simulated zero-gravity conditions. The data are provided on twelve pages with 6 graphs on each page. These are presented in Figures A-1 through A-12. Descriptive data is provided on each page to permit the reader to select the appropriate experimental conditions to answer specific force capability design questions. Table A-1 is a Summary Data Chart Index that specifies the experimental condition combinations included on each page.

Table A-1. Summary Data Chart Index

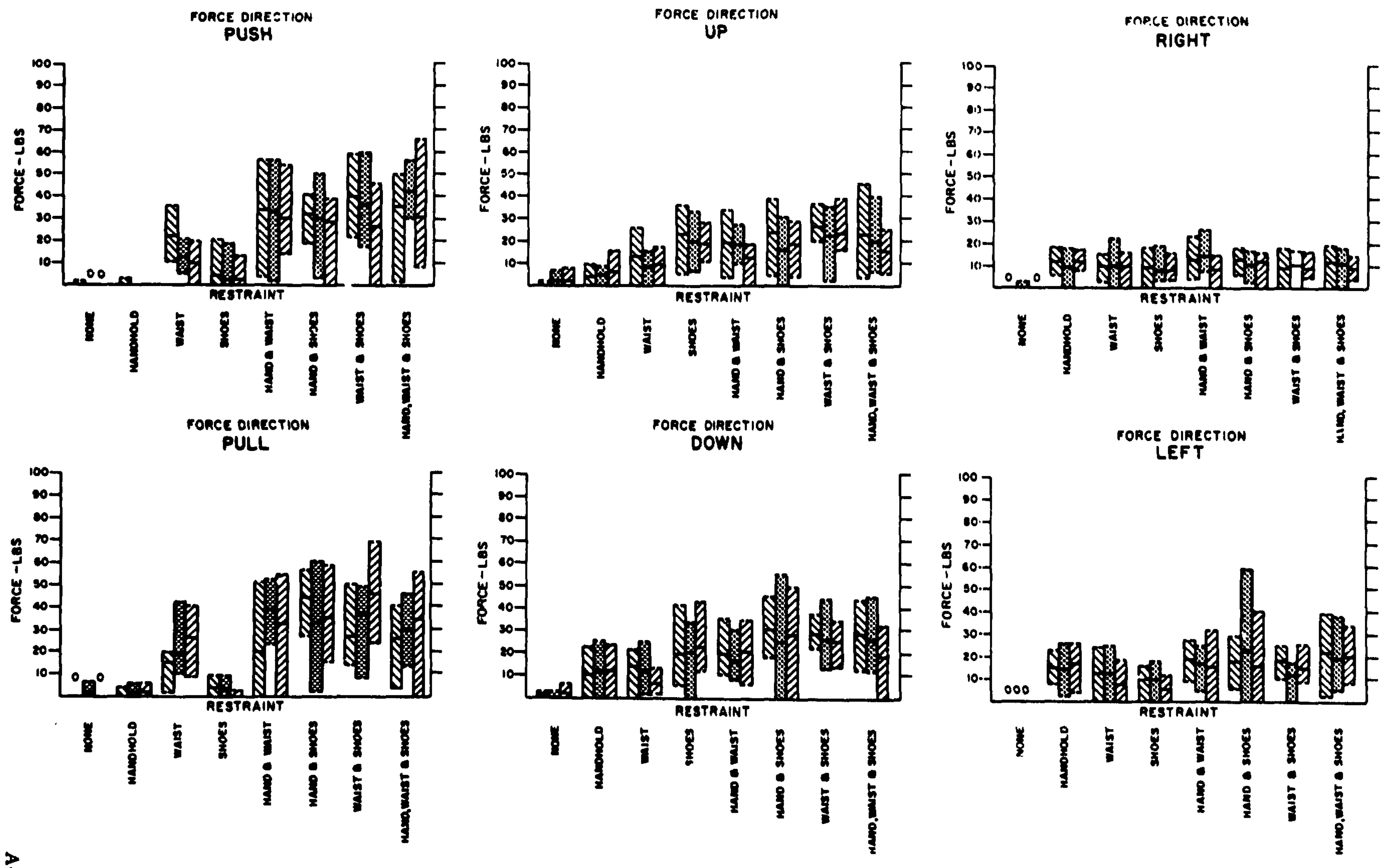
Figure	Title	Force Type (F/T)	Receiver Angle (R/A) (degrees)	Receiver Orientation (R/O)
A-1	Summary Data Chart No. 1	Sustained	0	Horizontal
A-2	Summary Data Chart No. 2	Sustained	0	Vertical
A-3	Summary Data Chart No. 3	Sustained	-15	Horizontal
A-4	Summary Data Chart No. 4	Sustained	-15	Vertical
A-5	Summary Data Chart No. 5	Sustained	45	Horizontal
A-6	Summary Data Chart No. 6	Sustained	45	Vertical
A-7	Summary Data Chart No. 7	Impulse	0	Horizontal
A-8	Summary Data Chart No. 8	Impulse	0	Vertical
A-9	Summary Data Chart No. 9	Impulse	-15	Horizontal
A-10	Summary Data Chart No. 10	Impulse	-15	Vertical
A-11	Summary Data Chart No. 11	Impulse	45	Horizontal
A-12	Summary Data Chart No. 12	Impulse	45	Vertical

FORCE TYPE: **SUSTAIN** RECEIVER ANGLE: 0° HANDLE ORIENTATION: **HORIZONTAL**



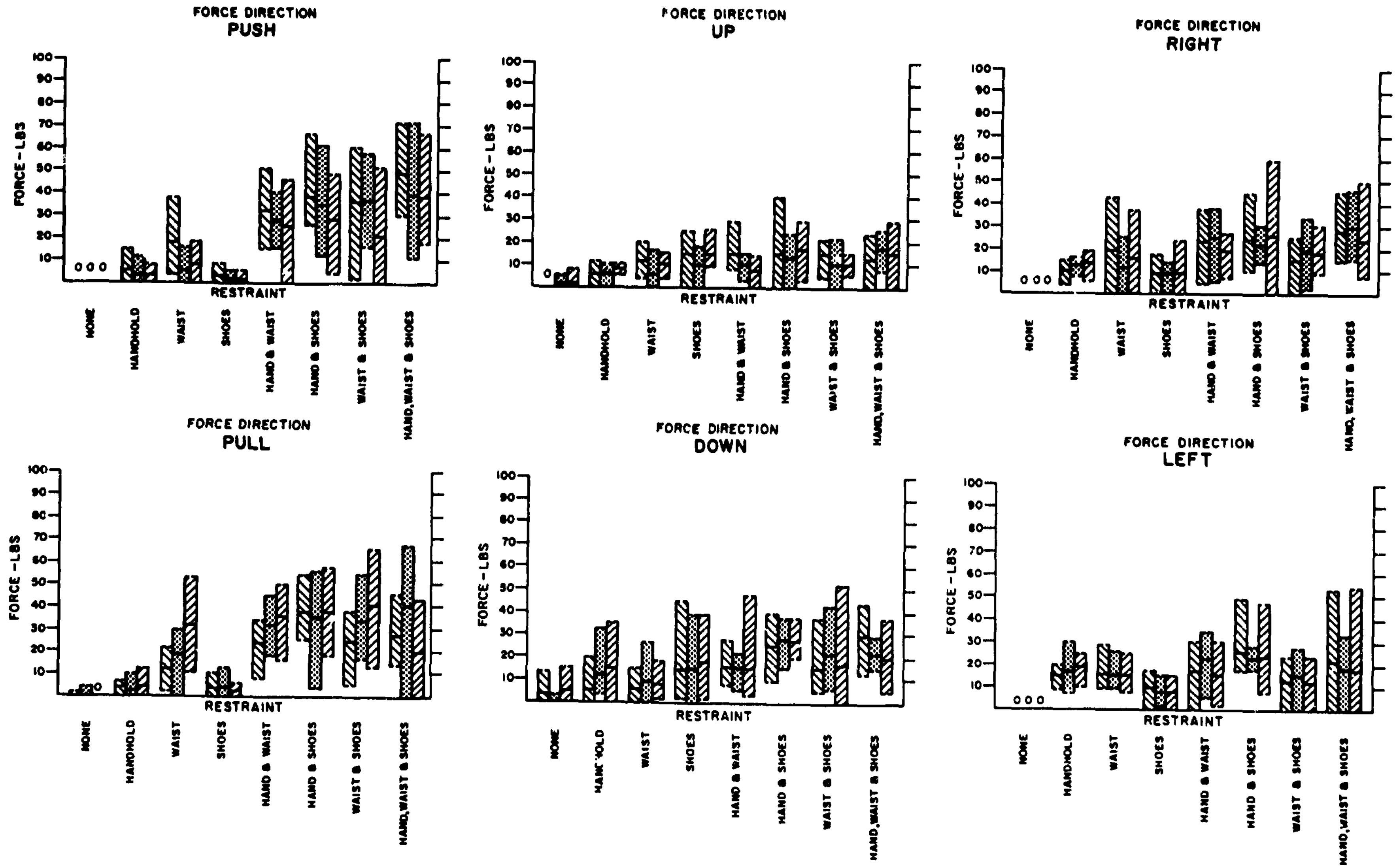
A–1 Summary Data Chart No. 1

FORCE TYPE: **SUSTAIN** RECEIVER ANGLE: **0°** HANDLE ORIENTATION: **VERTICAL**



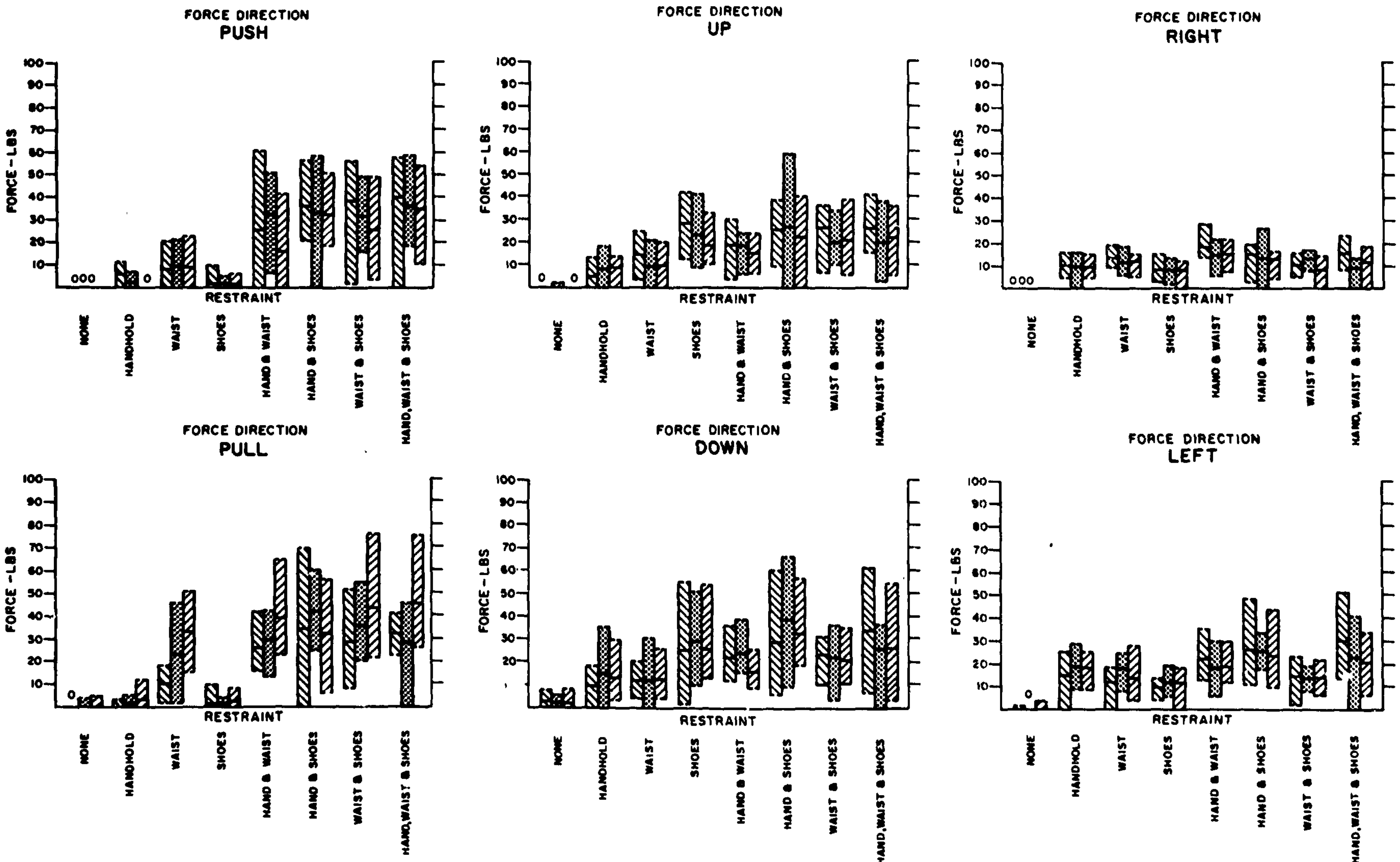
A-2 Summary Data Chart No. 2

FORCE TYPE: **SUSTAIN** RECEIVER ANGLE: -15° HANDLE ORIENTATION: **HORIZONTAL**



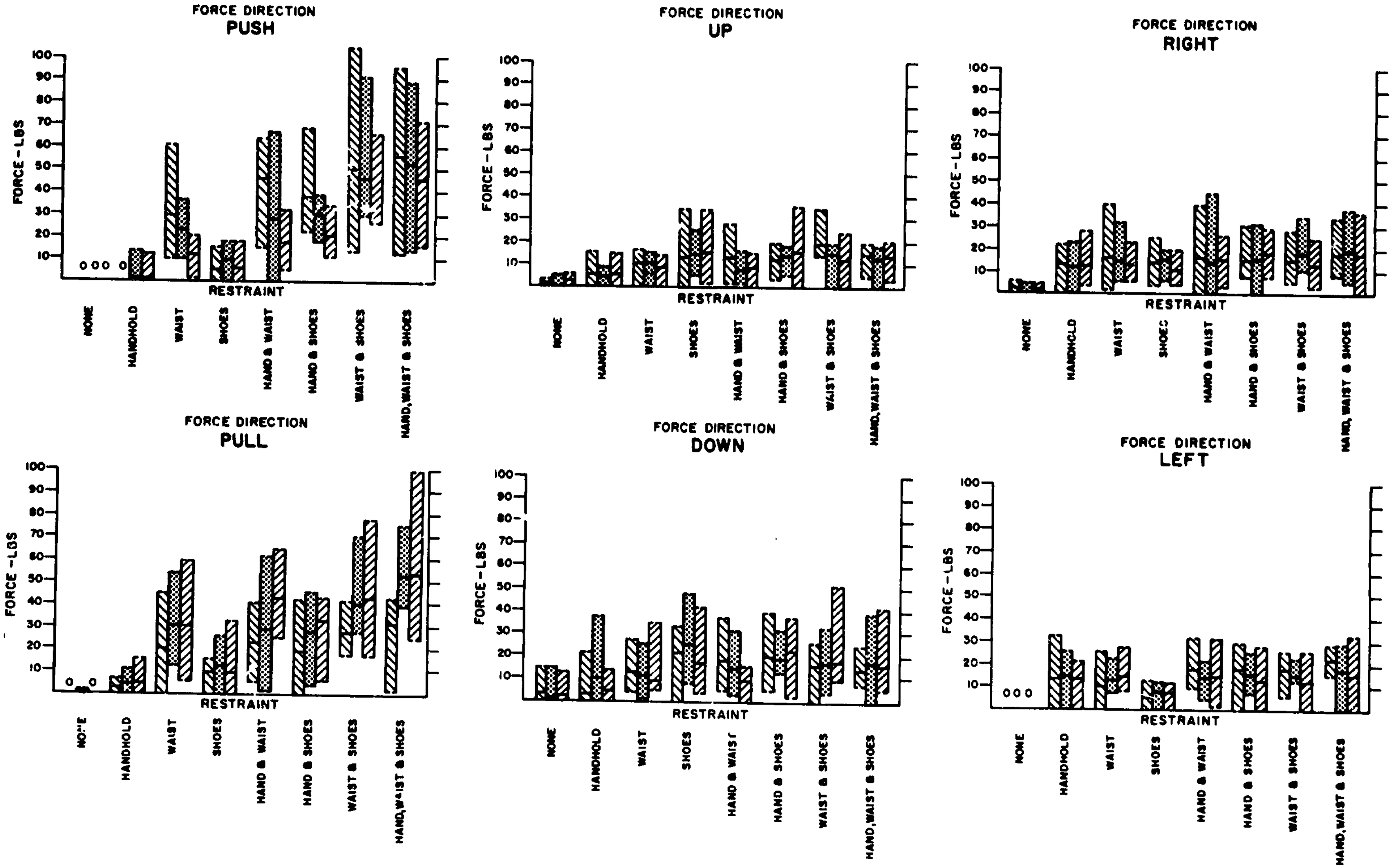
A-3 Summary Data Chart No. 3

FORCE TYPE: **SUSTAIN** RECEIVER ANGLE: **-15°** HANDLE ORIENTATION: **VERTICAL**



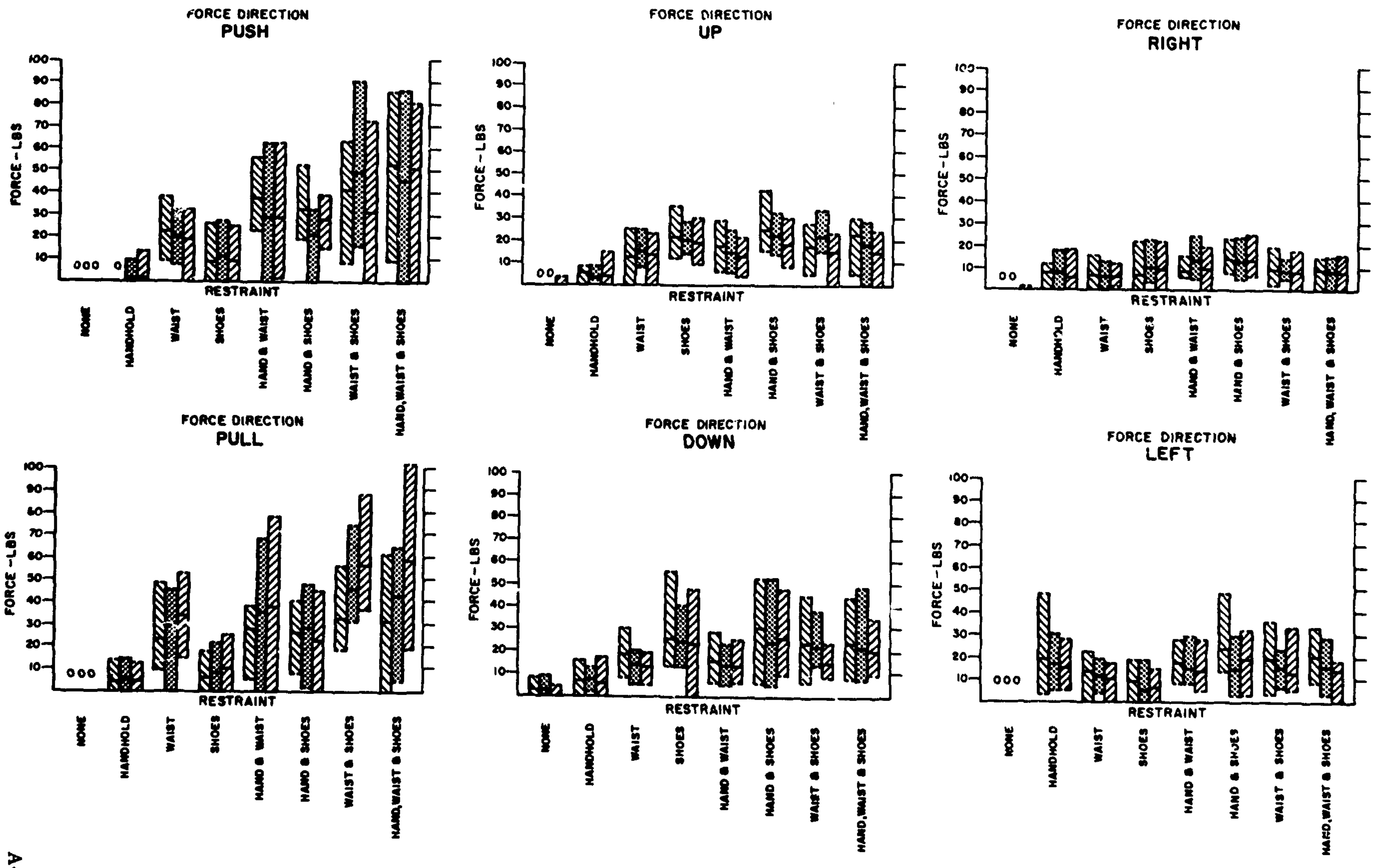
A–4 Summary Data Chart No. 4

FORCE TYPE: **SUSTAIN** RECEIVER ANGLE: **45°** HANDLE ORIENTATION: **HORIZONTAL**



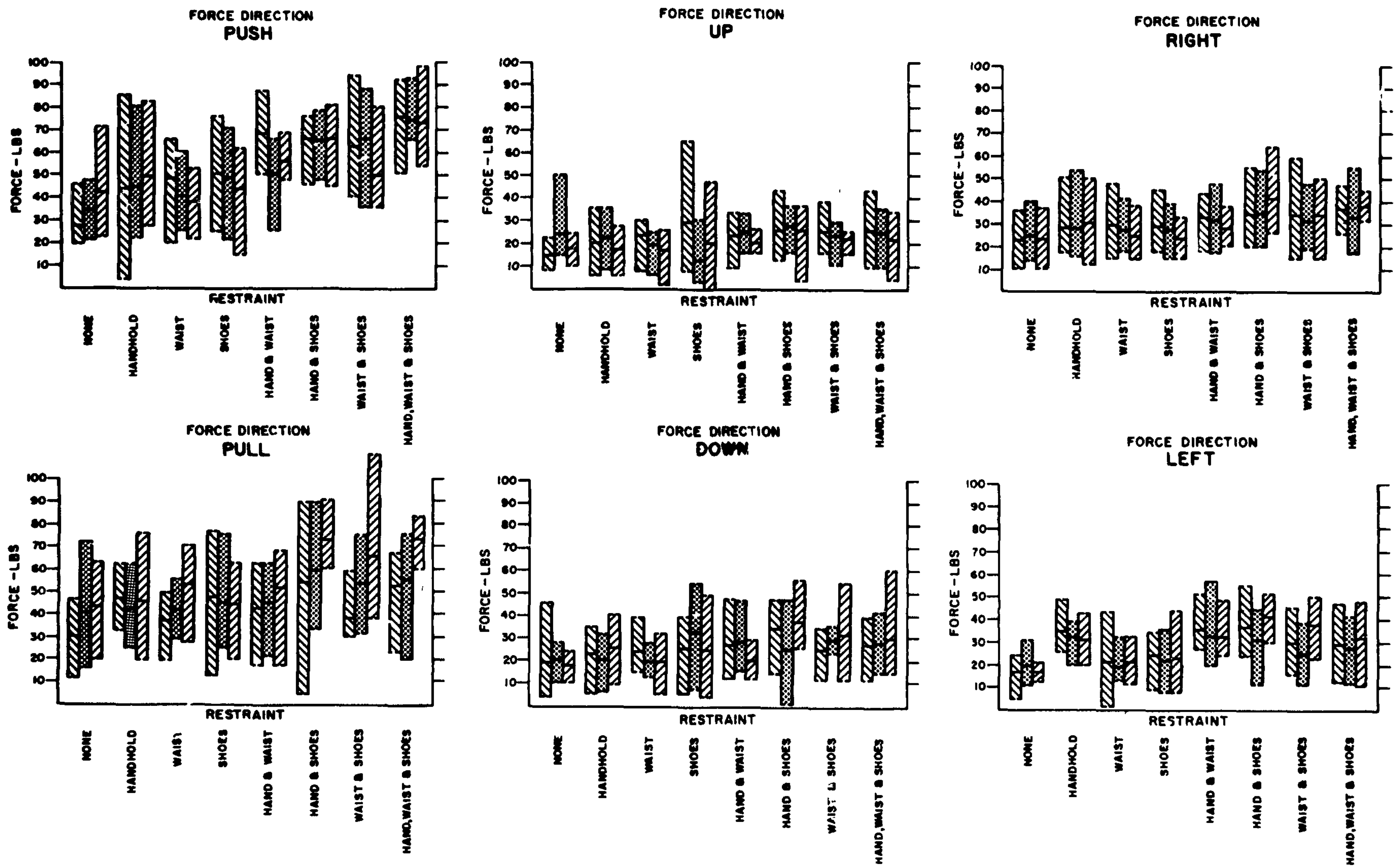
A-5 Summary Data Chart No. 5

FORCE TYPE: **SUSTAIN** RECEIVER ANGLE: **45°** HANDLE ORIENTATION: **VERTICAL**



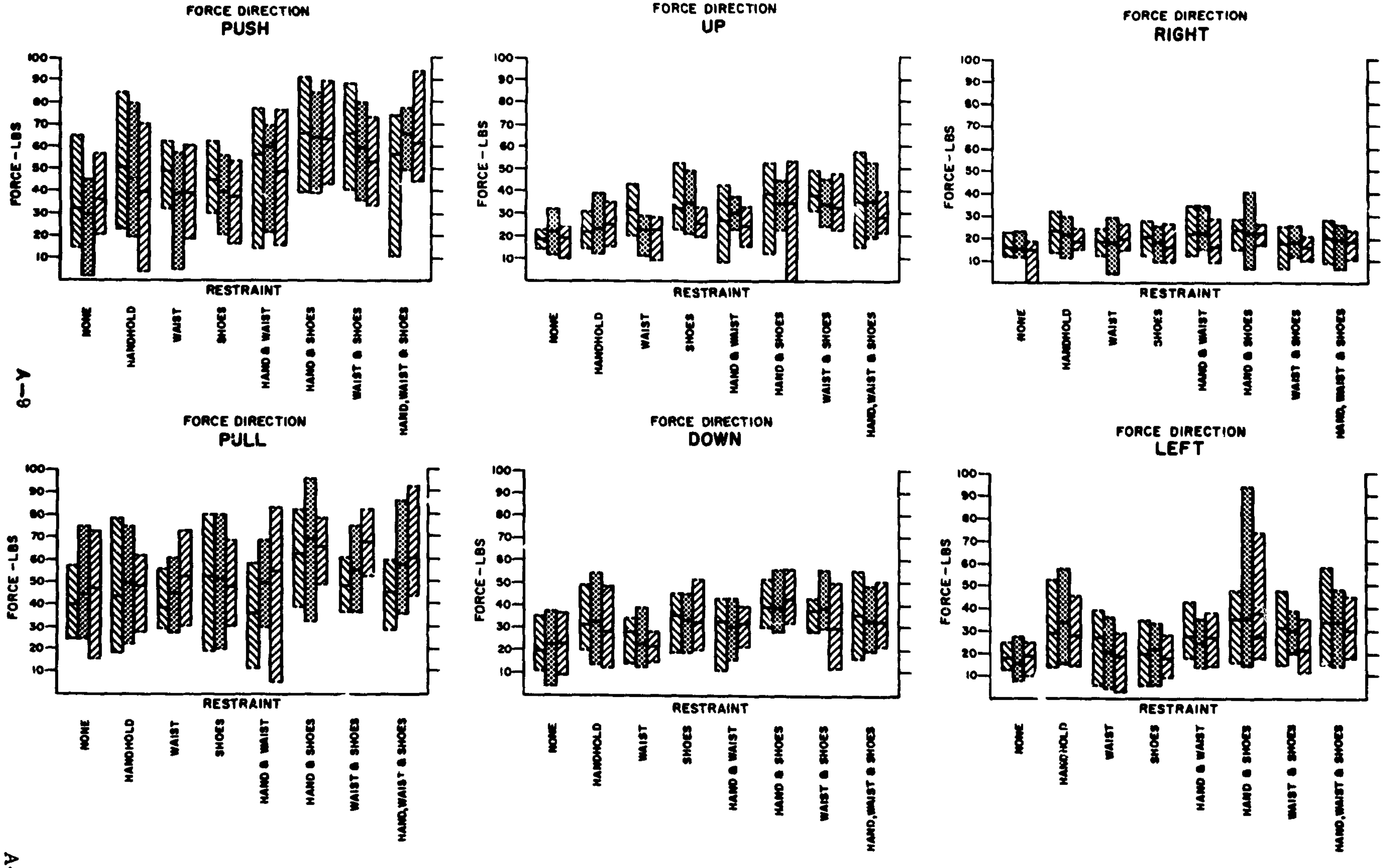
A–6 Summary Data Chart No. 6

FORCE TYPE: IMPULSE   RECEIVER ANGLE: 0°   HANDLE ORIENTATION: HORIZONTAL



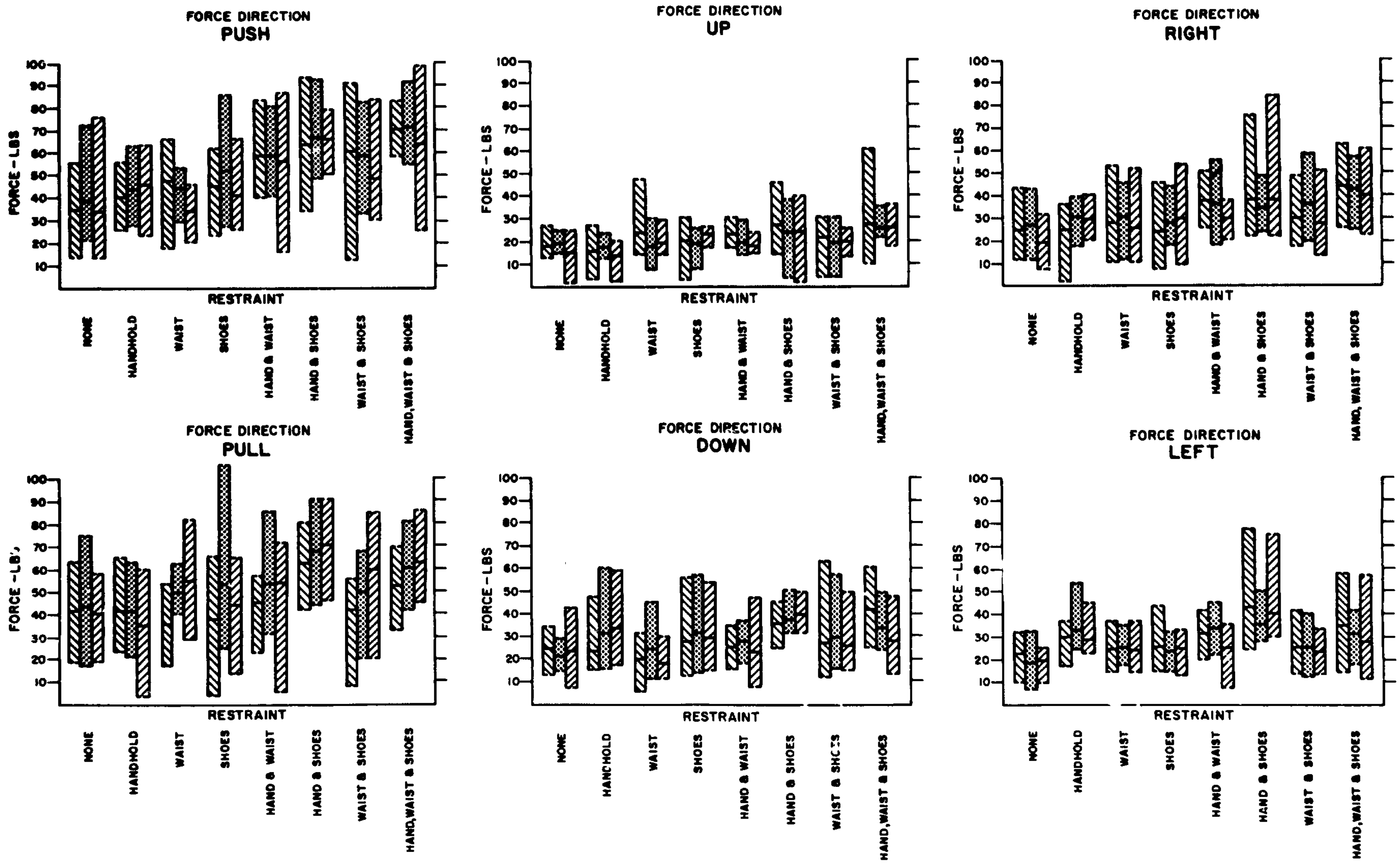
A–7 Summary Data Chart No. 7

FORCE TYPE: **IMPULSE** RECEIVER ANGLE: **0°** HANDLE ORIENTATION: **VERTICAL**



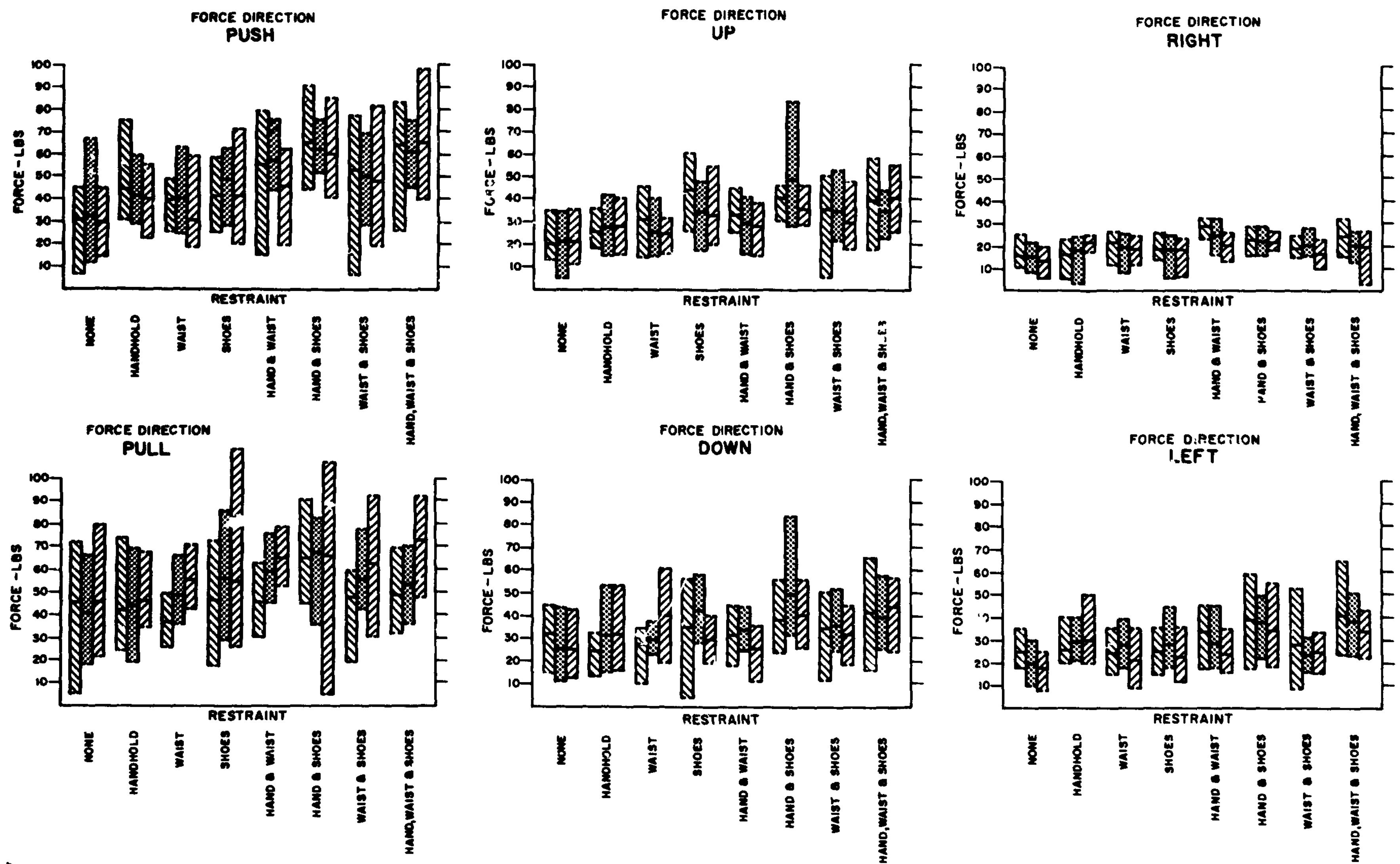
A-8 Summary Data Chart No. 8

FORCE TYPE: IMPULSE   RECEIVER ANGLE -15°   HANDLE ORIENTATION: HORIZONTAL



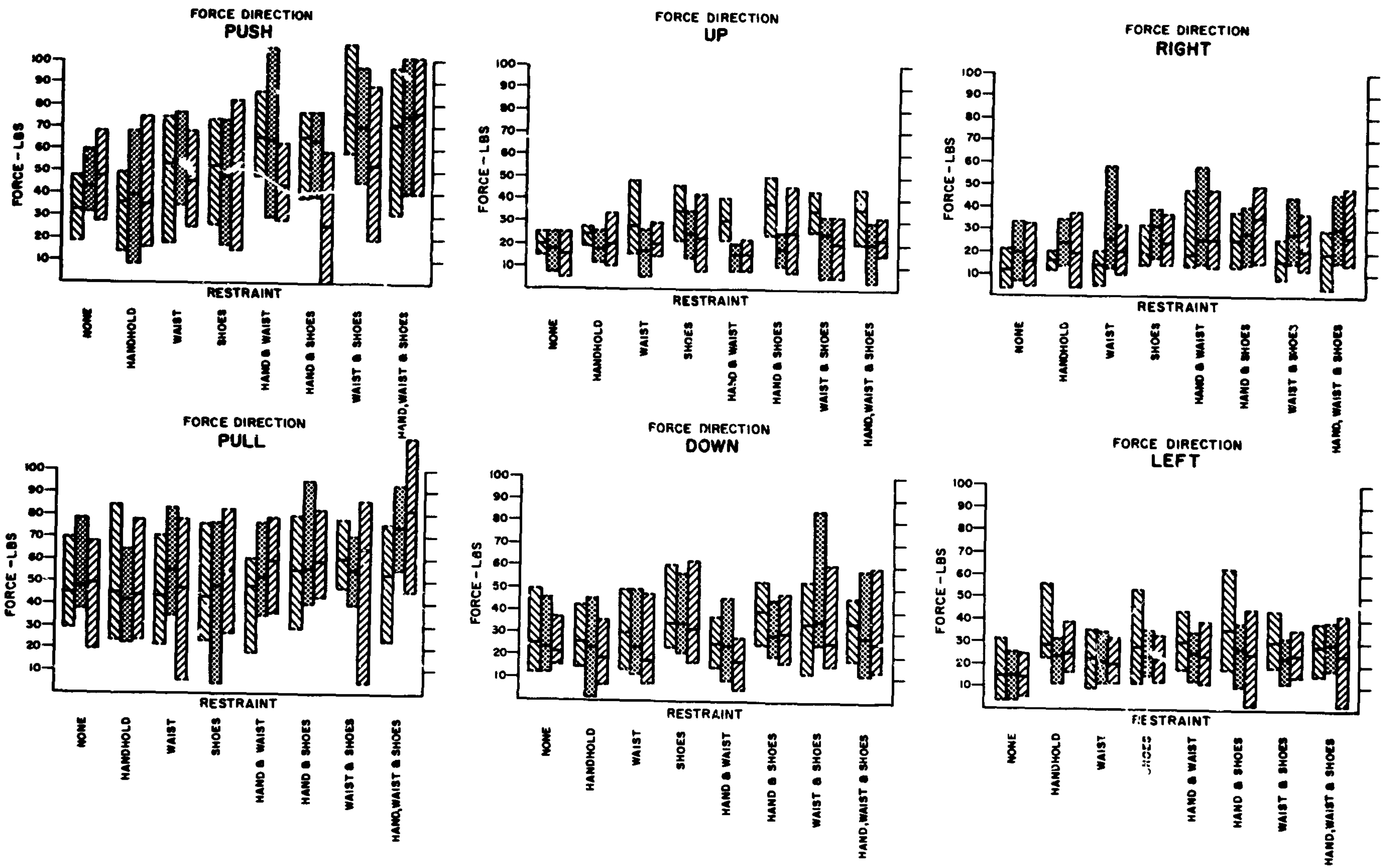
A-9 Summary Data Chart No. 9

FORCE TYPE: IMPULSE   RECEIVER ANGLE: 45°   HANDLE ORIENTATION: VERTICAL



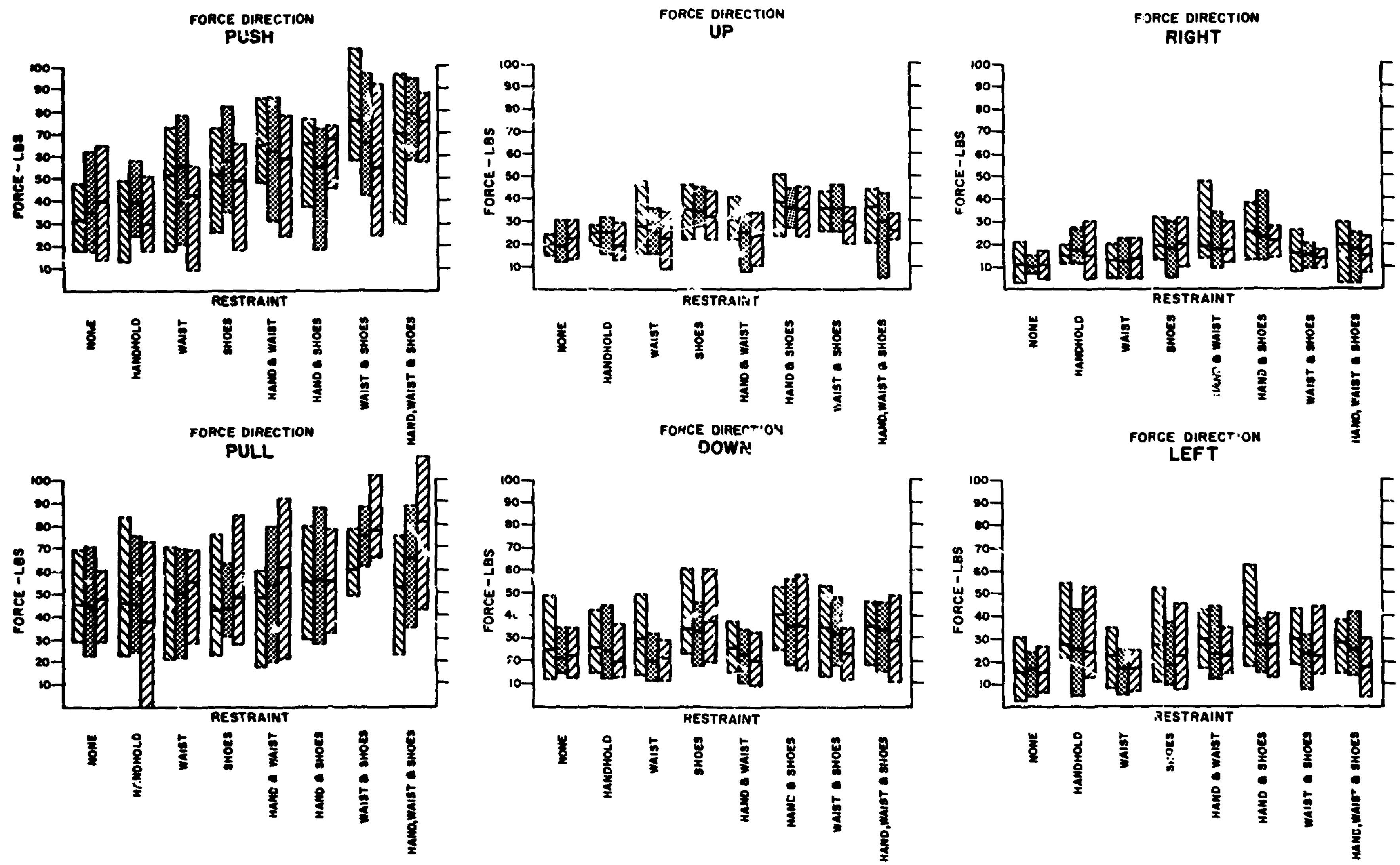
A—10 Summary Data Chart No. 10

FORCE TYPE: **IMPULSE** RECEIVER ANGLE: **45°** HANDLE ORIENTATION: **HORIZONTAL**



A–11 Summary Data Chart No. 11

FORCE TYPE: **IMPULSE** RECEIVER ANGLE: **45°** HANDLE ORIENTATION: **VERTICAL**



A–12 Summary Data Chart No. 12

SESSION V

MANEUVERING-UNIT TECHNOLOGY

Session Chairman: Colonel G. Lewis
AFSC (SCTSM)

STUDIES OF PILOTING PROBLEMS OF ONE-MAN FLYING UNITS OPERATED IN SIMULATED LUNAR GRAVITY

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SUMMARY: Current interests in small flying units to extend the range capabilities of lunar explorers coupled with a serious lack of information on the flight behavior of such vehicles have stimulated a series of studies of the handling qualities and often piloting problems of this type of vehicle by the Langley Research Center of the National Aeronautics and Space Administration. This paper briefly reviews earlier experimental studies of small earth-based vehicles and illustrates several configurations that have evolved in recent preliminary design studies of possible small lunar flying devices. These configurations include back-mounted, stand-up, and sit-down arrangements that are being, or soon will be, evaluated in the Langley studies made possible by the recent development of lunar gravity simulation techniques that permit the use of full-scale man-operated flying test-bed vehicles. The general nature of these fundamental studies as well as a few of the preliminary findings are described in this paper.

INTRODUCTION

Much attention in the field of advanced planning for future lunar missions has been focused on small manned flying devices as a means for extending the range capabilities of the lunar surface explorers. Although several such devices have been studied over the past several years with various degrees of effort, there has been little information obtained concerning the handling qualities and other critical piloting characteristics of such devices, due primarily to the lack of practical methods and facilities with which to undertake meaningful investigations of experimental operational hardware.

The purpose of this paper is to discuss briefly a recently initiated series of pertinent flight investigations some of which have been made possible as a result of the successful operation of the lunar landing research facility (LLRF, ref. 1, fig. 1) and new techniques being applied to it at the Langley Research Center of the National Aeronautics and Space Administration. In these studies, pilot evaluation of the flight behavior of full-scale test devices are being obtained under conditions of "real-live" operations in simulated lunar gravity during take-off, landing, and near hovering flight maneuvers, using research pilots experienced in lunar gravity simulations. The information gained from these flight investigations should be directly applicable to the design and development of specific lunar flying units, and should help approach maximum assurance of astronaut safety in the performance of the flying activities.

RECENT DESIGN EFFORTS AND PRIOR FLIGHT EXPERIENCE

A brief summary chart is presented in table I, in order to show the relation of the current Langley studies to recent design studies of small lunar flying devices as well as to prior efforts with experimental flying vehicles akin to these lunar designs. These various devices are categorized with respect to the stance of the pilot, general location of the main thrusters and the mode of attitude control. A list of some of the physical characteristics of the lunar designs is given in table II, and is accompanied by sketches and photographs in figures 2 through 7 to illustrate the various configurations.

The geneses of most of the actual flight experiments are in the pioneering works of Wendell Moore of Bell Aerosystems who developed the flying belt concept (ref. 2), and Charles Zimmerman of the Langley Research Center who conceived the balance reflex or kinesthetically controlled flying platform (ref. 7). Although most of these flight experiments were more or less successful, the information gained was only exploratory in nature and, of course, applied only to flight behavior in earth gravity. One exception, however, is the project involving piloted flight in simulated lunar gravity carried out at Langley Research Center in about 1962 using a simple sit-down configuration with manually operated air jets for pitch and roll control, and foot-operated air jets for yaw control (ref. 12). Some pilot handling evaluation was performed; however, the maneuvering capabilities of the vehicle and system were very limited.

The design studies listed in tables I and II, and depicted in the accompanying figures, illustrate the many possible approaches which can be taken to integrate a pressure-suited man with any of several different propulsion, control, and landing systems. The overall objective of the current Langley studies is to explore these various approaches using test-bed equipment which are representative of the various design concepts and whose systems can be readily modified in an attempt to identify characteristic handling qualities of each type. Of course, subsequent efforts can be directed toward refining our knowledge of those system approaches which appear most promising. Those Langley studies related to the differing design approaches are listed in the final column of table I and are described in the following discussion.

PROJECT ICARUS

The current Langley study of the backpack propulsion unit, which probably represents the minimum-sized flying system, that can be used for lunar locomotion was initiated with the in-house design and construction of the experimental hardware (depicted in fig. 8) about 2 years ago. This project, nicknamed "ICARUS" (after the mythical minimum flying system that suffered system deficiencies due to the lack of updated technical information) was based in part on the earlier design studies carried out by Bell Aerosystems for the Langley Research Center and discussed in reference 3, as well as on their experience with the flying belt system which had been developed with funds from the Army.

Exploratory studies of man's self-locomotive capabilities in lunar gravity using the Lunar Walking Simulator at Langley, revealed that loads up to 500 earth pounds could be carried on the back with relative ease (ref. 14); consequently, a target weight somewhat below this value was selected. The completed unit, which includes a built-in life-support system for the pressure-suited subject, weights about 300 pounds with 110 pounds of hydrogen peroxide for a maximum flight time of about 3 minutes. Existing off-the-shelf hardware was used inasmuch as this experimental unit was not intended as a finalized space-qualified system. The unit consists of two main hydrogen peroxide fuel tanks and two nitrogen pressurizing tanks which feed two independently controlled thrusters, each mounted on the end of an arm pivoted near the pilot's shoulder. The thrust level and direction of each thruster are controlled by motions of the arm and hand adjacent to each thruster. The manually operated controls are readily adjustable through a range of linkage settings giving the pilot a selection of control sensitivities.

This project is divided into two phases, the first of which involves flight attempts with three-degrees-of-freedom motion in the pitch plane only, as provided by the previously mentioned Lunar Walking Simulator (LWS). A photograph showing the installation of the test unit on the back of the test subject suspended in the LWS is shown in figure 9. The subject and propulsion unit are supported by separate cables attached to two lightweight trolleys free to roll along an overhead track that is parallel to the 200-foot-long walkway on which the test subject is standing. The walkway is displaced from directly beneath the overhead track so that the cables are at approximately 9.5^{o} from the vertical. In this manner the component of the weight of the subject and his flying unit equal to their equivalent lunar weight is acting in the plane of motion so as to simulate lunar gravity. The subject with his flying unit is free to rotate in pitch but is restrained in roll and yaw and, also, he is free to travel 200 feet in his fore-and-aft direction and 30 feet in his up-and-down direction. For the initial flight attempts, tether cables are attached to the flying unit and are handled by ground crewmen who keep the cables slack except in the case of an emergency.

The specific objectives of phase I are: (1) to evaluate the feasibility of the new flight technique involving the inclined-plane lunar gravity simulator, (2) to determine if the pilot can use his legs effectively to replace a structural landing gear that would otherwise be required, and (3) to evaluate pilot handling qualities in this flying mode of limited degrees of freedom for both "shirt-sleeve" and pressure-suited conditions. Several test sessions have been completed and the initial results indicate that the technique is feasibile and that the pilot can use his legs, at least in the "shirt-sleeve" conditions, quite effectively to absorb the landing impacts.

The second phase of the ICARUS project will involve testing the unit with six degrees of freedom using a vertical suspension technique similar to that which had been developed for the next project to be described. The objectives of the second phase effort are: (1) to explore in depth the handling qualities of the back-mounted unit, and (2) to determine the significance of the differences between the two testing techniques.

PROJECT POGO

Project POGO, which has been already completed, is an exploratory study undertaken jointly by the Marshall Space Flight Center and the Langley Research Center, using the experimental POGO flying device provided by Bell Aerosystems and space suits provided by the Manned Spacecraft Center. A photograph of the experimental unit flown in simulated lunar gravity using the lunar landing research facility is shown in figure 10, and a complete description of this project is covered in reference 4. This stand-up unit represents a somewhat larger size vehicle than the ICARUS unit and incorporates an elementary type landing gear. All controls are manually operated. The flying unit is mounted in a gimbaled-whiffletree system attached to an overhead constant-tension unit suspended on cables from the servo-controlled traveling bridge crane of the LLRF. The gimbaled support system permits the vehicle to rotate with the three degrees of angular freedom, and the constant tension unit allows the vehicle to travel vertically over a distance of about 10 feet above the ground. A force equal to about five-sixths of the total system weight is applied to the vehicle by the constant-tension unit. Cable angle sensors attached at the top of the cables cause the bridge crane to stay directly over the rocket-powered vehicle so that the cables remain vertical at all times.

The basic vehicle was flown first in earth gravity at Bell facilities and then in simulated lunar gravity at Langley by two experienced pilot subjects. The conclusion was drawn that lunar flight with such a vehicle was feasible although a number of shortcomings with the particular design and hardware were noted. Furthermore, it was found that lunar gravity provided a generally favorable effect by slowing down all of the vehicle's responses to pilot control inputs so as to give him more "think time" while performing low altitude, near hovering maneuvers. Although use of pressurized "soft" and "hard" type space suits produced some unfavorable effects, no major problems were experienced as long as the suits were fitted properly to the pilot and the vehicle was fitted properly to the suits. Because of the exploratory nature of this project no instrumentation was provided, and no detailed studies of the handling qualities were undertaken.

PROJECT OMPRA

This project, utilizing the hardware shown in figure 11, is directed toward the study of a waist-mounted propulsion system and was developed primarily for zero gravity space applications. Downward firing thrusters have been provided, however, so that the system can be adapted to the lunar gravity application. The system utilizes clusters of cold gas thrusters operated in either on-off or proportional fashion through hand-operated electronic control systems with optional use of different types of stabilization systems. The pilot and propulsion unit are gimbal-supported from a cable system which is attached to the Rendezvous and Docking System (RDS, ref. 15) through a constant-tension unit which either fully or partially supports the suspended weights. The RDS uses cable angle sensors in the same manner as the LLRF so as to track the flying unit and keep the cable vertical. The status of this project is that the hardware, built under contract by Bell Aerosystems, is currently being installed and checked out for system performance.

PROJECT FLEEP

This fourth project involving a vehicle intended to be representative of larger flying units is currently in its preliminary design and fabrication stages at Langley. A conceptual sketch of this flying test bed is illustrated in figure 12, showing the stand-up version used to study the balance reflex (kinesthetic) as well as other control system concepts. It is anticipated that the unit will be convertible to a sit-down version for evaluation of the other possible configurations. The unit will be tested using the LLRF to achieve six degrees of freedom within a flight envelope of about 360 feet long, 42 feet wide, and 150 feet high. A special servo-controlled vertical cable system to support five-sixths of this approximately 1300-pound unit will be attached to the bridge system which normally operates with a 12 000-pound vehicle.

Special attention to the problem of lunar-gravity simulation is required for the case of the kinesthetic control configuration because for this control concept the body weight is shifted to produce the desired control moments for the pitch and roll axes. The weight of the pilot must, therefore, be reduced to his equivalent lunar weight in order that the resulting control moments produced by shifting his body relative to the thrust vector will be of the proper magnitude. Consequently, five-sixths of the pilot's weight must be suspended independent of the vehicle. Various schemes for achieving this unusual requirement are currently being evaluated so as to find an arrangement that will have a minimum interference with the pilot and the flying unit. Flight testing of this unit is expected to be underway in about 1 year. The objectives of this project are, of course, essentially the same as those of the previous projects.

LUNAR LANDING RESEARCH VEHICLES

Although not mentioned previously in this presentation, the recent and continuing studies of the Apollo lunar landing mission using full-sized lunar landing research vehicles operating in simulated lunar gravity provide some very useful information relative to their "little brother" counterparts, inasmuch as these larger vehicles are involved in lunar "flying" while they are being landed. One of the research vehicles is the LLRV (fig. 12) which had been flown originally as a research vehicle at the Flight Research Center (ref. 16) and was more recently flown as a training vehicle at MSC until being completely demolished in the recent crash. A second type vehicle is the one continuing to be used at Langley with the LLRF (ref. 1, fig. 14). The LLRV weighed about 4000 pounds and was equipped with a pilot's compartment in a sit-down configuration, whereas the LLRF vehicle weighs about 12 000 pounds and is equipped with two interchangeable cabs; one a sit-down version (somewhat similar to the LLRV), and the other a stand-up version duplicating the flight commander's portion of the Apollo lunar module's manned compartment. Both research vehicles incorporated sophisticated control systems which had different modes of operation.

This type of vehicle, of course, will provide our first experience in actual lunar flying during the Apollo mission and, consequently, will provide the initial opportunity for correlation of simulated lunar gravity flight experience with that of actual lunar flight. It should be noted that the buildup for the Apollo mission involves cross training of the astronauts and MSC pilots in LLRV-type vehicles and the LLRF vehicle which should provide pertinent information in this correlation. A further consideration is the fact that the Langley and MSC pilots and astronauts who have gained experience in simulated lunar flying with these vehicles will be able to apply this experience to the evaluation of the smaller vehicles which are of more direct interest in this particular paper.

CONCLUDING REMARKS

Emphasis has been placed on studies of handling qualities and other piloting problems as applied to general types of lunar flying units. However, the techniques and experience developed in carrying out these particular studies should also be directly applicable to the training program for the astronauts involved in the actual lunar flying activities with the newly developed devices.

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TABLE I.- SUMMARY OF ONE MAN FLYING UNITS AND DESIGN STUDIES OF LUNAR FLYING UNITS

Pilot Stance	Point of Thrust Application	Attitude Control	Source	Prior Flying Experience	Design Studies	Current Lunar Flight Studies
Stand	Back	Manual rotation of thrust	Bell Aero.	Flying Belt Reference 2.	———	———
			LRC-Bell	———	Reference 3	———
			LRC	———	———	ICARUS
	Chest	Manual rotation of thrust.	LRC-MSFC-Bell	Free flight. POGO Reference 5	Reference 4	POGO
	Waist	Auxiliary thrusters	Hamilton Standard	———	Reference 6	———
			LRC - Bell	———	———	OMPRA
	Floor	Pedal rotation of thrust.	LRC	Flying platforms. References 7&8	———	FLEEP
			Hiller	Airborne personnel platform Reference 9.	———	———
			Delackner	Aerocycle. Reference 10	———	———
			North American Rockwell	OMLTD test bed. Reference 11	OMLTD Reference 11.	———
		Auxiliary thrusters.	LRC - Bell	———	Reference 3	———
			LRC	———	———	FLEEP
Sit	Back	Manual rotation of thrust.	Bell	Flying Chair	———	———
	Floor	Auxiliary thrusters.	LRC	*Simulated Lunar Flying Vehicle. Reference 12.	———	———
			LRC- Bell	———	Reference 3	———
			MSFC-Bell	———	Reference 13	———

* Simulated Lunar Gravity

TABLE II - [illegible]SICAL CHARACTERISTICS OF ONE MAN FLYING UNITS DEVELOPED IN VARIOUS DESIGN STUDIES

Configuration	Waist mounted OMLS	Stand-on platform (OMLTD)		Advanced Pogo	Back-pack ΔV-4000 fps	Two-man sit-down ΔV-2000 fps	One-man stand-on ΔV-8000 fps	Two-man sit-down ΔV-6000 fps	Lunar Flying Vehicle	Simulated Lunar Flying Vehicle
Source	Ham. Stand. MSC	North American Rockwell		Bell Aero. MSFC	Bell Aero. LRC	Bell Aero. LRC	Bell Aero. LRC	Bell Aero MSFC	Bell Aero MSFC	LRC
Type of pitch and roll control	Differential main thrust	Bal[illegible] reflex		Manual thrust gimbal	Manual thrust gimbal.	Auxiliary thrusters	Balance reflex	Auxiliary thrusters.	Differential main thrust	Auxiliary thrusters
Pilot weight, lb.	260	2[illegible]0	260	294	200	245	245	245	293	185
Empty weight, lb.	114	83	83	146	201	208	300	310	448	175
Fuel weight, lb	59	150	150	273	230	170	770	740	635	——
Payload, lb.	0	0	260	100	30	245	30	245	300	——
Gross weight, lb	433	493	753	813	636	868	1344	1540	1677	360
Thrust-to-lunar weight ratio	11	7.3	4.8	1.8	3.0	3.0	3.0	3.0	18	106
Pitch inertia, slug ft^2	———	———	———	———	35	74	175	[illegible]3	195	17
Landing gear, span, ft.	Feet	5	5	5.5	Feet	7.0	8.5	7.0	7.3	2.5
Platform height, ft.	None	1.5	1.5	1.0	None	3.2	1.0	3.5	3.5	1.0
Design range, miles	0.8	24	———	5.7	———	———	———	———	15	———
Figure number	2	3		4	5				6	7
Reference number	6	11		5	3				13	12

Figure 1.- Photograph of the Langley lunar landing research facility (LLRF).

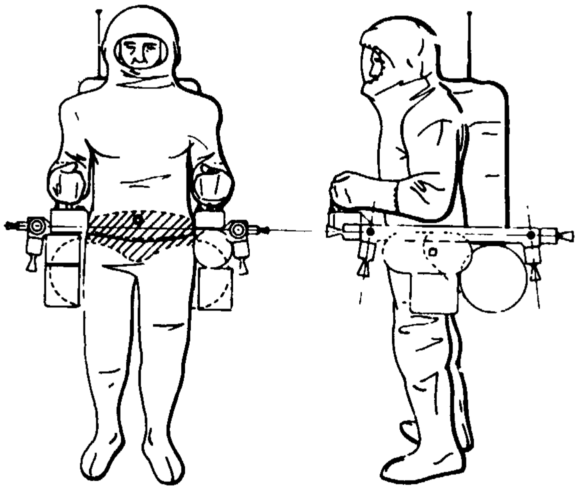

Figure 2.- Sketch of the Hamilton standard one-man location system developed for Manned Spacecraft Center (ref. 6).

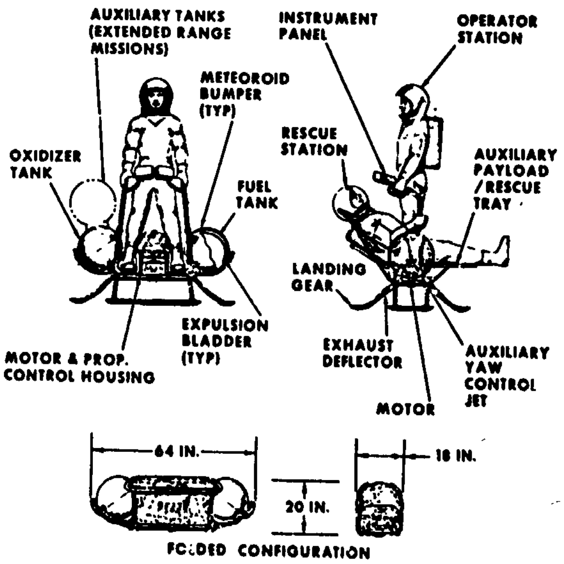

Figure 3.- Drawing of North American Rockwell's one-man lunar transportation device (ref. 11).

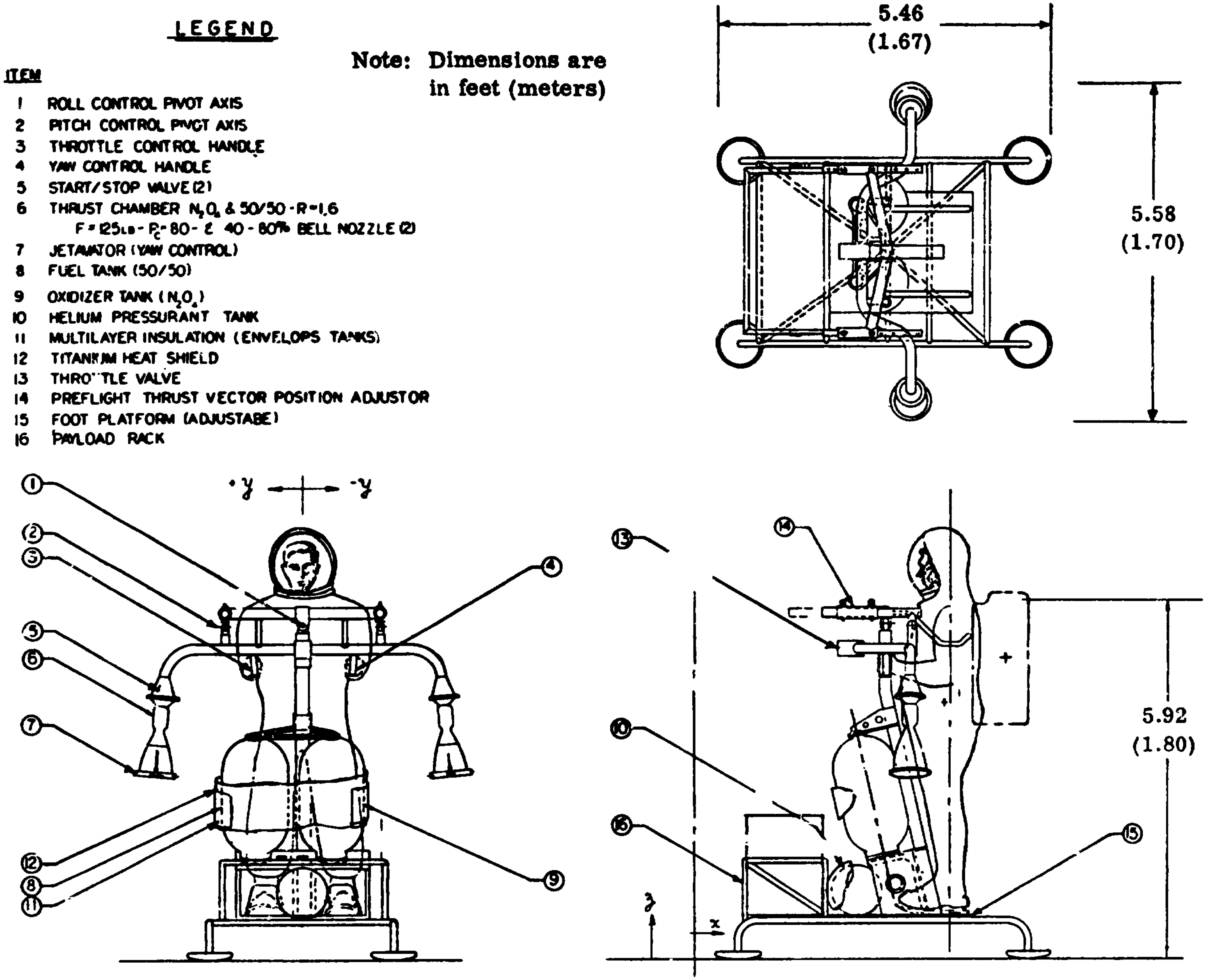

Figure 4.- Drawing of Bell Aerosystems advanced POGO based on early flying belt and original POGO experience (ref. 5).

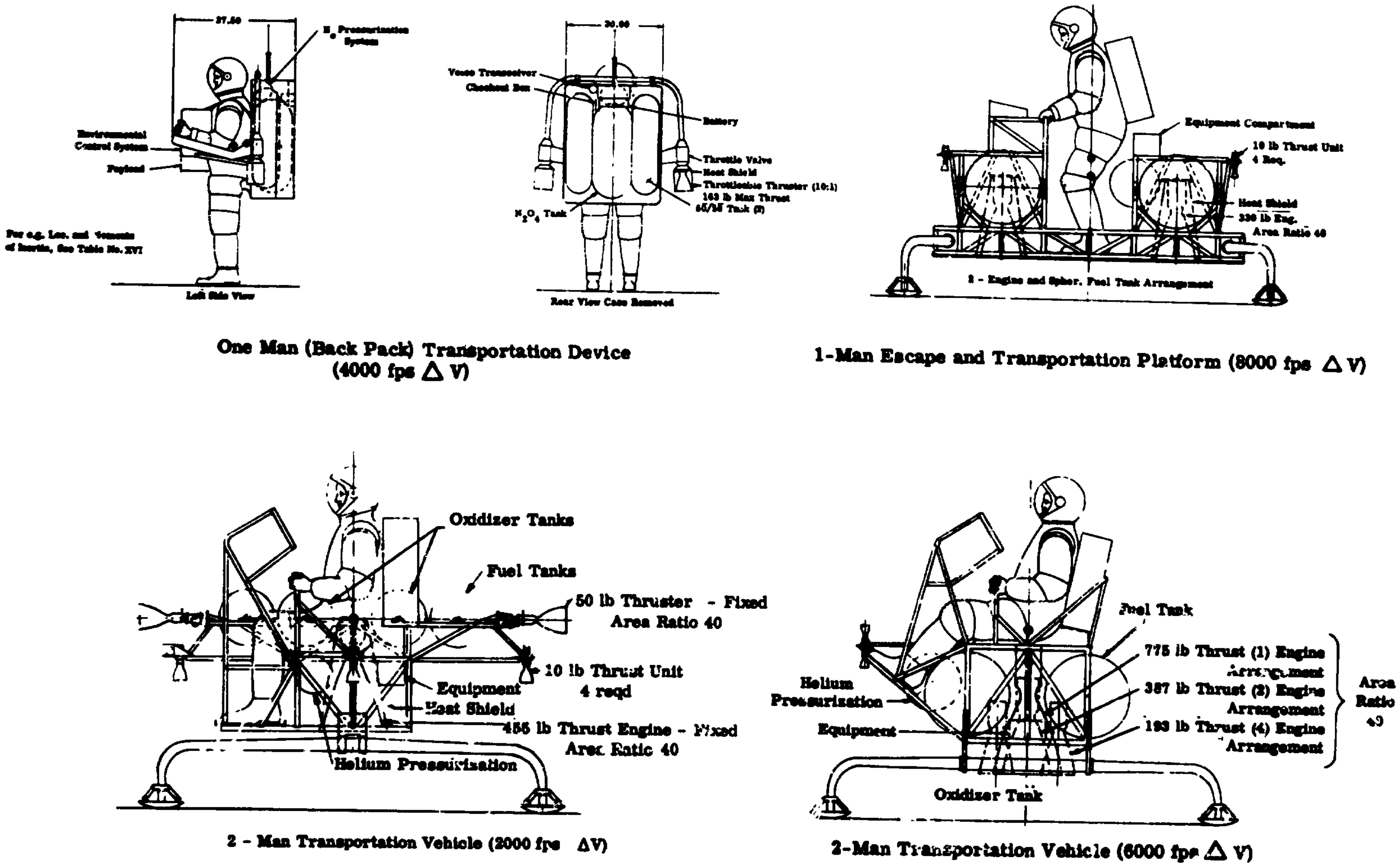

Figure 5.- Drawings of various one-two man lunar flying vehicle configurations developed in a design study by Bell Aerosystems for Langley Research Center (ref. 3).

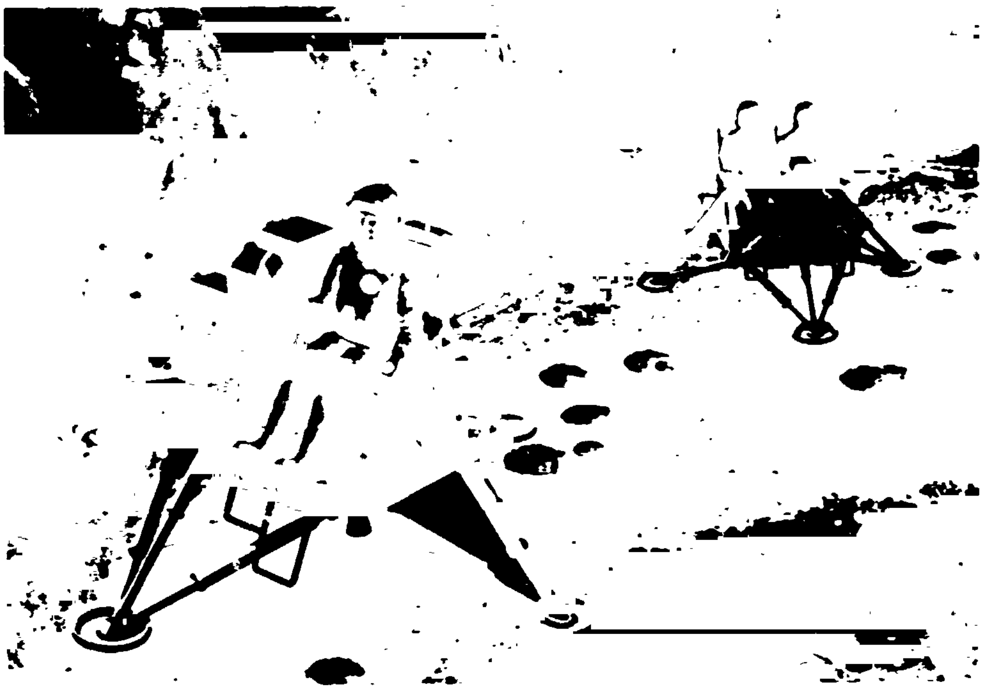

Figure 6.- Drawing of the Bell Aerosystems lunar flying vehicle "sit-down" configuration developed for Marshall Space Flight Center (ref. 13).

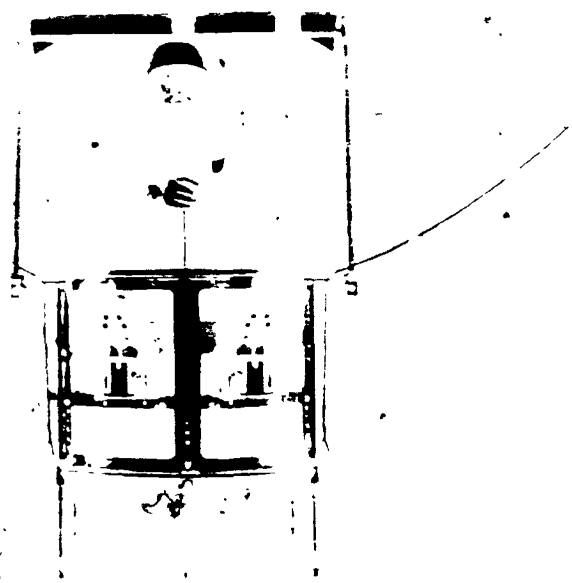

Figure 7.- Photograph of one-man test vehicle used in evaluating servo-controlled suspension system for lunar gravity simulation (ref. 12).

Figure 8.- Photograph of the Langley lunar flying backpack mounted on the research pilot wearing a pressurized space suit. During test flights a personnel life support system will be housed within the backpack assembly.

Figure 9.- Photograph of ICARUS flying unit and subject mounted in the Langley lunar walking simulator. The subject is shown in the "shirt-sleeve" condition.

Figure 10.- Photograph of the Bell Aerosystems "POGO" lunar flying research vehicle as tested at the Langley lunar landing research facility. The operator is wearing a pressurized space suit and life support backpack. Vehicle is partially supported by the whiffletree attached to the overhead constant tension unit (ref. 4).

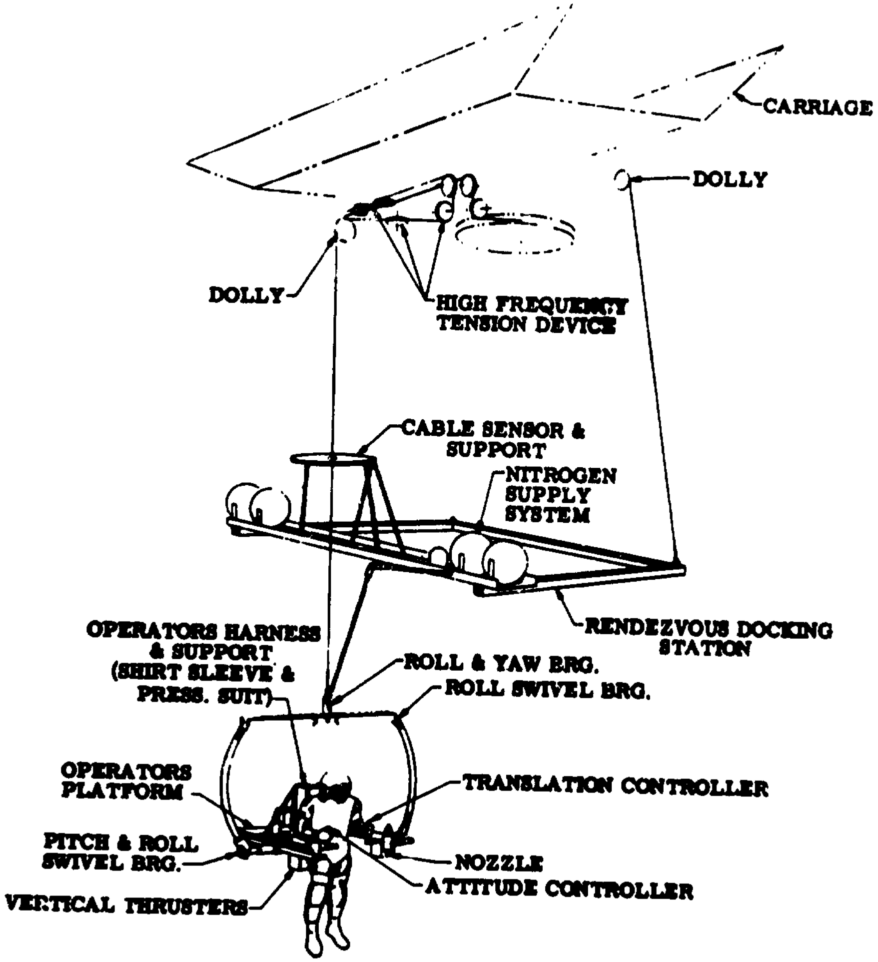

Figure 11.- Sketch of the general arrangement of the OMPRA flying unit suspended from the Langley rendezvous and docking simulator (RDS).

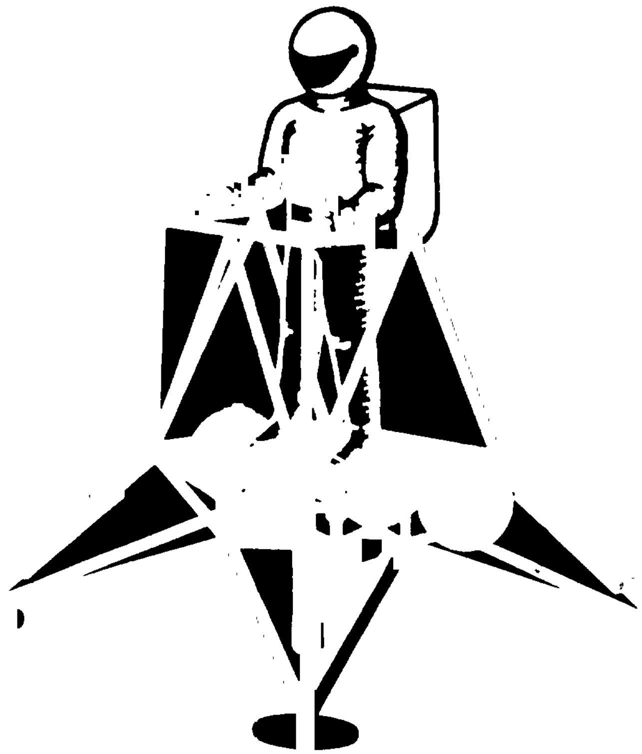

Figure 12.- Sketch of proposed "stand-on" configuration for the one-man lunar flying research vehicle.

Figure 13.- Photograph of the lunar landing research vehicle (LLRV) flying at Flight Research Center.

Figure 14.- Photograph of the Langley lunar landing research facility showing the research vehicle with a "stand-up" pilot's compartment in the flying attitude. The pilot's compartment simulates many of the features of the Apollo lunar module.

APPLICATION OF CONTROL MOMENT GYROS TO ASTRONAUT MANEUVERING UNITS

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SUMMARY: The basic requirements for stabilization and control of an astronaut maneuvering unit (AMU) for free-space operation are defined. Use of reaction jet control in present AMU systems is described. The application of control moment gyros (CMGs) to AMU attitude control and stabilization is discussed in detail. A final system concept is selected and justification for the selection is presented.

INTRODUCTION

Stabilization and propulsion devices, to increase the utility of an extravehicular astronaut, have been under study since 1959. Many concepts have been defined, and a few have been carried into experimental hardware for flight test in the AF KC-135. Two such devices have been carried into orbit on the Gemini spacecraft; the operational devices were the NASA Hand-Held Maneuvering Unit (HHMU) and the AF/MSD Gemini Astronaut Maneuvering Unit (AMU). Only the HHMU was flown in space, and the flights performed were limited in scope. The requirements for an extravehicular propulsion device therefore, are not firmly established by flight test. Also, no firm requirements for "mission-dependent" extravehicular activity (EVA) have been established at this time. In the absence of a firm mission requirement, the design criteria for extravehicular propulsion are heavily dependent on the "assumed" mission.

NASA/MSC, working with short-tethered EVA missions in the vicinity of a spacecraft, has developed a hand-held propulsion unit employing completely open-loop, manually stabilized, control. The Air Force propulsion laboratory assumed an orbital assembly and repair mission requiring untethered flights of approximately one mile with up to four hours of independent operation. This mission resulted in the design of a stability augmented backpack unit typified by the Gemini AMU. NASA/MSFC assumed missions requiring extended work periods and the handling of large objects, which resulted in the definition of maneuvering work platforms and taxis employing manipulators.

The Missiles and Space Division

of LTV Aerospace (MSD) developed the Gemini AMU and several experimental vehicles, all employing stability augmented propulsion systems. In each case, several stability augmentation techniques were studied and traded off against mission requirements, system weight, system performance, and the state of the art of the candidate systems. A monopropellant propulsion system providing stabilization, attitude and translation control, and attitude rate limiting, and which was referenced to strapped down rate gyros, proved to be lightest and most easily implemented in each case. However, the state of the art is changing constantly and the same tradeoffs, if made today, might provide a different answer. This is particularly true in the case of control moment gyros (CMGs). Their energy consumption is closely related to the state of the art and is almost independent of the momentum transfer they are required to provide in a "cyclic" system.

In 1967, MSD joined with the Navigation and Control Division of The Bendix Corporation to study the application of CMGs to a "backpack" type AMU. This paper reports the result of that study.

AMU CONTROL REQUIREMENTS

An extravehicular astronaut, operating away from the surface of his spacecraft, must have a means of propulsion and a means of orienting the propulsion thrust vector. In a practical sense, this means he must be able to control pitch, roll and yaw, and be able to generate a force for translational acceleration and deceleration in at least one axis. These "minimal" control provisions should permit an EVA astronaut to maneuver within a few feet of his parent spacecraft.

If EVA transfers to a distant target are required, it becomes desirable to correct velocity errors normal to the line of sight without twisting the body and losing sight of the target. This capability can be achieved by adding translational jets in the other two axes. During long transfers, where the target appears as a point, it becomes difficult to differentiate between line-of-sight deviations caused by drift off the line-of-sight and those caused by an attitude change of the astronaut. Such deviations can be positively identified as drift off the line-of-sight if the astronaut is inertially stabilized.

The Air Force AMU, prototyped for Gemini flight testing, was designed for orbital assembly and maintenance tasks requiring long transfers. Therefore, it was designed with multiple-fixed thrusters and stability augmentation to permit pure (uncoupled) translation and attitude control, and stabilized coast. A mode select switch was made available to the astronaut to permit evaluation of unstabilized flight. Although the Gemini/AMU was not flown in orbit, it was accurately simulated and flown by both engineering and flight crew personnel in a simulated rendezvous and docking maneuver. These simulations provided a means to evaluate AMU control requirements. Control parameters evaluated by engineers and pilots, and later validated by the Gemini flight crew included: translational acceleration; angular acceleration; translational rates; angular rates, and stabilization deadbands. The evaluated flight techniques included: open-loop acceleration command control; attitude stabilization with angular rate command and translation acceleration command control rendezvous with a simulated Gemini by establishing a closing rate and correcting to stay on the line-of-sight; and rendezvous by a modified bird-dog technique.

A translational thrust of about five pounds, producing approximately 0.4 ft/sec^2 acceleration, was selected. This thrust level, coupled with constraints on thruster location associated with the geometry of the man, produced angular accelerations on the order of 10 deg/sec^2 in roll, 13 deg/sec^2 in pitch, and 23 deg/sec^2 in yaw. These values were controllable and appeared satisfactory. Angular rates, on the order of 15 to 18 deg/sec, were selected as optimum for the AMU. Note that these rates are substantially higher than Mercury and Gemini rates. Translational rates, originally estimated at 5 to 10 ft/sec, seldom exceeded 4 ft/sec on 200-foot transfers and were most frequently 1 to 2 ft/sec. The stabilization deadband was set to what was considered a low practical limit of approximately $\pm$2 degrees, by pilot preference. The original transfer technique set up for long translations (i.e., coasting down the line-of-sight making corrections normal to the line-of-sight) was largely abandoned by the flight crew in favor of a modified bird-dog technique for the 200-foot transfer. They started from a station-keeping (no closing velocity) situation and applied small amounts of thrust which directed them toward Gemini. As they drifted off, they oriented themselves toward Gemini and again applied thrust. If closing speeds became too high, they applied retrograde thrust.

Various AMU missions and some of the simulation data were analyzed to determine the amount of fuel consumed in generating and nulling translational velocities, and in generating and nulling rotational rates. For the Gemini AMU, fuel consumption was about equally divided on the average. It is important to note that fuel consumption is a function of rates, rather than distances or total angles traversed. Therefore, both the total fuel used for a mission and the distribution between translation and rotation can be changed by varying the rate.

Open-loop rendezvous were successfully accomplished; however, more time, effort, and fuel were required to accomplished these rendezvous.

CURRENT REACTION JET SYSTEMS

Most of the AMU devices designed to date employ reaction jets for stabilization, attitude control and translation. This choice is based primarily on the fact that a reaction jet system is an absolute necessity to produce translation and to counteract any noncyclic torques imposed on the system. Momentum exchange systems that can produce cyclic torques must therefore be supplemented with a reaction jet system and results in a greater total system weight and complexity. Also, all of the systems to date have used a single set of thrusters for both translation and attitude control.

In the Gemini AMU, which is a typical reaction jet system, multiple-fixed hydrogen peroxide thrust chambers are arranged symmetrically around the EVA system center-of-gravity. When flown "open loop," the commanded thrusters come on at rated thrust producing a pure translation or attitude acceleration at a constant value. When flown in "automatic," translation is unchanged but attitude commands are cut off by control electronics at a preselected rate which is derived from a rate gyro signal. When the control is returned to neutral, the control electronics sense an error signal and fire appropriate jets to stop the rotation. This same control operation provides attitude hold about all three axes. In the Gemini AMU, the control electronics were pulse-ratio modulated to provide both pulse rate and pulse duration proportional to the error signal, thereby producing a proportional effect with simple off-on thrusters. Manual control was configured to bias the control electronics by the full command rate. This system could be changed to "proportional-rate" control by adding proportional hand controllers. An integrating circuit is employed in the control electronics to convert the rate gyro signal to a deadband angle.

The Gemini AMU system was based on the assumption that the astronaut could frequently update his attitude reference by visual cues. Total drift in the system could be as high as 6 deg/min. This system characteristic posed no problem in the simulations, and limit cycle operation was never a factor. The AMU limit cycle periods of 30 to 60 seconds far exceeded the time between commands.

CMG CONTROL CONCEPT

An AMU can be rotated by changing the angular momentum in inertia wheels or by rotating the angular momentum vector of gyroscopes. Rotating the momentum vector, the control moment gyro concept, is most adaptable to an AMU application and is the only momentum exchange system that will be discussed in this paper.

In the previous section, it was noted that a gyroscopic momentum transfer system is completely additive to the required mass expulsion system. It must, therefore, have significant advantages to warrant its consideration as part of an AMU system. CMGs have four significant advantages which might motivate adding them to an AMU system, depending on the specified mission.

Fuel Economy - Cyclic Maneuvers

The "fuel" consumed by a CMG system is electrical energy. Total energy required, after initial "run-up," is a function of windage, bearing friction, gyro motor efficiency, inverter efficiency, and the loading and efficiency of servomotors used to torque the gyro gimbals. Energy associated with maintaining gyro speed is virtually independent of the number of times the gyros are used to generate and subsequently negate equal and opposite control torques. A system can be devised using manual gimbal torquing and free gimbal response for stabilization, thereby eliminating electrical gimbal torques. Given a mission with a specified duration, AMU system moments-of-inertia, angular rates required and total number of cyclic attitude maneuvers, and given the current state of the art in CMG and battery technology, a CMG system weight can be determined. Given a system geometry from which thruster moment arms can be derived, and given a propulsion specific impulse and tankage weight per pound of fuel, the weight of an all-jet system can be determined. Obviously, the system performance point at which CMGs become weight effective is sensitive to the effective specific impulse (I_{sp}); i.e., fuel and tankage weight/ total impulse. For instance, stored high-pressure gas has an I_{sp} around 60 seconds, and tankage weight at least twice the gas weight for an effective I_{sp} of 20 seconds. Hydrazine diluted 25% to reduce plume temperatures has an I_{sp} of 175 seconds and a tankage weight of one-third the fuel weight for an effective I_{sp} of 130 seconds. The crossover point is also sensitive to the state of the art in CMG design which is currently evolving more rapidly than monopropellant technology.

Fuel Availability

Electrical "fuel" can be derived from solar energy, atomic power plants, and fuel cells, all of which yield high total energies for the total weight orbited. None of these energy sources are applicable directly to an AMU, but in a multi-mission system, the parent space-craft might use them to recharge the AMU batteries.

High Pointing Accuracy

By precise control of the torque, applied to a CMG gimbal, the total momentum transfer can be precisely controlled. In comparing a CMG system to an all-jet system, the controllability of momentum transfer is compared to the jet's minimum-pulse-bit. For instance, in the Gemini AMU the minimum impulse from a peroxide thruster was 0.05 lb-sec. Two thrusters were fired consuming approximately 0.02 pound of fuel and resulting in angular rates from 0.16 to 0.29 deg/sec. This results in a practical limit on the size of the limit cycle deadband.

High pointing accuracy has not been required on AMU missions defined to date.

Absence of Expelled Products

If a mission requires rotational maneuvers without expelling gas, such as the Apollo Telescope Mount the CMG is an obvious choice. No AMU missions of this type have been specified to date.

Another factor that may favor the addition of CMGs to an AMU system is the possibility of reducing total jet thrust requirements, to the point where a less toxic propellant or a propellant with lower plume temperatures may be used. Factors generally unfavorable to adding CMGs include increased complexity, gyro cooling requirements, and the time and power required to run-up the gyros.

CANDIDATE CMG SYSTEMS

Design Requirements

In conducting this study, which was directed toward first determining the momentum storage control system configuration, and then its mechanization and integration into an existing AMU package, a study plan was developed as shown in Figure 1. The study commenced with a definition of control requirements and physical constraints. These definitions included:

1. Astronaut Body Inertias

Roll Axis - 22.0 slug ft^2

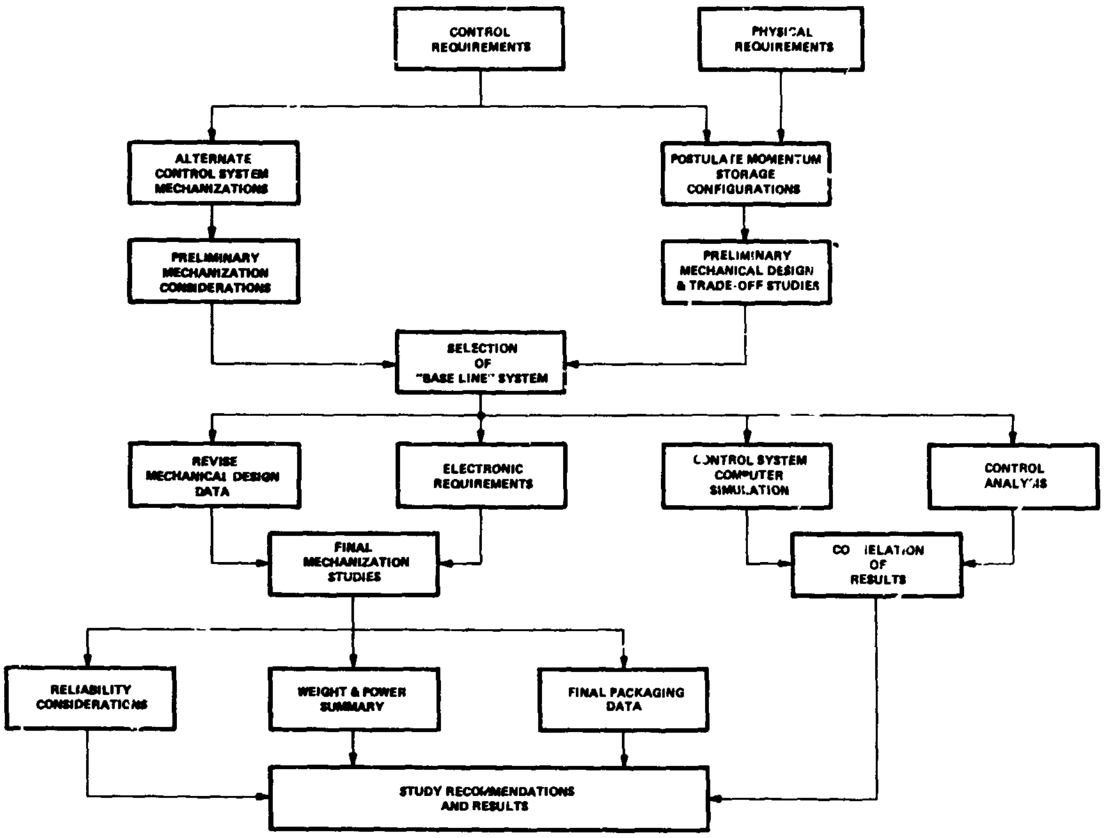

FIG. 1 - AMU/CMG STUDY PLAN

Pitch Axis - 22.0 slug ft^2

Yaw Axis - 7.0 slug ft^2

2. Desired Astronaut Body Rotational Rates

   (Augmented control via reaction jets will always provide a 15 deg/sec rotational rate)

   Maximum - 15 deg/sec about each axis

   Minimum - 5 deg/sec about each axis

3. Desired Astronaut Body Acceleration - 10 deg/sec^2

4. Control Stick Travel - ± 5 deg (all axes)

5. Astronaut Body Drift Rate (Stabilization Mode) $\leq$ 0.1 deg/sec

6. AMU/CMG System Weight Allotment - 40 lb max

7. AMU/CMG System Volume Assignment - 7.0 inches x 7.0 inches x 18.0 inches

8. Mission Time - 4 hours

9. Available Power (nominal) - +28.0 Volt DC

10. Power Regulation Range - 24.0 - 32.0 Operate to Specifications; 22.0 - 34.0 Transient Operation

11. Power Penalty Factor - 15 watt-hr/lb

12. Reliability Goal - Not assigned, but must be consistent with 0.995 for overall AMU Package

13. Operating Temperature Range - +20°F to +120°F

14. Operational Life - 300 hours

15. Orbital Workshop Atmospheric Composition - 20% to 100% O_2

16. Orbital Workshop Atmospheric Pressure - 7.5 psi

Momentum Storage Configurations

Utilizing the astronaut body inertia data and the desired rotational rates, seven control moment gyro configurations were postulated for possible AMU application. These configurations employ both single and double gimbal control moment gyros, and were selected without any restrictions on cross-coupling effects or whether electrical or mechanical gimbal torquing was to be used. The candidate momentum storage configurations are presented in Table 1. It is significant to note that all configurations which employ double gimbal control moment gyros also use and/or require an electric drive for gimbal torquing. This is due to the difficulty encountered in transmitting torques across a gimbal pivot mechanically. Configurations which employ single gimbal control moment gyros can utilize either mechanical or electrical gimbal torquing. In addition, they can provide gyroscopic control torques to the astronaut body when used in the "free mode" where the normal gimbal torquing mechanism is deactivated. These configurations can provide the desirable feature of passive stabilization.

TABLE 1 - MOMENTUM STORAGE CONFIGURATIONS

Configuration	Controlled Axes	Mechanical Or Electrical Drive	Column A H/gyro ft-lb-sec		Column B H/gyro ft-lb-sec	
			5°/sec	15°/sec	5°/sec	15°/sec
1. 6 SGCMG	2 X	M,E.	1.15	3.46	1.15	3.46
	2 [illegible]		1.15	3.46	1.15	3.46
	2 Z		0.36	1.10	0.36	1.10
2 3 SGCMG	2 XY	M,E.	1.15	3.46	2.00	6.00
	1 Z		0.73	2.20	0.73	2.20
3. 4 SGCMG	2 XY	M,E.	1.15	3.46	1.63	4.89
	2 Z		0.36	1.10	0.36	1.10
4. 3 DGCMG	3 XYZ	E	0.73	[illegible]	[illegible]	3.33
5. 2 DGCMG	2 XYZ	E	1.15	3.46	1.67	5.00
6. 2 SGCMG on a Single Gimbal Platform	XYZ	E	1.15	3.46	1.67	5.00
7. 1 DGCMG	1 XY	E	2.30	6.92	3.26	9.78
2 SGCMG	2 Z		1.15	4.56	2.32	6.96

NOTE: 20% increment in H provided to accommodate ±60° gimbal angle restriction. Values in the tables are computed based upon the parameters:

$$I_x = I_y = 22 \text{ slug ft}^2 \qquad I_z = 7 \text{ slug ft}^2$$

$$\omega_x\ (\max) = \omega_y\ (\max) = \omega_z\ (\max) = 5°/\text{sec},\ 15°/\text{sec}$$

The momentum storage capacity that is required of the gyros in each configuration was calculated using two approaches. The results shown in Column A assume that control is exercised about one axis at a time. The values listed in Column B are independent of prior control action and counterpart momentum storage about any other control axes. The calculations, in both cases, are based on the desired maximum and minimum body rates, and allow a 20% excess in storage capacity which is associated with the desirability to restrict the gimbal angles to ± 60 degrees.

Control System Mechanizations

To allow a consideration of both mechanical and electrical gimbal torquing schemes, and to provide a passive attitude stabilization mode, a control moment gyro configuration compatible with these objectives was tentatively selected for the AMU/CMG system. This configuration employs a scissored pair of single-gimbal CMGs about each control axis (configuration 1 in Table 1).

The AMU/CMG system should be able to perform three distinct operations:

1. Control body rates
2. Hold a body attitude position
3. Eliminate the torquing effect of the CMGs by holding the SGCMG scissor pair at a null or caged position.

Five basically different implementations meeting these three requirements are appraised for relative complexity and system behavior.

1. All-Mechanical System

The all-mechanical system, has three modes of operation: passive mode (attitude hold), command mode (command body rates), and caged mode.

To achieve attitude hold, the control stick is left at its null (detent) position, and the SGCMG scissor pair tends to move in such a manner as to reduce body rate along its controlled axis. Should the scissor pair reach its maximum momentum storage capacity (60 degrees), the reaction jets fire, and the scissor pair is desaturated, moving away from its 60-degree position.

When the pilot moves the control stick out of its detent on a specific axis, the cabling is clutched into the scissor pair controlling that axis, and the pair is manually positioned by the control stick. If the scissor pair reaches its saturated position, the jets will fire until the control stick repositions the SGCMG pair off its stop (60-degree position). A thumb switch is also provided to declutch the control stick from the scissor pair at any control stick position. This declutching allows the control stick to be brought to null without moving the scissor pair.

When the pilot commands caging, the scissor pair is mechanically torqued to its null position and it is mechanically held in that position.

2. All-Mechanical Analog System

This electrical system is similar to the all-mechanical system. However, a position servo and gear train are used to replace the mechanical linkage to the control stick. It is also used as a substitute for the dynamic feedback in the all-mechanical system.

With the control stick at its null position, the scissor pair is declutched from the torquer motor and gear train; system behavior is then identical to the all-mechanical system in the passive mode.

Caging is accomplished electrically by means of a solenoid and caging cam. Note that this and other caging schemes are applicable to most of the systems surveyed here, and may be interchanged.

3. Torque Balance System

The torque balance system can be conveniently described by considering three modes of system operation.

For an attitude hold operation, the control stick is at its null position, the scissor pair is declutched from the torquer motor and gear train, and system behavior is identical to the all-mechanical system in passive mode.

When the control stick is moved off its null on a specific axis, the scissor pair controlling that axis is clutched to its torquer motor through a gear train. The torquer motor drives the scissor pair until the input command torque just equals the gyroscopic torque. Since the gyroscopic torque is proportional to body rate, a given input torque is used to command a particular body rate.

If the control stick position is proportional to input torque, a thumbswitch declutching mechanism is not required because control stick position is a direct measure of body command rate; when the stick is at null, the rate is commanded to zero.

Caging is accomplished by using the SGCMG scissor pair position angle as an input signal to the torquer motor. This commands the scissor pair to null.

4. Combination Torque Balance/All-Mechanical System

Because the torque balance and all-mechanical systems have many of the same components, by providing the appropriate switching, one system may be readily converted into the other. This may be helpful for a laboratory comparison of the two systems.

5. Actively Stabilized System

In this electrical system, the input to the servomotor is the difference between the control stick and rate gyro outputs. The control stick commands body rates directly proportional to control stick position. The high forward loop gains allow zero body rate to be commanded accurately, eliminating the need for a clutch in passive mode operation.

Caging is achieved by grounding the position servo input. This commands the scissor pair to null.

6. Combination Jet and CMG System

When the stick is at null, the system is in the passive mode as described in the all-mechanical system. When the control stick is moved off the null position, the scissor pair is electrically caged and reaction jets are used to obtain commanded rates.

Table 2 provides a comparison of the candidate systems by emphasizing system implementation differences and pilot interface effects. It also outlines the relative system complexity and the difficult design areas associated with each system mechanization. From these comparisons, the torque balance system was selected for the AMU/CMG application based on system complexity, implementation requirements, pilot interfacing, system performance and power requirements.

COMPONENT OPTIMIZATION

Introduction

Studies were conducted on a number of gimbaled rotor arrangements, to assist in the selection of a control moment gyro mechanical configuration. The momentum storage range study covered units capable of developing reasonable body acceleration torques in a package size suitable for a back-pack application.

Mechanical Design

There are many factors which influence the selection of the AMU/CMG prototype system. These include availability of state-of-the-art hardware, cost, maintainability, and incorporation of features to permit adjustment of system gains and performance parameters.

TABLE 2 - COMPARISON OF CONTROL CONCEPTS

		All Mechanical	All Mechanical Analog	Torque Balance	Combination	Active	Combination Jet and CMG
Pilot Interface Effects	Stick	1. Torque feedback due to vehicle rate 2. Large stick forces when stick travel is small 3. Stick stops when CMGs reach stops 4. Declutch button	1. Fixed travel and Torque gradient 2. Declutch button	1. Fixed travel and Torque gradient	1. Fixed travel and Torque gradient	1. Fixed travel and Torque gradient	1. Fixed travel and gradient 2. Commands jet firing
		1. Approximately proportion rate to stick input 2. Passive stabilization limited by gyro friction and damping 3. Difficult control with more than one axis rate at a time	1. Approximately proportional rate to stick input 2. Passive stabilization limited by gyro friction and damping 3. Difficult control with more than one axis rate at a time	1. Proportional rate to stick input 2. Passive stabilization limited by gyro friction and damping 3. Difficult control with more than one axis rate at a time	1. Either strictly or approximately proportional rate to stick 2. Passive stabilization limited by gyro friction and damping 3. Difficult control with more than one axis rate at a time	1. Proportional rate to stick 2. Active stabilization limited by rate gyro and integrator 3. No difficulty controlling more than one axis rate at a time	1. Proportional rate to stick with some finite resolution 2. Passive stabilization limited by gyro friction and damping 3. Some jerkiness with jets 4. No difficulty controlling more than one axis rate at a time
Implementation Differences	Mechanical Design Problems	Yes	No	No	No	No	No
	Clutch	Mechanical	Electrical	Electrical	Electrical	None	None
	Motor and Amplifier	No	Yes	Yes	Yes	Yes	No
	Caging Mechanics	Yes	No	Yes	Yes	No	Yes
	SGCMG Configuration Required	Yes	Yes	Yes	Yes	No	Yes
	Electrical Switching	None	Small	Small	Most	Small	Small
	Jet Pulsing Circuit	Probable	Probable	Possible	Possible	No	Possible
	Rate Gyros	No	No	No	No	Yes	Yes (in RC AMU)
	RC AMU Req's.	No	No	No	No	No	Yes
	Tachometer	No	Yes	No	Yes	Yes	No
	Integrator	No	No	No	No	Yes	No
	Demodulators	No	Yes	Yes	Yes	Yes	No (other than RC AMU)
	Threshold Circuits	No	Yes	Yes	Yes	*	Yes (in RC AMU)

The selection of rotor proportions, size, weight, speed, and material involves the interrelation of several factors. To minimize weight for a required momentum, a large diameter rotor, with a corresponding large radius of gyration operating at high speed is indicated. However, rotor size is restricted by package size considerations. Figures 2 and 3 present power losses for gyro rotors operating in concentric and cylindrical housings. The values indicated are conservative, since worse case gas pressure and bearing friction were assumed. Power loss is not only proportional to speed but increases exponentially due to windage drag. Based on these factors, a more favorable equivalent weight may be achieved by selecting a lower operating speed in spite of the somewhat heavier rotor required.

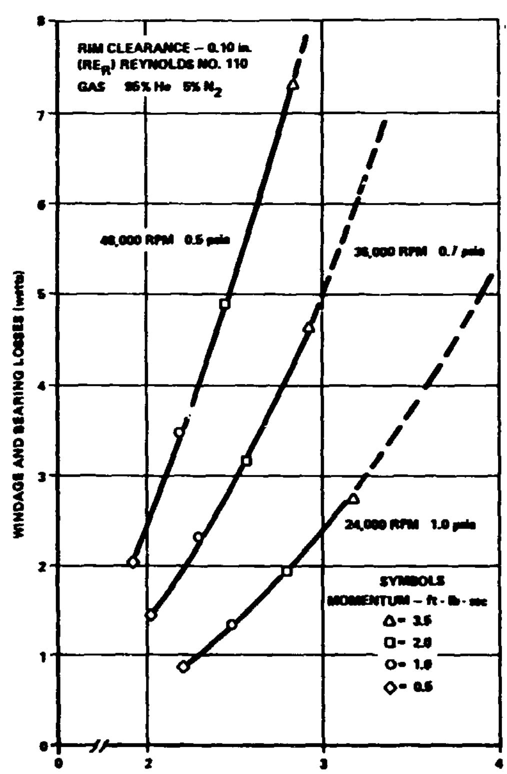

FIG. 2 - WINDAGE AND BEARING LOSSES (MAX.) - CONCENTRIC HOUSING SINGLE ROTOR

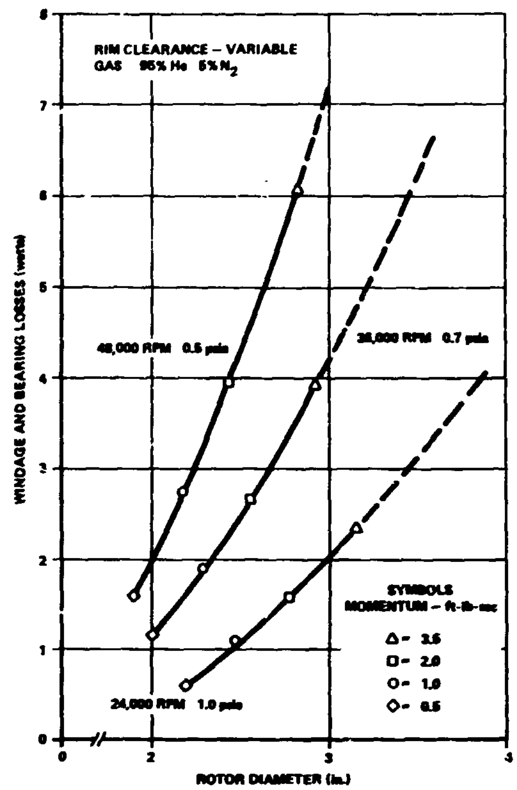

FIG. 3 - WINDAGE AND BEARING LOSSES (MAX.) - CYLINDRICAL HOUSING SINGLE ROTOR

The induction motor chosen operates slightly below synchronous speed, the actual value of speed being determined by the particular motor characteristic. Motor operating speed determines the required rotor inertia which leads to a rotor diameter. Table 3 and Figure 4 present the relationships of weight to momentum for typical rotors with safety factor values of 2.5 and 4.0. The curve representing a theoretical safety factor of 1 is included for reference only, since it represents the limiting case based on published yield stress values for the selected rotor material. The

TABLE 3 - PARAMETRIC DATA - CMG ROTORS

		Stress-Limited-Value Given For Information Only											
R in.	W lb	S.F. (Stress)	H ft-lb-sec	H/W Ratio	rpm	S.F. (Stress)	H ft-lb-sec	H/W Ratio	rpm	S.F. (Stress)	H ft-lb-sec	H/W Ratio	rpm
1.0	1.13	1:1	1.37	1.21	88,250	2.5:1	0.86	0.762	55,250	4:1	0.679	0.601	43,700
1.1	1.51						1.26	0.840	50,300				
1 2	1.95						1.79	0.918	46,200				
1.3	2.49		3.92	1.57	67,900		2.46	0.989	42,500		1.95	0.784	33,700
1.4	3.11						3.32	1.07	39,500				
1.5	3.82						4.30	1.13	36,900				
1.6	4.63		9.00	1.95	55,200		5.63	1.21	34,500		4.44	0.960	27,300
1.7	5.56						6.96	1.25	32,500				
1.8	6.59						9.03	1.37	30,700				
1.9	7.76		17.90	2.31	46,500		11.20	1.44	29,100		8.86	1.1[illegible]	23,000
2.0	9.05						13.80	1.54	27,700				
2.1	10.46						16.70	1.60	26,300				
2.2	12.03		32.10	2.66	40,200		20.10	1.67	25,100		15.90	1.32	19,850
2.3	13.75						24.20	1.76	24,000				
2.4	15.58						28.50	1.83	23,000				
2.5	17.62		53.70	3.04	35,400		33.60	1.91	22,100		26.50	1.50	17,500

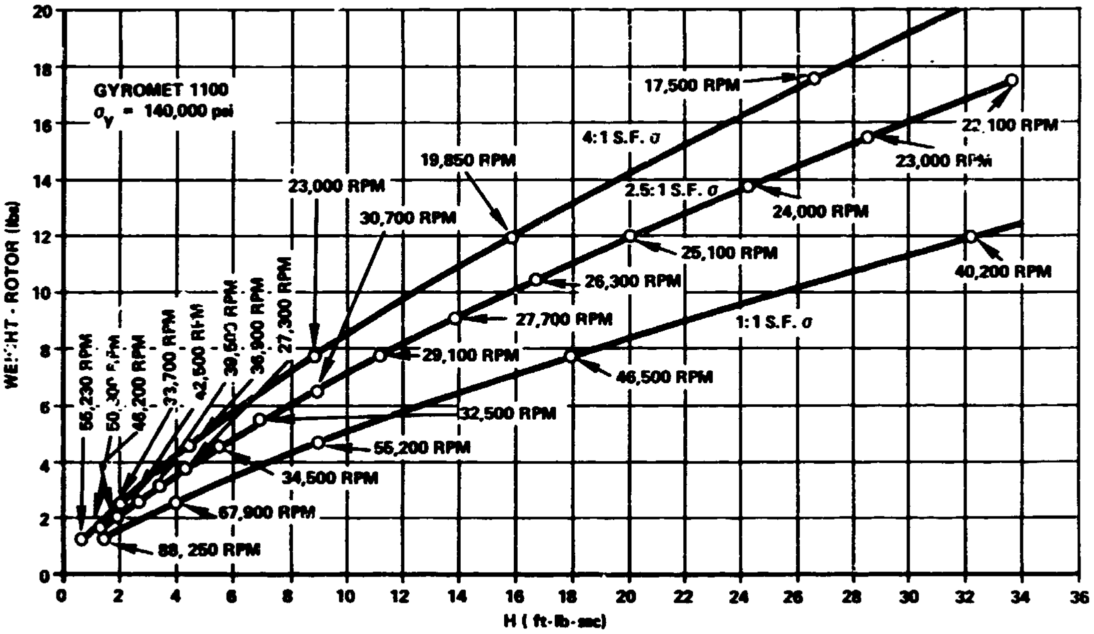

FIG. 4 - ROTOR WEIGHT VERSUS MOMENTUM FOR VARIOUS STRESS SAFETY FACTOR

proportions for the rotors used are presented in Figure 5.

The single axis CMG assembly presented in Figures 6 and 7 employs a scissored pair of momentum rotors, spin motors, bearings, gimbal housings, and flex-lead devices supported in a common mounting frame by means of ball bearings. A servo gear-box forms a portion of this frame, and contains the speed-reduction gear train and mounting pads for motor and clutch. A permanent-magnet DC motor torques the gimbals through a four-pass gear train and electromagnetic clutch.

An eddy-current viscous damper is attached to the clutch. The damper and clutch rotate at approximately four times gimbal velocity, thus reducing component size. Gimbal angle ($\pm$60 degrees) readout information is provided by a transducer mounted on one pivot.

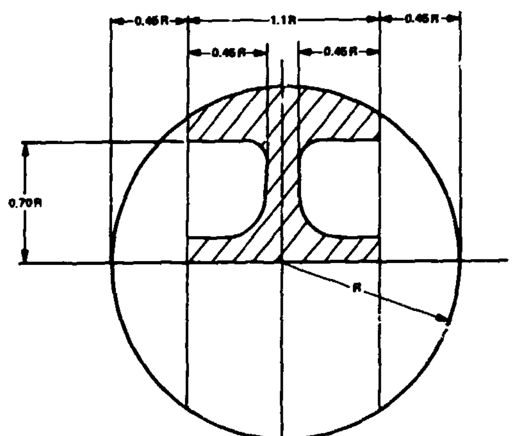

FIG. 5 - ROTOR PROPORTIONS FOR SMALL CMG

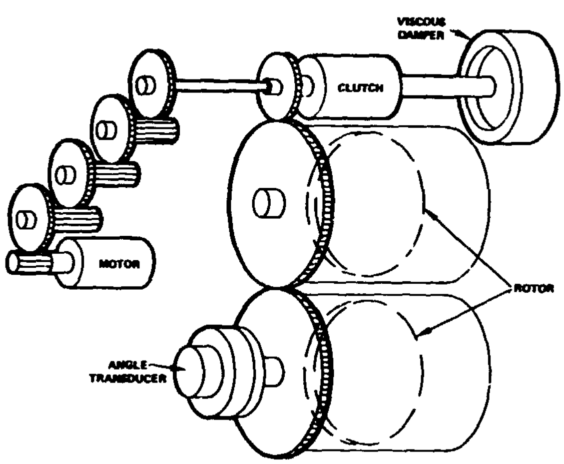

FIG. 6 - MECHANICAL SCHEMATIC SCISSORED PAIR

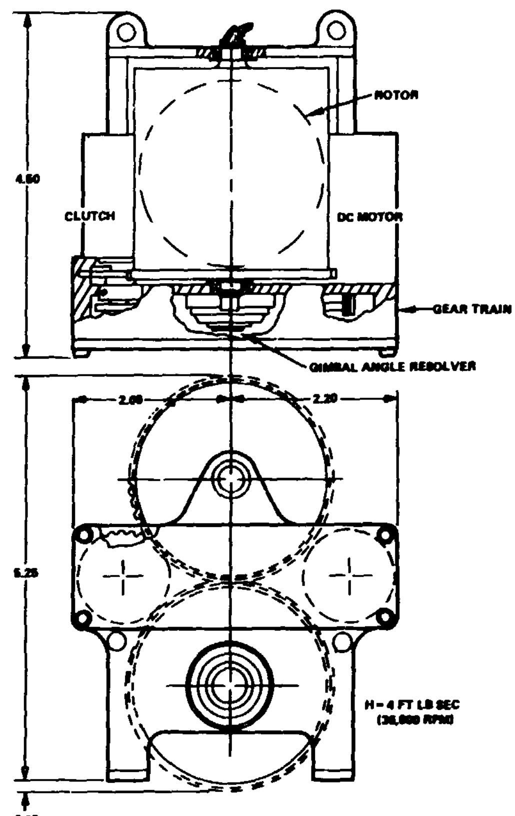

FIG. 7 - LAYOUT - CMG SCISSORED PAIR

Weight Studies

Weight studies were performed on various assemblies involving gimbaled gyros with single degree-of-freedom, scissored pairs of gyros, and double-gimbaled units with two degrees of freedom.

Weight versus momentum curves for various assemblies are presented in Figures 8 and 9. In each case, an assembly consists of the gimbaled rotor housing, support structure and where required, clutch, rate feedback tachometer, viscous damper, and slip ring assembly. The weight shown on the curves is for a single unit assembly. Weight of a given three-axis system will therefore include the weight of the several units required to make up the selected system. In some cases, rpm curves have been projected into higher momentum regions for comparison purposes. Windage and friction losses, or insufficient rotor stress safety margin, may make a certain radius-

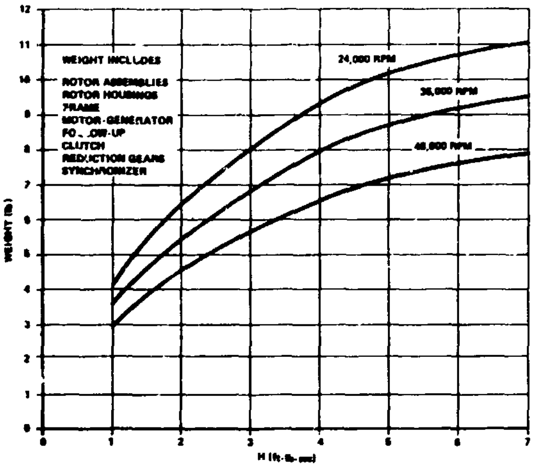

FIG. 8 - WEIGHT OF GYRO ASSEMBLY PER PAIR VERSUS MOMENTUM PER PAIR (SINGLE AXIS)

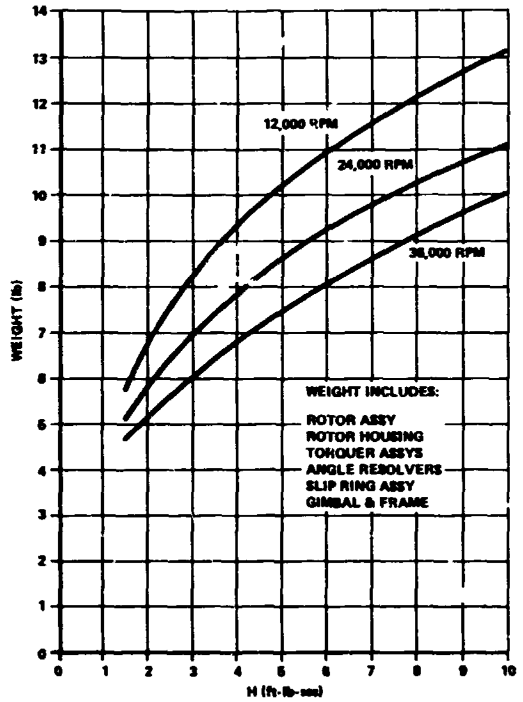

FIG. 9 - WEIGHT VERSUS MOMENTUM STORAGE - DOUBLE GIMBAL CMG

speed combination in a projected region unattractive. For example, in Figure 9 the 36,000 rpm curve is projected out to include a momentum of 10 ft-lb-sec. The stress safety factor for this 3.5-inch diameter rotor would be 1.8 and the windage and bearing losses, according to Figure 2, would exceed 7.0 watts, thus eliminating this design from further consideration. A control gyro module, consisting of a pair of identical gyro assemblies, whose momentum vectors together effect attitude control about a single body axis, is shown in Figure 7 and schematically, in Figure 6. Figure 8 presents weight versus momentum storage values for the scissored pairs shown. It should be noted that momentum values reflect the momentum stored by a pair of gyros. A value such as 4 ft-lb-sec, therefore, is representative of a pair, each of which produces 2 ft-lb-sec.

CMG SYSTEM SELECTED

Baseline System

Based on the candidate systems considered and the component optimization studies, a system which utilizes six single-degree-of-freedom gyros, in scissored pairs about each of the control axes, and which is mechanized in the standard torque balance manner was chosen as the baseline system. In making this choice, control moment gyro configurations, which employed double gimbal devices, were eliminated because they did not permit the desirable feature of passive attitude stabilization. In addition, the double gimbal gyro configurations, in general, employ larger momentum units and these would have resulted in a package design which exceeded the specified outlines. The scissored-pair concept was chosen from the remaining configurations because of its inherent low cross-coupling between control axes.

Electronic Requirements

The electronic requirements for the chosen AMU/CMG system are presented in Figure 10. The block diagram consists of the following electronic circuits:

Motor Drive Amplifier

Gain Amplifier

Demodulator

Pulsing Circuit

DC Power Supply

AC Power Supply

Static Inverter

The only power excitation needed by the CMG system is +28 VDC. Gyro spin motor excitation is 52 volts peak-to-peak, 3-phase and 600 Hz. This excitation is generated by the CMG inverter. The +12 VDC and -12 VDC excitation, for the integrated circuit operational amplifiers, is generated from a regulated power supply in the CMG electronics box. AC excitation, for resolvers, and the keying signal for the demodulators, is tapped off the oscillator in the inverter and amplified through a drive amplifier to provide sufficient driving power. The power losses of the electronic circuits are presented in Table 4. The construction of the CMG Electronics will be potted (epoxy encapsulated) modules employing soldered printed circuit board construction. Module interconnections are made via a wiring harness. Table 5 gives size and weight information for the proposed circuits.

The output of the gimbal angle limit switches interfaces with the jet thruster logic electronics which controls the operation of the jet thrusters. The jet thrusters are used for desaturation of the CMG system when

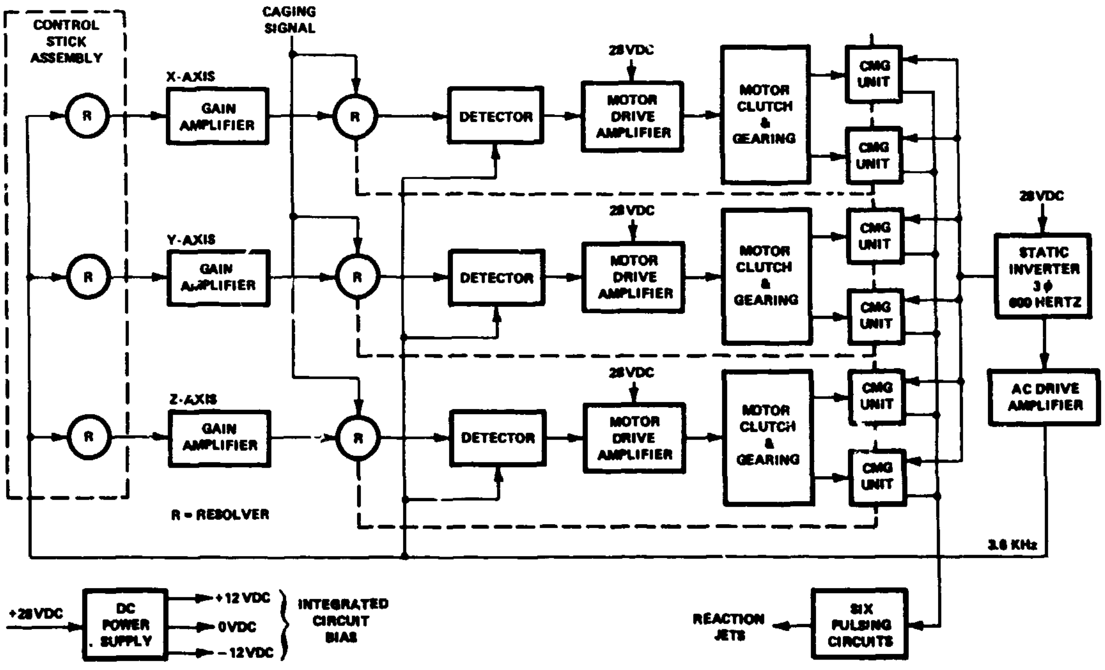

FIG. 10 - FUNCTIONAL BLOCK DIAGRAM - AMU-CMG ELECTRONICS

TABLE 4 - CMG ELECTRONIC POWER LOSSES

Circuit	Quantity	CMG Spin Up No Command	CMG - RUN No Command	CMG - RUN Maximum Commanded Rate
Gain Amplifier	3	0.15 W	0.15 W	0.15 W
Demodulator	3	0.30 W	0.30 W	0.30 W
DC Power Supply	1	0.50 W	0.50 W	0.50 W
AC Power Supply	1	0.60 W	0.60 W	0.60 W
Pulse Circuit	6	———	———	0.99 W
Motor Drive Amplifier	3	0.30 W	0.30 W	62.37 W
Static Inverter	1	25.00 W	16.00 W	16.00 W
	TOTALS:	26.85 W	17.85 W	80.91 W

TABLE 5 - PHYSICAL CHARACTERISTICS (SIZE AND WEIGHT)

Circuit	Vol. (in^3)	Wt. (lb)	Quantity	Total Vol (in^3)	Total Wt. (lb)
Motor Drive Amplifier	30.0	1.35	3	90.0	4.05
Gain Amplifiers	.84	0.04	3	2.5	0.12
Demodulators	2.0	0.09	3	6.0	0.27
Pulsing Circuits	2.0	0.09	6	12.0	0.54
DC Power Supply	5.8	0.26	1	5.8	0.26
AC Drive Amplifier	6.4	0.29	1	6.4	0.29
Static Inverter	60.0	2.50	1	60.0	2.50
			TOTALS:	182.7 in^3	8.03 lb

the stored momentum is used up. Stick commands and mode switching come from the hand controller. The +28 VDC power excitation comes from either a battery on the backpack or from power supplied through an umbilical from the orbital workshop supply.

The torque balance system chosen is presented in Figure 11 in the passive mode. In this case, the dynamic response of the body rate, with respect to the system, takes the form:

$$\omega_B(s) = \frac{\frac{F H \cos\alpha}{S}}{\left[I_2 S^2 + D_2 S\right] I_1 + H^2 \cos^2\alpha} \qquad (1)$$

where

I_2 = total SGCMG pair gimbal inertia

D_2 = total added system damping

I_1 = body inertia about the controlled axis

H = total momentum storage capability (4 ft-lb-sec)

α = SGCMG scissor pair angle

F = total friction seen at SGCMG pair gimbal

S = LaPlace transform variable

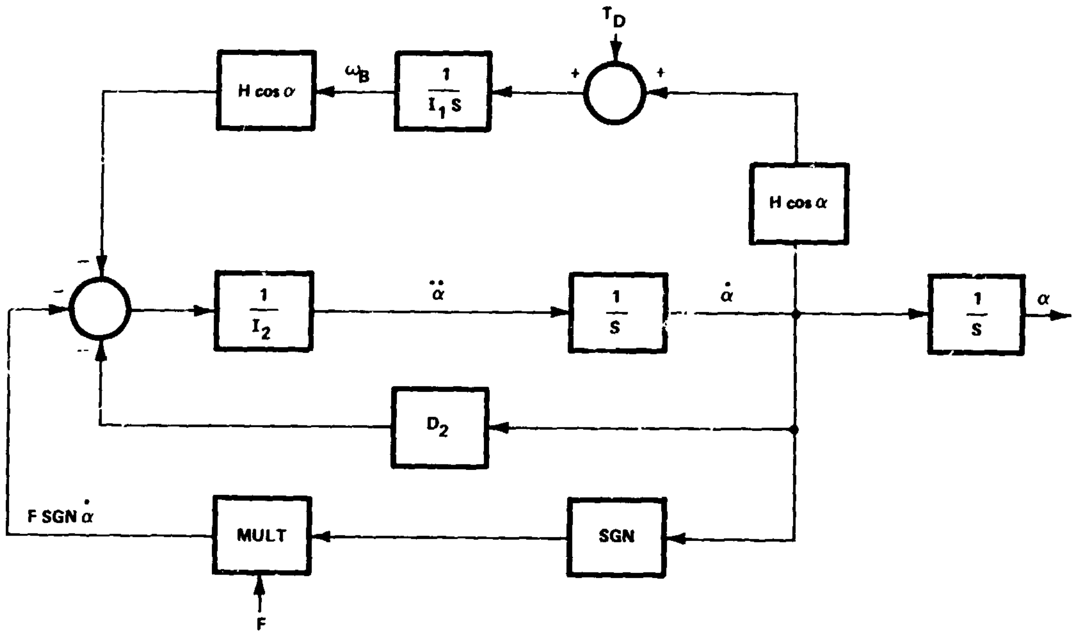

FIG. 11 - AMU/CMG TORQUE BALANCE SYSTEM - PASSIVE MODE

D_2, the damping added to the system in passive mode, is chosen to provide critical damping when the scissor pair is near its stop. Without damping a steady-state limit cycle exists.

The steady-state body drift rate level is calculated from the above equation using the final value theorem:

$$\omega_B(t) \text{ steady-state} = \frac{F}{H \cos \alpha} \quad (2)$$

If a constant disturbance occurs, the steady-state rate is increased by

$$\frac{D_2 \alpha}{H \cos \alpha}$$

In the torque balance system, the gyroscopic feedback torque is $\omega_B \cos \alpha$. In steady-state, this torque is balanced by the input control torque. However, $\omega_B \cos \alpha$ is not monotonic but increases with α; beyond some value of α, determined by initial conditions, cos α decreases faster than ω_B increases with α. For these conditions, the control torque must be decreased to increase body rate. This represents an uncontrollable situation in which the scissor pair tends to ride into its stop once α increases above a certain angle.

The system may be made controllable by introducing a variable gain in the forward loop of the form K cos α. This forces the control torque to decrease with α as the gyroscopic feedback torque decreases with α. In steady-state, the torque balance equation (neglecting friction) becomes:

$$K \delta s = H \omega_B \quad (3)$$

where

δ s = control stick position

K = constant gain

The body rate is now directly proportional to the control stick position.

It can be shown that a considerable error can be tolerated in measuring cos α for use in the forward gain. If the LaPlace transform of the linearized dynamic equations derived from Figure 12 is taken, the result is:

$$\omega_B(s) = \frac{K_R K_S A K_T N H \cos^2 \alpha \, \delta_s(s)}{\left[\left(J_m N^2 + I_2\right) S^2 + \left(D_m N^2 + D_2\right) S\right] I_1 + H^2 \cos^2 \alpha} \quad (4)$$

For the parameters employed in the system, this represents an overdamped response whose major time constant is determined by N, the gear train ratio. The smaller N is, the faster the time response of the system. However, the smaller N becomes, the higher the power dissipation in the torquer motor. For N = 68 (close to the lower limit placed on the gear ratio due to power dissipation), it takes one-third of a second for the system to reach 90% of its commanded rate in pitch and roll.

In the caged mode, the SG CMG scissor position angle is used as an input to the torquer amplifier to torque the scissor pair to null. The torquing gain is chosen large enough to torque the scissor pair within 0.5 degree, when the body rate is at 15 deg/sec. When the scissor gets to this angle, it is mechanically constrained, so an underdamped system response is considered acceptable.

A digital computer simulation of the AMU/CMG torque balance system was performed. Figure 13 presents the system response to a control stick "step" input of 2.5 degrees. This corresponds to a commanded 7.5 deg/sec body rate. Due to the high friction level in this mode, there is a body rate uncertainty of approximately ± 0.86 deg/sec.

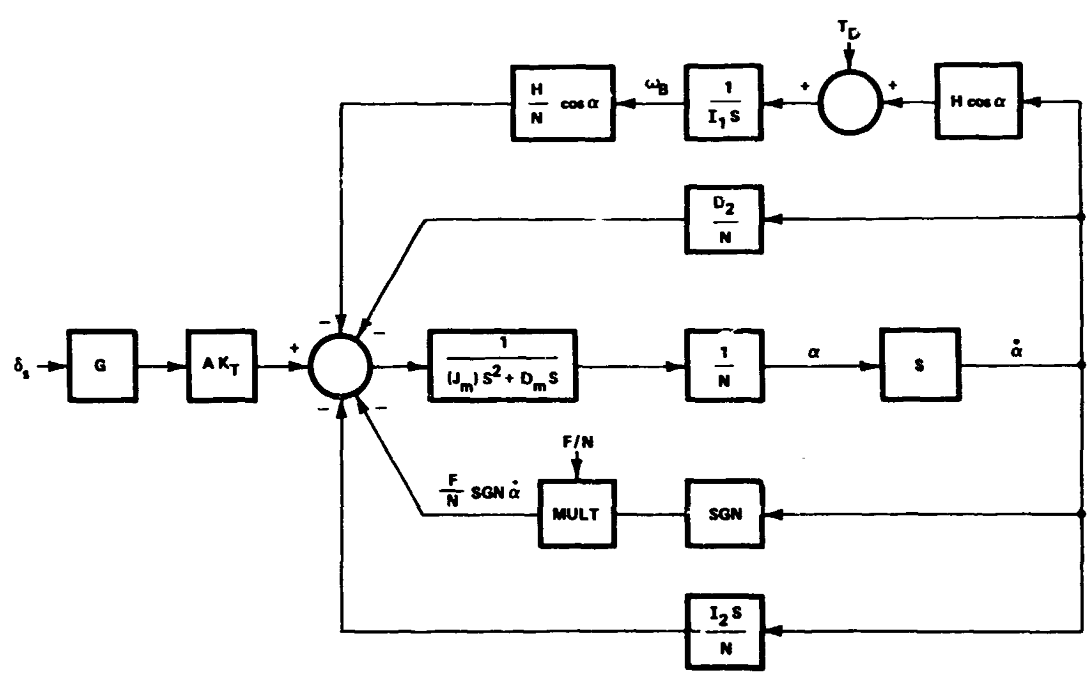

FIG. 12 - AMU/CMG TORQUE BALANCE SYSTEM - ACTIVE MODE

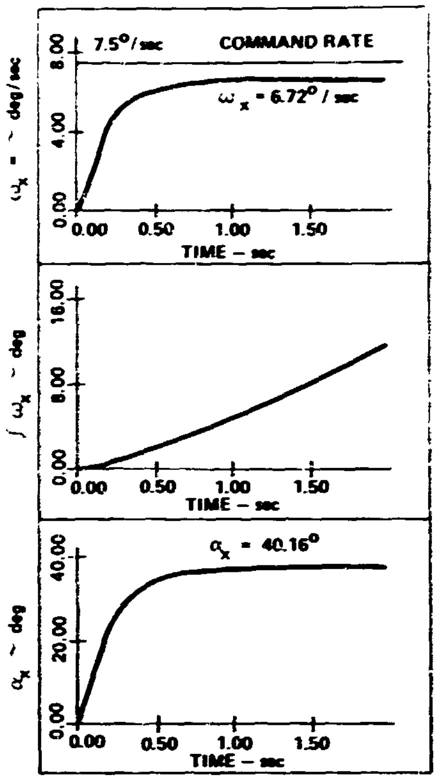

FIG. 13 - AMU/CMG TORQUE BALANCE SYSTEM RESPONSE: 7.5 DEG/SEC RATE COMMAND

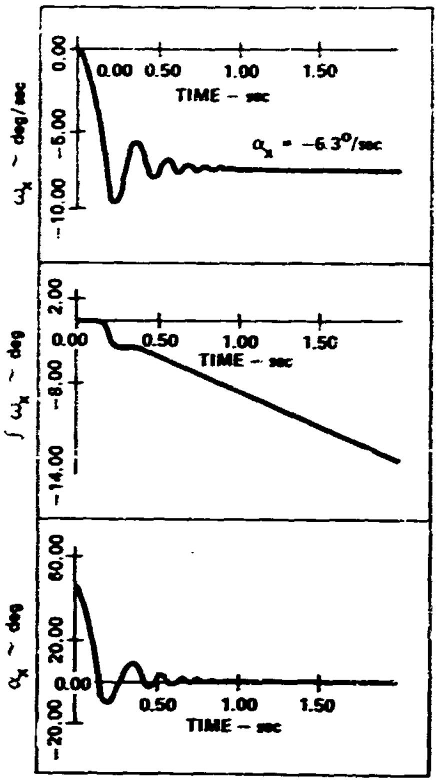

FIG. 14 - AMU/CMG TORQUE BALANCE SYSTEM RESPONSE: CAGING

Figure 14 illustrates the body rate experienced by the astronaut, if he should cage the gyros when he is not holding onto a massive object, or if he uses reaction jets for attitude control. For simplicity, only one axis (X) is shown. Because of the low damping, the simulated system takes approximately one second to reach steady state. In the actual system, the scissor pair will be mechanically held at zero as soon as it reaches this point.

The major deficiency of the torque balance system in command mode is its rather poor body rate resolution. The high friction in this mode can lead to a body rate differing from the commanded rate by as much as 1.2 deg/sec.

An improvement in rate resolution can be achieved by adding a high frequency sinusoidal signal to the torquer amplifier input. This signal reduces the effective friction at the gimbals of the scissor pair.

The major reason for the body rate resolution problem is the low forward loop gain due to friction in the torque balance system. The compliant system, shown in Figure 15, has a high forward loop gain. Because the servomotor and gear train are in this high gain loop, the motor and gear train friction is effectively removed.

In the torque balance system, the gyroscopic torque, equivalent

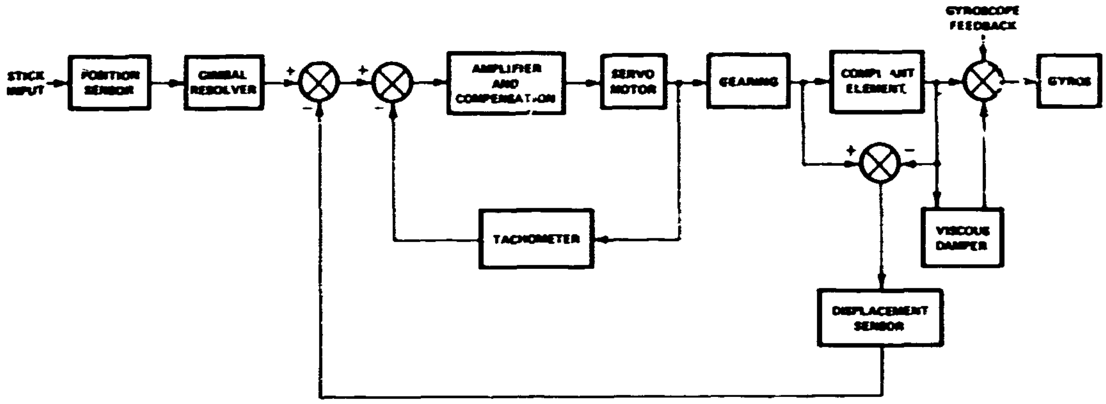

FIG. 15 - AMU/CMG CONTROL SYSTEM - SCHEMATIC DIAGRAM

to a load on the motor, is just balanced by the command torque at the motor input. In the compliant system, the gyroscopic torque is measured by a compliant element, scaled, and compared to the control stick command at the servo amplifier input. The amplifier gain is chosen to make the difference between the command rate and actual body rate arbitrarily small. If the command stick is at null, the body rate is commanded to zero with a close enough tolerance to eliminate the need for declutching the motor and gear train from the scissor gyro pair.

If the body rate resolution associated with the standard torque-balance AMU/CMG system proves to be objectionable to the astronaut, a mechanization utilizing the compliant system features may be employed. This latter system is a bit more complex but it will provide the required body rate resolution.

Reliability

Reliability predictions for the torque balance and compliant systems were made using an applicable computer program. The mean time between failure (MTBF) predictions for the two configurations are:

	MTBF (Hours)	
	Fixed Ground	Space
Torque Balance System	6247	10,410
Compliant System	5959	9931

Packaging

Several possibilities were considered during the course of arriving at a suitable CMG system installation concept. Modular units could be distributed about the AMU packboard or be grouped to form a tightly integrated arrangement. The latter was deemed advisable, since such an assembly could employ point-to-point hard wiring, and simplify checkout and qualification as a subsystem. Sealed construction was chosen in view of the measure of protection afforded electrical and mechanical components, the ability to maintain adequate heat rejection, and attendant safety considerations. In the ultimate application, where an AMU will be exposed to hard vacuum situations, the seal will afford protection to the gear train and bearings, thus reducing lubrication problems, and improving the life of motor commutators.

Figure 16 shows the package construction features. Dimensions for an assembly with 4 ft-lb-sec momentum storage per axis, where rotor speed is nominally 36,000 rpm, are 5.75 x 7.50 x 16.44 inches. With the addition of viscous dampers in each axis, the overall weight will amount to 38.7 pounds.

Weight and Power Summary

The estimated weight of the AMU/CMG system is 38.69 pounds. The system, as presented in Figure 16, consists of: a chassis bedplate; 3 gyro scissored-pair subassemblies with a total angular momentum of 4 ft-lb-sec per pair; inverter module; control electronics; pressure relief valve; electrical connector, and gasket-sealed cover. Table 6 gives a detailed weight breakdown for the AMU/CMG system.

The estimated power drain from the +28 VDC battery is 149.8 watts during spin up mode, and 60.75 watts during CMG run with no command. The peak power drain is 196.41 watts. Power drain includes the power requirements for the inverter, control electronics, gyro spin motors, gimbal drive torquers, and clutches. Table 7 presents a power summary of the CMG system.

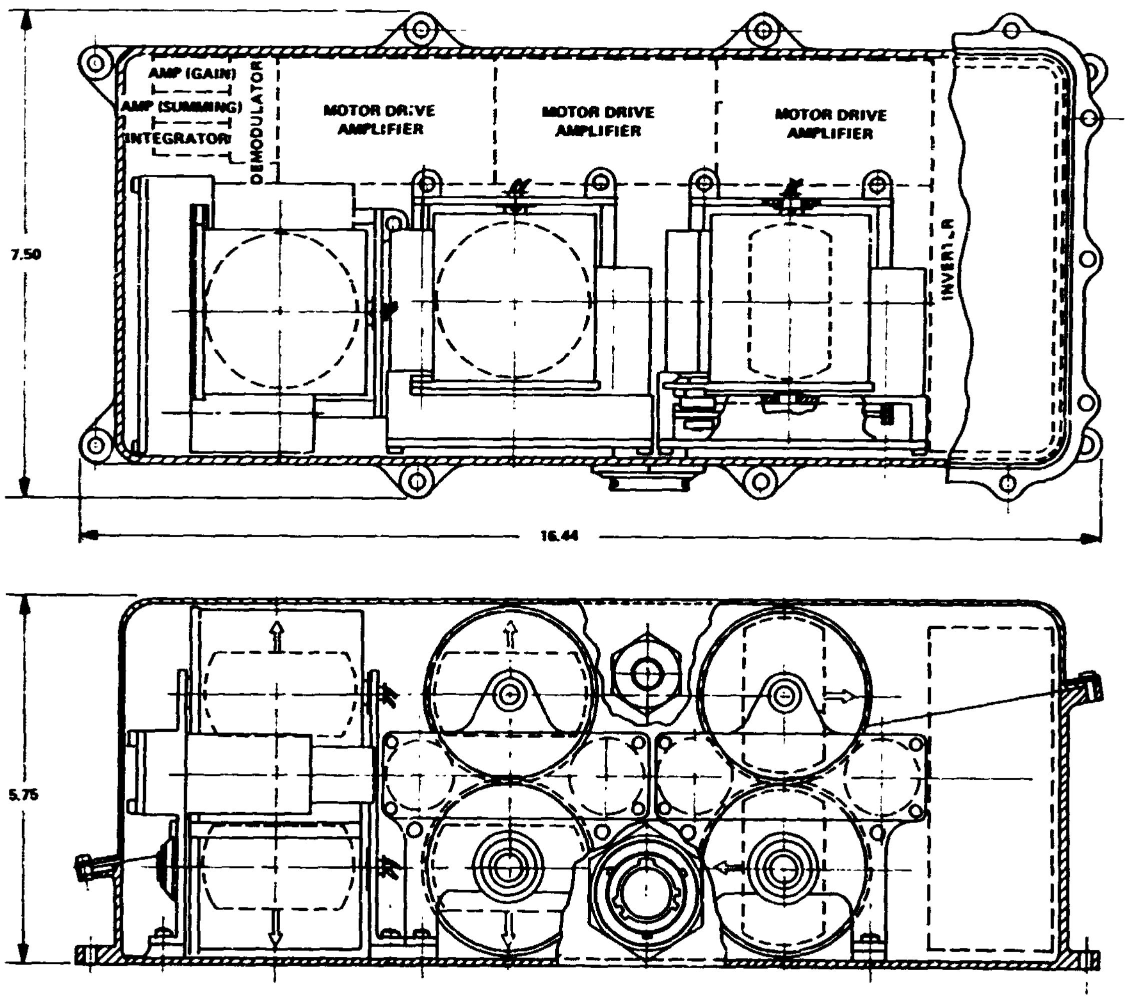

FIG. 16 - AMU/CMG ASSEMBLY

TABLE 6 - AMU/CMG WEIGHT BREAKDOWN

Quantity	Description	Weight
1	Chassis	3.26
1	Cover	1.30
3	Gyro Scissored Pair Assemblies at 8.45 each	25.35
1	Inverter	2.50
1	Control Electronics	5.53
1	Relief Valve	0.10
1	Connector	0.10
—	Hardware	0.25
—	Cables	0.30
		38.69 pounds

TABLE 7 - POWER SUMMARY TABLE (DISSIPATION IN WATTS)

	Operation Modes**		
Quantity	CMG Spin Up No Command	CMG Run No Command	CMG Run Maximum Commanded Rate
6 Gyro Rotor	7.8	15.6	15.6
6 Spin Motor	108.0	27.0	27.0
1 Static Inverter	25.0	16.0	16.0
3 Gimbal Clutch*	---	---	9.[illegible]
3 Gimbal Brake	9.0	---	---
3 Torque Motor	---	---	53.60
3 Gain Amplifier	---	0.25	0.25
3 Demodulator	---	0.30	0.30
1 DC Power Supply	---	0.50	0.50
1 AC Driver Amplifier	---	0.60	0.60
3 Motor Drive Amplifier	---	1.30	52.[illegible]
6* Pulsing Circuit	---	---	3.00
6 Resolvers	---	0.30	0.30
TOTALS:	149.5	60.75	196.[illegible]

*Only three circuits can operate at any one time.

**Operation Modes

CMG SPIN UP

CMG gimbals are mechanically caged using gimbal brake coil in clutch, gimbal drive electronics are turned off, and CMG wheels are accelerated to run speed.

CMG RUN, NO COMMAND

The CMG wheels are at run speed, the hand controller is in its neutral position, and the CMG gimbals are de-clutched allowing the CMGs to be in their free gyro mode.

CMG RUN, MAXIMUM COMMANDED RATE

Maximum commanded rates in all three axis, the CMGs are in gimbal limit stops and three gimbal limit switches activated, three pulsing circuits energized, and the CMG gimbals are clutched.

AUTOMATICALLY STABILIZED
MANEUVERING UNIT
(ASMU)

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SUMMARY: Various types of equipment has been proposed to allow powered maneuvering by an astronaut on Extra Vehicular Activity (EVA) missions. Complete evaluation of these devices on the ground is extremely difficult, if not impossible. The Automatically Stabilized Maneuvering Unit (ASMU) was designed and developed to facilitate comparative evaluation of various powered astronaut maneuvering techniques in a controlled experiment at true zero-g within an earth orbital laboratory. The unit allows evaluation of "fixed-thruster" type maneuvering equipment utilizing three selectable attitude control and stabilization modes; it can also provide propellant, power and flight data instrumentation for evaluation of devices which utilize thrusters that are "aimed" by the astronaut's limbs. The ASMU can operate in a self-contained mode, or with a parent-craft propellant/power umbilical; it can be used by shirt-sleeved and/or space-suited astronauts.

INTRODUCTION

During the past several years many widely different approaches have been proposed for providing an astronaut with a powered maneuvering capability for his Extra Vehicular Activity (EVA) tasks. The definition of operational EVA techniques for future U.S. manned space programs hinges upon true evaluation of these maneuvering unit concepts as to their capabilities and limitations in an absolute, as well as comparative sense. Attempts at performing this evaluation by analytical and ground simulation techniques of various types have met with only limited success. Evidence of unsatisfactory correlation between actual EVA and these analyses/simulations was brought into sharp focus during the Gemini-EVA efforts. As stated in the introduction to Reference 1: "One of the most difficult aspects of developing an extravehicular capability was simulating the extravehicular environment. .... Zero-g aircraft simulations were valuable, but the results of the simulations were occasionally misleading. .... The novel characteristics of the extravehicular environment and the lack of comparable prior experience made intuition and normal design approaches occasionally inadequate."[1]

These experiences clearly indicate the need for a controlled experimental evaluation of the various powered maneuvering unit concepts in a true zero-g earth orbital laboratory. In order to minimize the weight and cost of the hardware needed for such an experiment it is essential that a multi-mode device be utilized which can represent several maneuvering unit concepts without duplication of equipment. Recognition of this need prompted the design and development of the Automatically Stabilized Maneuvering Unit (ASMU).

In order to properly relate the ASMU design features to the intended mission, this paper first briefly reviews the various maneuvering unit concepts and the difficulties related to their evaluation by analysis and simulation. This is followed by a description of the ASMU vehicle and its development program.

THE PROBLEM

The design of any device intended to provide an astronaut with a powered maneuvering capability must be consistent with the dynamics involved. In the free floating environment with the EVA astronaut moving about near his spacecraft or going from one space object to another, the astronaut himself must be considered as a space vehicle which is subject to all the classic laws of motion. In classic theory, to accelerate this body in a given direction, one has to apply a force vector (i.e., a jet thrust) in the desired direction. In order to obtain pure translation this force (thrust) vector must pass through the center of mass thus avoiding generation of torque(s) which, if not countered, would cause undesired rotation of the body. In order to obtain pure rotation, on the other hand, the rotation must be generated by a balanced force-couple consisting of two opposite and equal thrust vectors separated by a moment arm; if the two forces are not equal or not exactly parallel a resultant undesirable translation is also introduced.

Though the astronaut, like any spacecraft, is weightless in orbit, he is not massless, and so has both linear and angular inertias about all six degrees of freedom. Where the usual concept of a spacecraft is an essentially rigid body, this is not the case when the astronaut himself is the spacecraft. From the vehicle engineer's point of view, he is more closely equivalent to a 180 pound mass of jelly than to any known rigid body concept. His loosely articulated joints, his space suit with the built-in constraints, his life support, communications and maneuvering equipment, his camera, his tools and other dangling appendages all make determination of his exact center of mass (CM) and of his inertias quite problematical. The elusiveness of these fundamental parameters makes control of the astronaut-spacecraft a unique problem where the trade-offs developed for essentially rigid vehicles cannot be directly applied.

The various types of powered maneuvering devices approach this control problem in different ways. Two general categories can readily be identified: (a) units whose thrusters are positioned and aimed as needed by the astronaut's limb movements, and (b) equipment with thrusters which are essentially fixed as to location and orientation relative to the astronaut's body. The latter devices contain automatic stabilization equipment, the former do not.

Thrusters Aimed by the Astronaut

A typical example of these devices is the Hand Held Maneuvering Unit (HHMU) utilized by Lt. Col. E. White on his hystoric Gemini IV "Space Walk"; other designs mount the thrusters on the astronaut's fingers or on the shoes of his space suit. These units utilize a minimum number of thrusters (the HHMU has only three) which, by proper positioning and aiming, can produce accelerations in six-degrees-of-freedom. Due to the low number of thrusters (and related plumbing), and as a result of omitting the automatic stabilization equipment, these devices are somewhat simpler, lighter and smaller than the fixed-thruster equipment.

Omission of the automatic stabilization feature, places a considerably increased physical and mental burden on the astronaut. In utilizing these units for translation the astronaut must position and aim the thrusters (by moving his limbs) such that the resultant thrust vector passes through his CM with its direction toward or away from his target for accelerating or decelerating respectively. If he misses his elusive CM, he will rotate until he introduces an opposing torque with his thruster(s). Since these devices cannot produce pure torque couples, his corrective action will also introduce a translational acceleration thus resulting in a change of flight direction. A repetitive series of corrective thruster aiming/firing sequences will thus become necessary. Once the desired flight path is established, the astronaut will "tumble" unless he periodically aimes and fires his thruster(s) to obtain corrective torques.

Fixed Thrusters

An example of the fixed-thruster type equipment is the Air Force's Modular Maneuvering Unit (MMU) which was carried (but not used) on the Gemini-IX-A flight. These devices are usually in the form of a back-pack. In addition to their basic maneuvering equipment they may contain the life support and telecommunications gear (modular or integrated). Since the thrusters are fixed, a minimum of eight are required for attaining control in six degrees of freedom (most units have 12 to 16 thrusters for better fuel economy, control and redundancy). The larger number of thrusters and related plumbing and the addition of the automatic attitude control equipment results in a somewhat increased complexity, weight and size of the propulsion and flight control equipment.

The physical and mental effort required to maneuver with these units is considerably less since (a) in response to translation and/or rotation commands essentially pure accelerations are produced eliminating the need for subsequent reiterative corrections, (b) the strenuous limb movements of positioning and aiming the thrusters are eliminated, and (c) the astronaut's attitude remains essentially stable during the coasting phase of his flight without corrective action on his part (automatic "hands-off" stabilization).

Most fixed-thruster type maneuvering unit concepts also provide a capability of deactivating the automatic stabilization system thus allowing direct open-loop torque-commands to be issued to the thrusters when needed. This mode, referred to as "manual mode" is utilized at the work site or whenever attitude hold is not desired.

Two general types of automatic stabilization approaches have been proposed for these units: (a) conventional Rate Gyro (RG) type systems, and (b) Control Moment Gyro (CMG) type systems.

Rate Gyro (RG) Systems

The Rate Gyro (RG) type stabilization system, schematically depicted in Figure 1, utilizes the propulsion thrusters for producing vehicle control torques. In the absence of astronaut commands the system automatically fires the appropriate thruster combination whenever the angular rate or attitude error exceeds the predetermined deadband values. The system fires the thrusters in a cyclic "bang-bang" manner in response to the rate gyro feedback signals.

When a rotational command is issued by the astronaut the system fires the appropriate thrusters to produce the required vehicle torque. When the rate feedback signal of the rate gyros equals the commanded rate, the thrusters cease to fire.

When the astronaut terminates the rotation command the system automatically fires the required thrusters to produce deceleration torque until the rotation ceases. The astronaut must pre-estimate the time he should terminate his command to assure that he faces in the desired direction when the rotation ceases.

Control Moment Gyro (CMG) Systems

The Control Moment Gyro (CMG) type system utilizes the angular momentum of its wheels to produce the primary control torques; thruster produced torques are utilized only when the momentum capacity of the wheels is exhausted. Since almost all disturbances and commands encountered in frictionless space are cyclic in nature, the momentum stored in the wheels is basically self-replenished with very little, if any, support from the thrusters.

CMG type systems give continuous, proportional stabilization without cyclic "bang-bang" operation. Since transfer of momentum is essentially instantaneous, the astronaut does not have to pre-estimate the time when his rotation command should be terminated in order to achieve the desired attitude; his rotation stops essentially when his command is terminated. The CMG devices are also uniquely suited to coping with the non-rigid nature of the astronaut. Comparison can again be made to the bowl of jelly. When disturbed, it oscillates in a somewhat harmonic fashion about a null. RG equipment would call for jet corrections for both plus and minus errors as they occur, whereas the CMG uses jets only when the accumulative null has shifted beyond its capacity to absorb the disturbances.

Conventional CMG systems utilize rate gyros for feedback sensing purposes as schematically shown in Figure 2; some CMG systems, such as the unique Dual Purpose Gyro (DPG)* system, utilize the momentum wheels for both torque producing

*Patent applied for.

and rate sensing functions as depicted in Figure 3. The DPG approach eliminates not only the rate gyros, but also the feedback electronics. The DPG system is described in a later section in more detail.

Evaluation and Trade-Offs

The maneuvering unit concepts discussed in the foregoing vary as to complexity and weight. In general, the more complex units offer increased ease and quality of operation requiring less physical and mental effort on the astronaut's part.

The concepts also vary regarding the quantity of propellant required for a given mission. The hardware weight penalty of some concepts thus tends to be offset by the reduced propellant consumption.

The relative as well as absolute merits of these devices is totally dependent upon the extent to which these trends hold true in actual zero-g space use. The need to conserve the astronaut's limited energy capacity for useful tasks at the work site was well established by the Gemini EVA experiences. To what extent, and at what cost in weight, the various maneuvering unit concepts attain this goal has been the subject of numerous analyses and simulations. The same is also true of the propellant consumption vs hardware weight trade-off. The analytical and simulation techniques have met with little, if any, success in these areas for the following reasons:

a. Moving base laboratory simulators which utilize air bearings operating on precision floors yield excellent frictionless motion in three degrees of freedom. These devices are well suited for first-order testing and training purposes. While four and/or five degrees of freedom can be produced by the addition of air bearing type gimbals, the non-rigid body of the astronaut cannot be balanced to the level of accuracy required. These simulators cannot produce true six degrees-of-freedom frictionless conditions without distorting the center-of-mass/inertia/mass relationships to such an extent that the simulation becomes invalid. These devices also tend to "rigidize" the non-rigid body by having to support and balance the astronaut in the one-g field. Thus a complete simulation of the complex behaviour and control problem of a non-rigid body in space cannot be attained by the moving base simulator approach.

b. Zero-g KC-135 flights are handicapped by the short duration of the zero-g condition. As reported by Gemini EVA astronauts, the difficulties and physical efforts experienced on actual EVA missions were much greater than those noted in KC-135 flights. In describing the Gemini IX-A EVA mission, Reference 1 states: "While outside the spacecraft, the pilot discovered that the familiarization tasks and evaluations required more time and effort than the ground simulations. .... The tasks of preparing the AMU required much more work than had been expected .... Several corrective measures were initiated for the problems encountered during the Gemini IX-A EVA. .... Also, underwater simulation was initiated in an attempt to simulate the weightless environment more accurately than zero-g aircraft simulations."[1] In the same document, describing the Gemini XI EVA, it is stated that: "The zero-g aircraft simulations had not sufficiently duplicated the extravehicular environment to demonstrate the difficulties of the initial extravehicular tasks."[1] Thus, while KC-135 flights are a valuable tool in the overall development cycle of EVA maneuvering units, they fall short of accurately reproducing the actual EVA conditions.

c. The correlation with true zero-g conditions is much better in the case of neutral buoyancy tests. This observation is, however, limited to simulations where essentially no maneuvering is involved. The hydro-dynamic and viscous drag effects render such simulations invalid when the primary evaluation concerns translational and/or rotational maneuvers.

d. Neither purely analytical studies, nor computer simulations can evaluate physical and mental efforts of astronauts without data regarding typical profiles and energy-costs of the movements, degree and energy-cost of mental effort and anxiety, etc. These methods cannot determine the effect of the non-rigid astronaut on fuel consumption without a typical model of the non-rigid body as it behaves in true zero-g frictionless space. As previously indicated however, such empirical data cannot be obtained in moving base simulations, KC-135 flights and/or neutral buoyancy tests.

Thus moving base simulators, KC-135 flights and neutral buoyancy simulations provide valuable information and training for certain fragments of an EVA mission. None of these methods can, however, by themselves or jointly, give a true and complete reproduction of an EVA mission involving powered maneuvering.

Thus additional evaluation of the various powered maneuvering techniques in an earth orbital laboratory appears to be necessary for selecting the most suitable concept for future operational use. The need for such a reiterative process in the development of an operational maneuvering unit is depicted in Figure 4. This conclusion has prompted the development of the ASMU which allows this evaluation of the basic concepts without unnecessary duplication of hardware.

AN APPROACH TO THE SOLUTION

In order to obtain the most meaningful data for evaluation of the various maneuvering unit concepts, and to do so with minimum weight penalty, the design of the ASMU was based on the following ground rules:

a. The equipment must allow evaluation of all basic concepts (i.e., the fixed-thruster approach with RC, CMG and "manual" stabilization, as well as equipment utilizing thrusters positioned and aimed by the astronaut);

b. The various concepts must be evaluated under essentially identical conditions to achieve normalized data;

c. Hardware common to various units should not be duplicated;

d. The unit must provide a capability of operating in a totally self-contained mode as well as with umbilicals to allow evaluation of the disturbances caused by the umbilicals and to provide extended mission duration with alternate flexibility.

e. The unit must be refurbishable in the orbital laboratory to allow repeated use in various modes.

Vehicle Design

The basic configuration of the ASMU, as illustrated in Figure 5, is that of a backpack with two side-arm controllers. This configuration was chosen to allow representative evaluation of the fixed thruster type maneuvering units. The backpack was carefully contoured to fit both space-suited and shirt-sleeved 10th through 90th percentile astronauts utilizing techniques shown in Figures 6 and 7. The lightweight aluminum structure of the backpack proper, shown in Figure 8, is of a beam-stiffened box construction where the structural shelves and webs are also utilized for mounting components of the subsystems.

The control arms of the unit can be locked in three positions: in the operating position shown in Figure 5, in a 90° up position for stowage and in a down position to provide maximum access at the work site. The rotational and translational control handles are on the right-hand and left-hand arms respectively; functional mode control switches and displays are located on both arms in easy reach within the field of view of a space suited astronaut. Alternate concepts for command/control without the use of control handles are being studied in an attempt to liberate the astronaut's arms and hands for performance of work tasks.

The unit is secured to the astronaut by two adjustable shoulder clamps, an adjustable buttocks-gauge and a waist-harness. Neutral buoyancy tests illustrated in Figure 9 were utilized to develop techniques for donning/doffing of the unit by an unassisted astronaut at zero-g.

The externally located Propellant Supply Subsystem consists of the pressure vessel and all high pressure elements of the propellant distribution system. This integrated approach provides a low pressure interface with the backpack proper via a unique quick-disconnect coupling. By replacement of the Propellant Supply Subsystem with a fully charged one at the low pressure interface the ASMU propellant can thus be refurbished in a safe and rapid manner within the orbital laboratory.

The ASMU provides six degrees of maneuvering freedom and can operate totally self-contained. It can also be used with its propellant and part of its electrical power furnished from the orbital laboratory via umbilical and hardlines respectively. In either case four operational modes are selectable to provide performance representative of the basic maneuvering unit concepts to be evaluated: (a) fixed thrusters with automatic RG stabilization, (b) fixed thrusters with automatic CMG stabilization, (c) fixed thrusters with "manual" stabilization, and (d) thrusters aimed by the astronaut utilizing the HHMU, jet-shoes, jet-fingers or any similar unit. For the last mode the unit provides the HHMU or other similar device with its propellant and with its flight dynamics instrumentation including related telemetry and power.

The unit contains five subsystems: flight control/stabilization, propulsion, electrical power, displays and controls, and data management.

Flight Control/Stabilization Subsystem

The flight control/stabilization subsystem is a culmination of in-depth research and development work in the specialized field of controlling and stabilizing maneuvering units. Numerous analytic studies and hardware development/testing efforts were conducted in regards to various RG and CMG type systems in order to evaluate their applicability to the problem of stabilizing a highly non-rigid body in space.

The selected mechanization for the ASMU is illustrated in Figure 10. It utilizes the unique Dual Purpose Gyro (DPG) type system for the CMG mode of stabilization. In addition it also provides essentially conventional types of RG and "manual" stabilization modes. The thruster-torque commands produced in any one of these modes together with the on/off translational thrust commands, are processed in the jet select logic for firing of proper thruster combinations.

CMG Stabilization Mode Using the DPG System

Several generations of DPG hardware have been built, tested, and demonstrated over the past four years. The concept was tested on three degree of freedom air bearing tables, as shown in Figure 11, on three degree of freedom air bearing type moving base simulators as shown in Figure 12, as well as in KC-135 zero-g flights.

The DPG system was selected over other CMG devices for the ASMU application in light of the following advantages inherent in its unique mechanization:

a. It utilizes its momentum wheels for two simultaneous functions (hence its name); the same wheels provide the vehicle control torques by momentum exchange and also sense vehicle rates for self-contained feedback purposes. Thus the need for rate gyros is eliminated in the CMG mode.

b. Since the large momentum wheels provide the rate sensing, their inherent low sensing threshold and drift assures superior attitude hold and stabilization characteristics without sophisticated sensors and/or electronics.

c. The system requires no electrical feedback loop, summing, shaping, filtering or other computational functions. The feedback is attained mechanically by the unique utilization of the gyroscopic characteristics of the momentum wheels.

d. Improved reliability is provided by the inherently redundant nature of the system (2 wheels per axis; any one of the six wheels can fail without significant degradation in performance).

e. Simple, reliable mechanical linkages can be utilized with the system for input commands, eliminating the power demands of electrical torquing.

DPG Mechanization

The DPG utilizes two gyro wheels per axis which are mechanically coupled with sector gears. The gimbals have limited angular travel with nominal values of $\pm 45°$. When the gimbal angle is zero, the net angular momentum is zero, which eliminates gyroscopic cross-coupling as compared to a single gyro per axis concept. The wheels operate very efficiently within evacuated and hermetically sealed inner gimbal housings. The sealed gimbals are mounted in turn within a hermetically sealed outer housing structure which is filled with silicone oil for damping. Command torques are applied to the gimbals through a bellows mechanism which transmits the torque while maintaining the hermetic seal integrity. A mechanical linkage system connects the ASMU control handle to the respective DPG for three axis control. This arrangement requires no power for commanded turns and results in vehicle rates which are proportional to handle position. Further features of the DPG design include electric caging solenoids and desaturation switches to turn on thrusters via the jet select logic when the momentum wheels reach saturation.

The two gimbaled rotors are mounted so their spin vectors are coincident but oppositely oriented in space. The gimbal axis of each rotor is parallel to the other and mechanically constrained to rotate oppositely. The third axis normal to both the spin axis and the gimbal axis is designated as the sensing and control axis. This arrangement is illustrated in Figures 13 and 14, which show how a DPG stabilizes a vehicle disturbance and how a command torque will cause rotation of the vehicle respectively.

Stabilization by DPG

The objective for stabilization is that momentum should be exchanged from the DPG to offset the momentum of the vehicle so that the total momentum of the system will equal zero; i.e.,

$$H_{Vehicle} - H_{DPG} = 0$$

The vehicle rate about the sensing axis causes the gyros to be slewed in inertial space. They precess about their gimbal axes so that the spin vector will tend to align with the applied torque vector of the vehicle rate. The combined components of the two motor spin vectors will be subtractive to the vehicle angular rate vector and bring the vehicle rate to zero. This action is accomplished without cross-coupling to other vehicle axes as the two spin vector components along the original spin axis are still equal and opposite, their sum thus remaining zero. If the initial vehicle rate were so large that it causes the DPG wheels to precess to their limits of $\pm 45°$, then the jets are switched on to further reduce vehicle rate. As the vehicle rate reaches zero, the jet torque precesses the gyros off their switch and re-establishes momentum exchange control.

Command Turn by DPG

When a torque is applied to the gimbals of a DPG, the wheels precess and add their spin vector components to the vehicle control axis. The conservation of momentum law requires the total system momentum to remain unchanged so the vehicle rotates oppositely to the applied spin vector momentum.

The amount of DPG momentum transferred is:

$$H_{DPG} = 2\ H_G \sin \Theta$$

where: H_G is rotor momentum
Θ is the gimbal angle of tilt (precession).

The formula for vehicle rate is:

$$I_V \omega_V - 2\ H_G (\sin \Theta) = 0$$

where: ω_V is vehicle rate
I_V is vehicle inertia

and thus

$$\omega_V = \frac{2H_G (\sin \Theta)}{I_{Vehicle}} .$$

Upon release of the command torque, the system automatically re-exchanges momentum to restore the vehicle rate to zero. The dynamics of the system are governed by pre-selection of the damping fluid viscosity as required for stability.

RG and "Manual" Stabilization Modes

In the RG and "manual" stabilization modes the DPG units are caged in their zero degree gimbal positions; thus their net angular momentum is zero and they do not affect the performance in the other modes. The rotational commands are issued via the same control handle as that used in commanding the DPG's. Although the DPG's are caged, the command handle has sufficient freedom of motion to actuate electrical pickoffs which generate rotational command signals which are proportional to handle position.

In the RG mode these rotational command signals are processed by the RG electronics network; the network sums the command signals with the negative vehicle rate feedback signals of the rate gyros, imposes a rate deadband to the resultant error signal and, when the deadband value is exceeded, issues thruster-torque commands to the jet logic. The thrusters will fire until the vehicle rate, as indicated by the rate gyros, essentially reaches the commanded magnitude. The vehicle rate persists until the command handle is returned to null; since the command signal is now zero, while the rate gyros produce a signal in excess of the rate deadband, the electronics network initiates opposite thruster-torque commands. The thrusters will fire until the vehicle rate is reduced to essentially zero. Whenever the command signal is zero the electronics network adds an attitude feedback loop to the rate feedback; this is obtained by electronically integrating the rate gyro outputs and applying a suitable deadband to the resultant signal. This additional feedback reduces the drift of the vehicle thus providing improved "attitude hold".

In the "manual" mode the command signals produced by the handle pickoffs produce direct thruster-torque commands to the jet select logic. The thrusters will fire as long as the astronaut continues to issue the command. In order to stop his rotation the astronaut must command opposing thruster-torques. Thus the task of maintaining his attitude and nulling his rate is transferred to the astronaut in the "manual" mode.

Propulsion Subsystem

The propulsion subsystem of the ASMU utilizes gaseous oxygen propellant to allow safe operation within the orbital laboratory without contamination of the controlled atmosphere. The subsystem, illustrated in Figure 15, consists of two basic parts: (a) the integrated propellant supply subsystem which contains all high pressure elements and is removable via a low pressure quick disconnect interface, and (b) the low pressure distribution system which feeds 16 solenoid operated thrusters and also provides propellant to the HHMU via a quick disconnect.

Integrated Propellant Supply System

The Propellant Supply System (PSS) consists of a spherical inconel pressure vessel for storing gaseous oxygen at 6000 psi. Housed within the neck of the vessel is a gas supply assembly which contains seven elements (including all high pressure elements): pressure regulator, fill valve, bleed valve, burst disc, check valve pressure gauge and a pressure transducer actuator. The latter device engages a load cell in the ASMU backpack (whenever the PSS is attached to the backpack) permitting monitoring of remaining propellant pressure without an electrical backpack-to-PSS interface.

The pressure regulator reduces the oxygen pressure to 165 psig upstream of the disconnect fitting permitting safe exchange of the PSS by an unassisted astronaut in the orbital laboratory. The unique quick disconnect mechanization provides interlocks to prevent accidental uncoupling of the PSS.

Low Pressure System

The translational acceleration forces and the rotational acceleration torques required for 6 degree-of-freedom (DOF) control of the ASMU are produced by 16 solenoid operated thrusters. These thrusters are arranged in two sets, each set being supplied by a separate distribution system. While for normal operating mode all 16 thrusters are utilized, complete 6 DOF control can be obtained at somewhat degraded performance in the backup mode with one set of thrusters deactivated. Each of the two distribution systems is controlled by an isolation solenoid: actuation of either solenoids controls both the oxygen flow to the respective distribution system and the electrical power to the thruster solenoids supplied by it.

Of the 16 thrusters 8 produce a nominal thrust of 2 pounds each, while 8 produce 1 pound each. Four of each size are supplied by each of the two distribution systems. The large thrusters are used to produce fore/aft, yaw and pitch accelerations, while the smaller ones furnish up/down, transverse and roll accelerations. Two of the large thrusters are located in the end of each sidearm , while the remaining 12 thrusters are located in the backpack proper.

Mounted in the low pressure manifold are two check valves; these control the flow of supply oxygen from the PSS and from the ASMU/orbital laboratory umbilical for self-contained and umbilical modes of operation respectively. The manifold also contains a pressure/temperature transducer to provide flight data regarding propellant consumption. A relief valve located in the manifold protects the low pressure system. A separate oxygen line is provided from one of the two distribution systems to furnish propellant to the HHMU via a quick-disconnect fitting.

Electrical Power Subsystem

The electrical power subsystem of the ASMU, illustrated in Figure 16, supplies the power demands of the ASMU subsystem elements via two busses. One bus supplies all the intermittent duty components from the battery of the ASMU. The other bus supplies the continuous duty elements (i.e., DPG motors, rate gyros, flight data subsystem). The latter bus draws its power from the battery of the ASMU when the unit is in a self-contained flight mode; at other times the bus is supplied with power from the orbital laboratory via hardlines. Transfer of the bus from/to ASMU battery power is automatic as a function of power being applied via the hardlines. This arrangement conserves the ASMU battery power while preventing the transient surges of the intermittent duty components from reflecting upon the power supply of the laboratory.

The inverter utilized for driving the DPG momentum wheels is of a unique design resulting in approximately 80% efficiency during wheel runup as well as during steady state wheel operation at synchronous speed.

The battery is rechargeable while in the ASMU and can also be readily removed for exchange. Replenishment by either method can be accomplished within the orbital laboratory.

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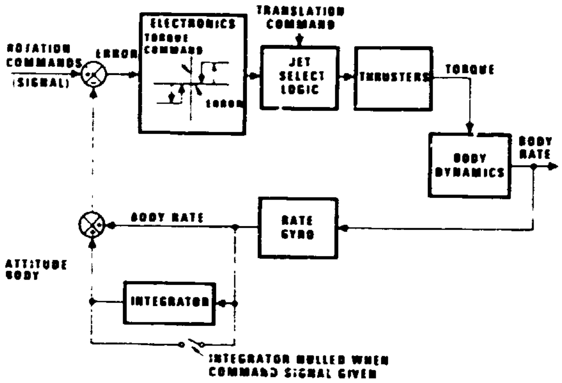

Figure 1. Typical Rate Gyro Type Attitude Control and Stabilization System

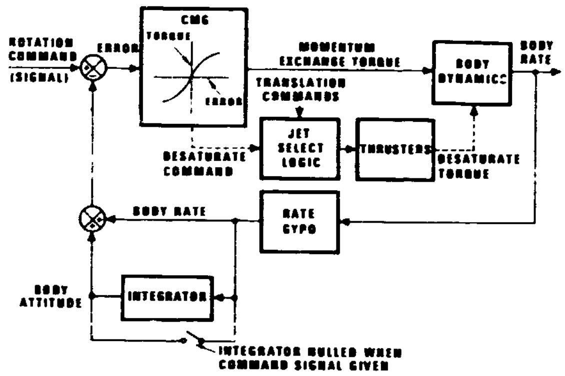

Figure 2. Typical Control Moment Gyro Type Attitude Control and Stabilization System Utilizing Rate Gyro Feedback

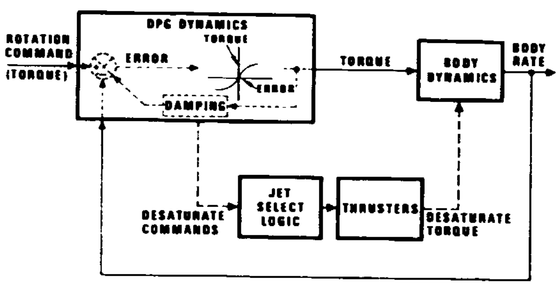

Figure 3. Dual Purpose Gyro (DPG) Type Attitude Control and Stabilization Control

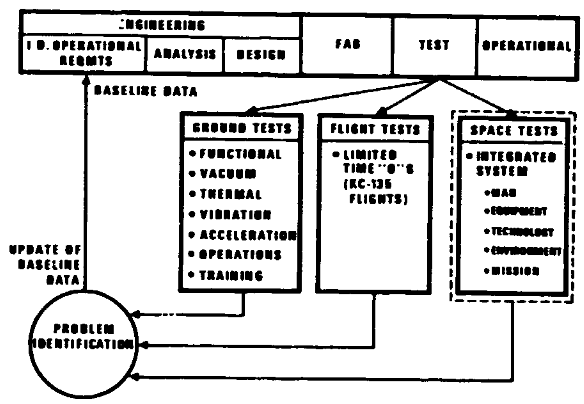

Figure 4. Development Process for an Operational Powered Maneuvering Unit

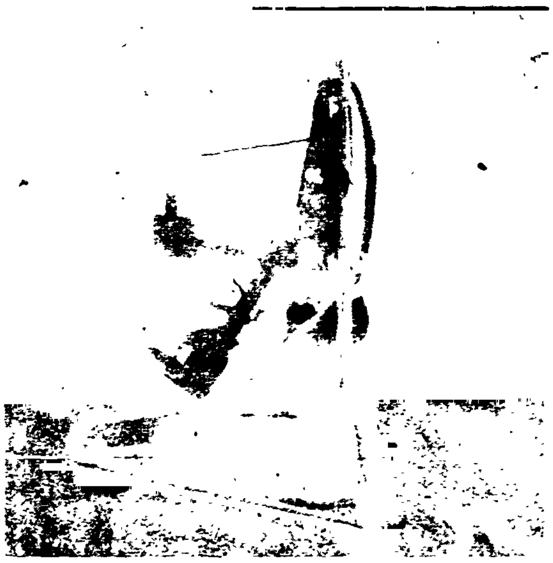

Figure 5. ASMU Configuration

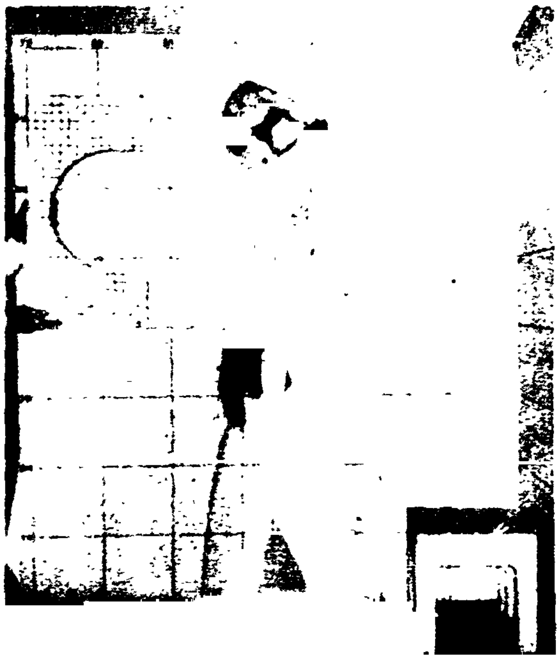

Figure 6. Measurement of Astronaut Contours

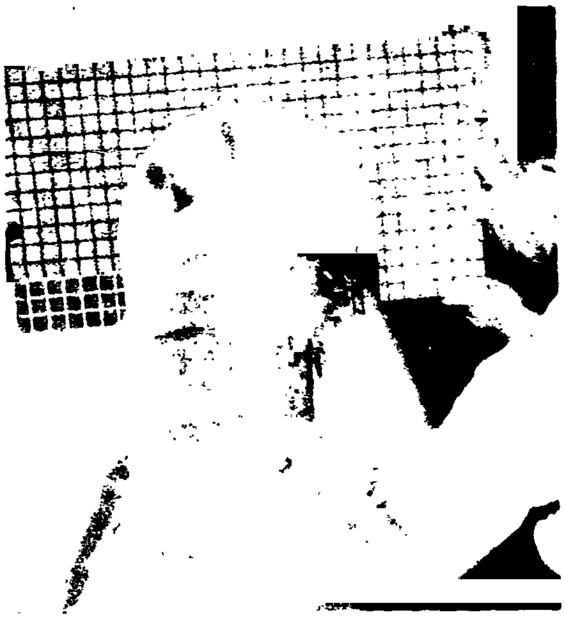

Figure 7. Form-Fit Test of ASMU

Figure 8. ASMU Structure

Figure 9. Donning/Doffing Techniques Evaluated in Under-Water Neutral Buoyancy Tests

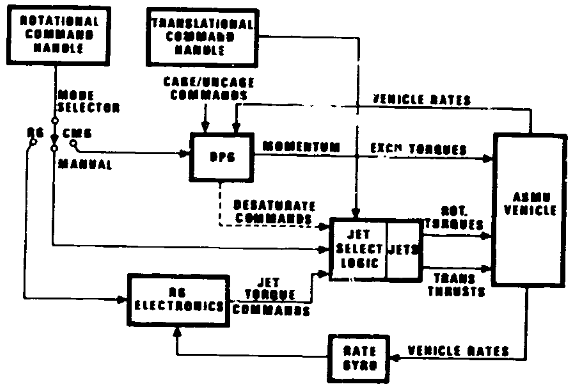

Figure 10. Functional Diagram of ASMU Flight/Control/Stabilization Subsystem

Figure 11. DPG System Tested on 3 DOF Air Bearing Table

Figure 12. DPG System Tested on 3 DOF Air Bearing Moving Base Simulator

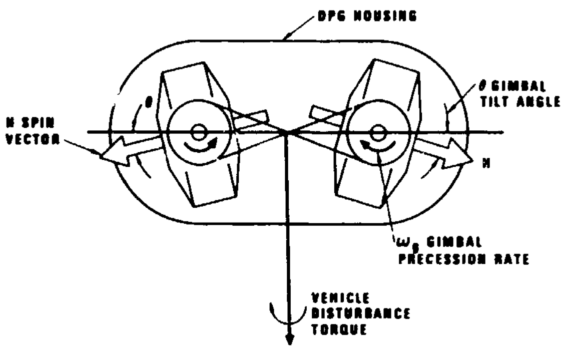

Figure 13. Stabilization With the DPG System

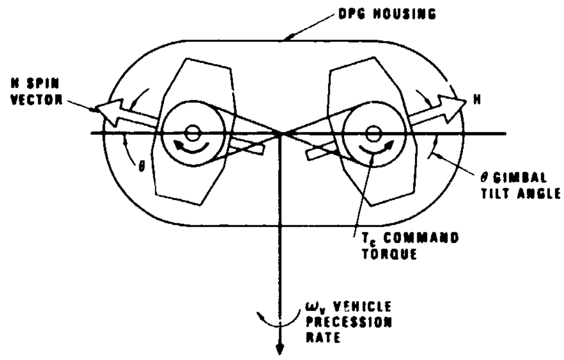

Figure 14. Command Turn With the DPG System

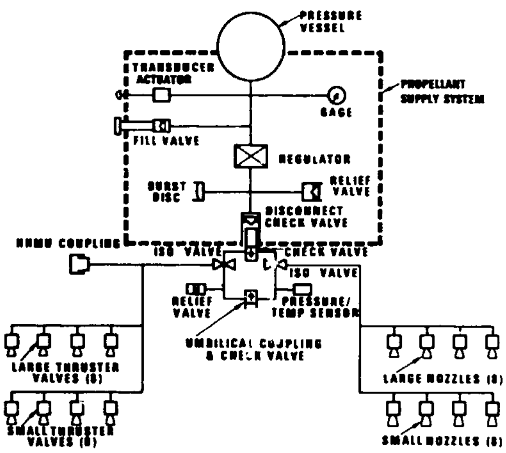

Figure 15. ASMU Propulsion Subsystem

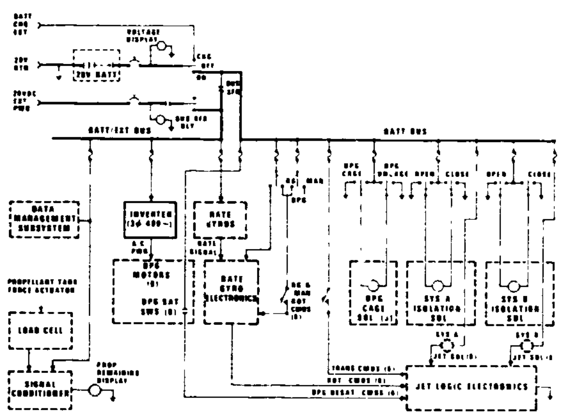

Figure 16. ASMU Electrical Power Subsystem

VOICE CONTROLLER FOR ASTRONAUT MANEUVERING UNIT

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SUMMARY: The requirement for an alternate to the hand control of astronaut maneuvering units has been recognized for some time. The objective of this program has been to design and implement an experimental limited-vocabulary speech recognition device for use as a voice controller.

INTRODUCTION

For several years, the Air Force has been developing and testing concepts for extravehicular space maintenance and maneuvering. Although efforts to date have achieved initial objectives, the typical astronaut maneuvering unit (AMU) requires the use of both hands for operation.

Air Force interest in utilizing a voice controller to command the AMU was generated primarily from the pay-off of hands-free operation in cargo transfer operations. Voice control would enable the astronaut to use his hands to carry the cargo. The savings in the time associated with, and the weight of, a cargo attachment device would more than offset the additional weight of the voice controller system which is projected to weigh less than 5 pounds. Another advantage of the voice controller is associated with rescue.

Once a voice controller is provided on the AMU, a space station astronaut can assume command of the AMU, if required, through the voice link. There are other advantages associated with using a voice controller on the AMU or with an RMU (Remote Maneuvering Unit), however; the potential cargo transfer and rescue pay-offs are sufficient to support exploratory development of a voice controller. It should be noted that it is not intended, at this point in time, to replace or eliminate hand controllers from the AMU. Hand control could remain the primary mode of control with the voice controller used only during cargo transfer and rescue. It is also possible that the voice controller could become the primary mode of control with a simplified hand controller as a back-up mode. The operational approach is yet to be determined by simulation studies using the experimental voice controller.

The use of a voice controller with a maneuvering unit was first simulated on Bell Aerosystem Company's Visual Simulation Facility as part of the Dual Maneuvering Unit (DMU) program (AF33(615)-3529). Simulation tests of the DMU in both manned and remote modes of operations were conducted to evaluate performance characteristics using a voice controller to initiate and terminate DMU translation and rotation commands. Performance in terms of fuel expenditure did not differ in these limited initial tests from results obtained while using the standard hand controllers. Simulation of the voice controller was most easily implemented by employing two operators. Operator No. 1 represented the astronaut and Operator No. 2 represented the voice controller.

These tests, together with LTV Aerospace Corporation's demonstration of the VOCON (VOice CONtroller), established the feasibility of using a voice controller with a maneuvering unit. The effort described in this paper with RCA, under the sponsorship (Contract F33615-67-C-1960) of the Air Force Avionics Laboratory and the Air Force Aero Propulsion Laboratory, was initiated to provide an experimental model of a voice controller.

The development of a successful voice controller for an astronaut maneuvering unit requires a real-time speech-recognition system that can recognize a limited vocabulary in continuous speech for a few speakers with high accuracy and that can ultimately be constructed with a minimum of volume and weight. The weight and volume restrictions require a speech-recognition system whose hardware is not so complex that the voice controller application is impractical. On the other hand, a crude recognition technique that has poor recognition and false response characteristics will be unsatisfactory no matter how small the unit can be constructed.

Although the experimental model controller is about the size of a small suitcase, it incorporates design principles which allow the space unit to be less than 100 in^3 in volume, weight less than 5 pounds, and consume less than 10 watts. The experimental unit will recognize the activate word, "command", from continuous speech and can be used without adjustment by three predetermined speakers. This voice controller will be tested with both manned and unmanned maneuvering units by the Air Force Aero Propulsion Laboratory at WPAFB, Ohio. Assuming these tests are successful, several problems associated with operational deployment of a voice controller remain:

1. The present voice controller model will accommodate an Apollo-size crew. What about a last-minute change of astronaut personnel? Ideally, this problem could be solved by substituting a new recognition card into the voice controller. A set of these inexpensive cards could be prepared in advance for each probable astronaut.

2. What about the background noise environment to which the voice controller will be subjected? If the characteristics of the noise and the communications system are accurately known in advance and are reasonably invariant, this problem could be handled.

3. What effect will the different spacecraft and space suit atmospheres have on voice recognition? These problems were not studied in detail during the development of the experimental voice controller, but they were considered. Although the specific recognition logic must be changed to accommo-

date different atmospheres, the basic system is satisfactory. Detailed attention can be deferred to the development of the space qualified voice controller.

The following sections describe the approach to speech recognition that has been developed by RCA and used in the experimental voice controller.

MACHINE RECOGNITION OF SPEECH

In recent years, the automatic recognition of speech by machine has been the goal of many investigators.[1] Past attempts to implement such a system have had limited success primarily due to the complexity of the problem. Although speech is a highly complex signal, it may be regarded as a sequence of articulatory events. In order for a human to completely recognize and comprehend speech, the incoming signals must be processed at many levels - e.g., at acoustic, linguistic, and semantic levels. The majority of studies of the automatic recognition of speech by machine have concentrated on recognition at the acoustic level since this level has presented the greatest obstacles.

Acoustically, speech can be considered as a succession of spectral steady states and transitions. These relations arise from the properties of the human vocal tract. In speaking, different positions of the tongue, lips, and jaw give the vocal tract different shapes. Each shape then gives rise to a distinct frequency spectrum, and each change of shape gives rise to a spectral transition. Vocal cord vibrations give rise to voiced sounds, and noise-like sounds are produced by the movement of air across the edges of the teeth and by partial closure of the vocal cords. The important point to be noted here is that the resonant frequencies generated are constantly changing. The phonemes* which are generated by the steady-state and transitional sounds often exhibit differing acoustic characteristics, depending upon the position of the phonemes within the words. Approximately 40 phonemes constitute the basic sounds of spoken English and are shown in Table 1. The variances of phonemes resulting for contextual considerations are called allophones. The rapid stringing together of phonemes gives the resultant acoustic signal.

The literature contains very little work on the recognition of continuous speech -- that is, with no pause between words or sounds. Obviously, a means of recognizing, in real time, a limited vocabulary in continuous speech must be employed in a practical AMU voice controller. The results obtained in the recognition of isolated speech cannot be extrapolated to predict recognition scores for continuous speech, because none of these techniques is suitable for real-time recognition of continuous speech. The problems of recognizing continuous speech require special considerations, in addition to those necessary for the recognition of isolated speech.

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*Phoneme - the smallest unit of speech that, in any language, distinguishes one utterance from another, such as /p/ in pin and /f/ in fin. (Diagonals / / enclosing a letter or letters indicate the phoneme represented and not the letter itself.)

TABLE 1. ELEMENTARY SOUNDS (PHONEMES) WHICH OCCUR IN ENGLISH

Phonetic Symbol	Key Word
1. Simple Vowels	
I	fit
i	feet
ɛ	let
æ	bat
ʌ	but
ɑ	not
ɔ	law
U	book
u	boot
ɜ	bird
2. Complex Vowels	
e	pain
o	go
aU	house
aI	ice
ɔI	boy
iU	few
3. Semivowels and Liquids	
j	you
w	we
l	late
r	rate
4. Plosives	
b	bad
d	dive
g	give
p	pot
t	toy
k	cat
5. Nasal Consonants	
m	may
n	now
ŋ	sing
6. Fricatives	
z	zero
ʒ	vision
v	very
ð	that
h	hat
f	fat
θ	thing
ʃ	shed
s	sat
7. Affricatives	
tʃ	church
dʒ	judge

It is an oversimplification to consider continuous speech as being composed merely of a sequence of separately produced sounds. It is important to be aware that, in continuous speech, sounds can easily modify surrounding sounds so that the waveform of a sound produced in continuous speech can differ markedly from the waveform of the same sound produced in isolation. Even for isolated words, many sounds are modified by preceding and following sounds.

There are many additional problems in recognizing continuous speech. Sounds can be eliminated, added, combined, or substituted for one another. The stress and intonation of the speaker cause variations in spoken words. Consequently, the very uncontrolled nature of continuous speech produces many difficulties when one attempts to automatically recognize sounds strung together with no pauses between the sounds or words.

RCA Advanced Technology's research in speech analysis proceeded from studies[2,3,4,5,6] of speech processing in the human auditory system to the development and design of a speech-recognition system utilizing a unique form of logic shown to be highly efficient for pattern recognition. The processing technique employed in the speech-recognition studies has been named analog-threshold logic (ATL) because the element has an output proportional to the algebraic sum of its inputs once this sum exceeds a threshold. Networks of ATL elements can abstract both the presence and magnitude of significant steady-state and transient features from the speech signals. To facilitate machine recognition of sounds in continuous speech, a hierarchical organization of feature-abstraction networks is utilized which provides the capability of acoustic recognition results that are accurate as the quality of speech to be analyzed. Three basic types of features are used: broad class features, common basic features, and unique phoneme features. Briefly, broad class features are relatively insensitive to localized noise and may be the only information that can be provided under poor communication conditions. The common basic features are those which are common to very similar phonemes, for example, /f,s/, but which do not differentiate between these phonemes.

A detailed listing of the types of speech sounds found in each of the class feature, common basic feature, and phoneme classes is shown in Fig. 1. It should be noted that some overlaps of various sounds occur within a class. These overlaps primarily are a result of the variability of those sounds found in continuous speech from many speakers.

MACHINE RECOGNITION OF THE AMU-VOICE CONTROLLER VOCABULARY

For most limited-vocabulary applications and, in particular, for the AMU vocabulary (see Table 2), only the broad class features and a few common basic features are necessary for recognition. This simplification results from the fact that, in the AMU vocabulary, no pair of words differs only by a single phoneme, so that no critical phoneme decisions are required. Since both the class features and common basic features can be abstracted with greater accuracy than can be obtained from a phoneme-by-phoneme decision process, it is neither necessary nor advantageous to utilize the phoneme decision for limited-vocabulary application. In fact, because of the substitution

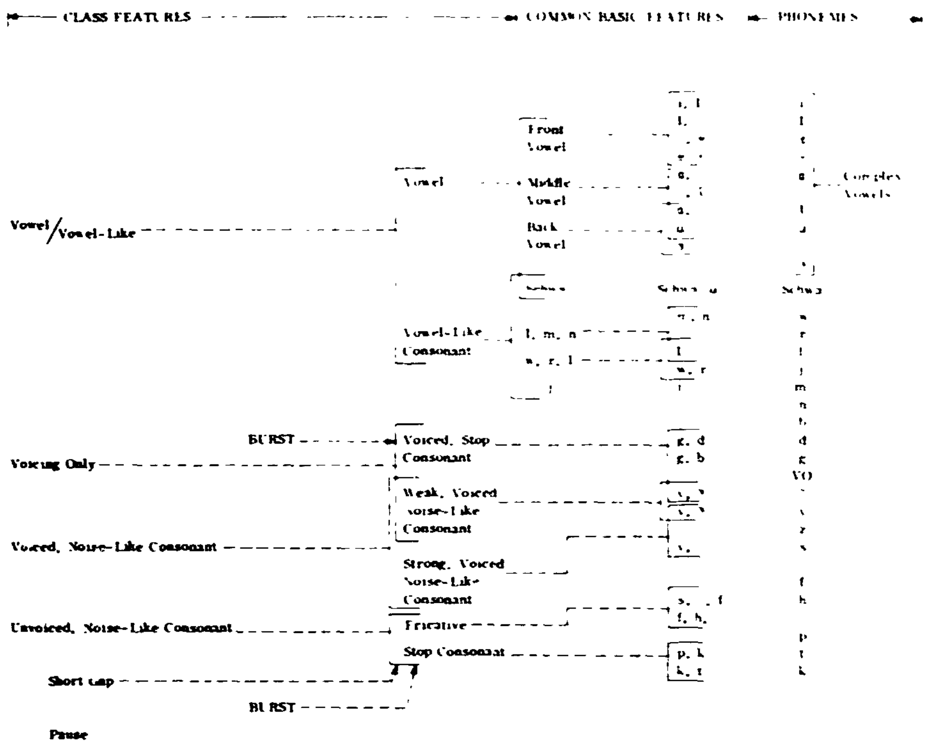


Fig. 1. Hierarchical organization of feature-abstraction networks.

TABLE 2. VOICE CONTROLLER VOCABULARY

COMMAND WORDS	
1. Command	8. Hold
2. Stop	9. Open
3. Forward	10. Pitch
4. Back	11. Yaw
5. Right	12. Up
6. Left	13. Down
7. Roll	14. Translate

phenomenon (the speaker substitutes one phoneme in a word in place of the correct one such as /z/ for /s/), the accuracy and reliability of the recognition can be seriously deteriorated. Thus, the minimum possible processing should be used to provide the discrimination necessary to separate the various words in a limited vocabulary. On the other hand, under operational conditions, the input to the voice controller is the same as the input to the normal communications channel, so that AMU commands and normal conversation will be interspersed on the same channel. This commonality requires that good discrimination be provided against words in ordinary communications which are similar to the AMU vocabulary in order that unacceptable false commanding of the AMU be prevented. This problem is somewhat simplified by requiring that a key word, such as "command", precede each legitimate command to the AMU. Now only this key word must be distinguished from all other words that may occur. Thus, all words are ignored except the key word, the AMU vocabulary being recognized only when preceded by the key word.

In practice, the experimental voice controller will have two modes of operation: a "command" activate mode and a "fast" mode. In the "command" activate mode, all control functions will be prefixed by the spoken word "command" followed by the desired function (e.g., "command, pitch up; command stop"). In the "fast" mode, the prefix word "command" is not used. In each of the modes of operation, some complex commands, such as "roll right", must be recognized by the voice controller.

The feature-abstraction networks employed in the AMU voice controller abstract from the time varying spectrum of the input speech signal features in the form of 1) combinational and sequential logic arrangements of what are referred to as primary features, 2) analog ratios of the primary features, and 3) higher-order logic based on the results of the first two types of features. The AMU vocabulary is recognized with only three primary features: broad positive slopes (PSB), broad negative slopes (NSB), and log of local energy (E_N). These features are further described by Fig. 2 which shows the spectral energy distribution of an idealized vowel. Broad positive slopes are those regions in the spectrum where the energy is rising with frequency (+dE/df). The regions where energy is decreasing with frequency (-dE/df) are called broad negative slopes. The logarithm of the analog value of the energy in a narrow portion of the spectrum (channel) is referred to as log of local spectral energy.

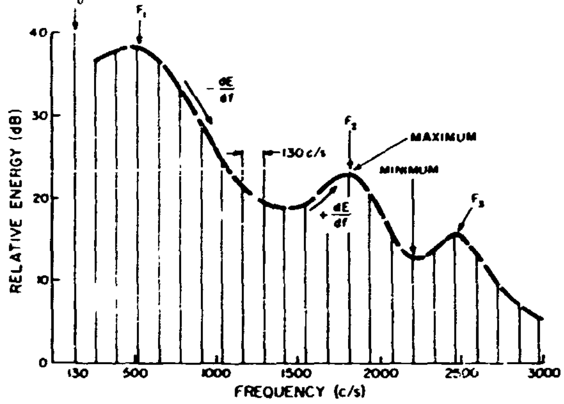
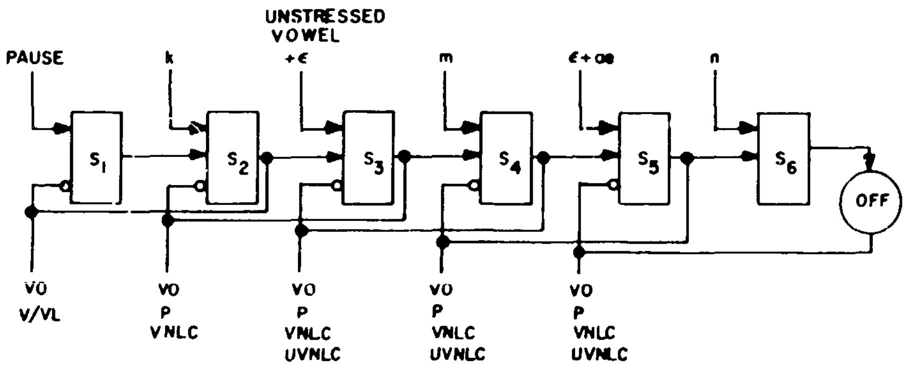


Fig. 2. Idealized characteristics of the vowel sound in the word "bed".

Following the primary feature-abstraction process described above, it is necessary to combine these abstracted features in sequence-recognition logic, a differen feature sequence for each voice controller vocabulary word. Fig. 3 shows the class feature sequences which characterize each word in the voice controller vocabulary. As a specific example, the sequence logic for the word "command" is shown in Fig. 4. An output from logic element S6 indicates recognition of the word "command". Recognition of the remainder of the vocabulary is obtained using similar sequence logic.

AMU Vocabulary Word	Class Feature Sequence
COMMAND	Pause → unvoiced stop → front/middle vowel → vowel-like consonant → front/middle vowel → vowel-like consonant
STOP	Unvoiced fricative → unvoiced stop → middle/back vowel → voiced stop
FORWARD	Unvoiced fricative → complex middle/back vowel → vowel-like consonant → back vowel → voiced stop
BACK	Voiced stop → front vowel → unvoiced stop
RIGHT	Vowel-like consonant → complex vowel → unvoiced stop
LEFT	Vowel-like consonant → front vowel → unvoiced fricative → unvoiced stop
ROLL	Vowel-like consonant → complex vowel
HOLD	Unvoiced fricative → complex vowel → voiced stop
OPEN	Complex vowel → unvoiced stop → unstressed vowel → nasal
PITCH	Unvoiced stop → front vowel → affricative
YAW	Complex front vowel → back vowel
UP	Middle vowel → unvoiced stop
DOWN	Voiced stop → complex vowel → nasal
TRANSLATE	Unvoiced stop → vowel-like consonant → front/middle vowel → vowel-like consonant → unvoiced fricative → vowel-like consonant → complex vowel → short pause

Fig. 3. Class feature sequences characterizing AMU vocabulary words.



VO	≡	VOICING ONLY	CLASS FEATURE RESET INPUTS
P	≡	PAUSE	
VNLC	≡	VOICED, NOISE- LIKE CONSONANT	
UVNLC	≡	UNVOICED, NOISE LIKE CONSONANT	
V/VL	≡	VOWEL/VOWEL-LIKE	

(A) COMMAND SEQUENCE RECOGNITION LOGIC

SYMBOL FOR SEQUENCE LOGIC RESPONDING TO X BEFORE Y

NOTATION INDICATING X BEFORE Y WITHOUT THE OCCURENCE OF I IN THE INTERVAL AFTER X AND BEFORE Y

(B) SYMBOL AND LOGICAL EQUIVALENT OR SEQUENCE RECOGNITION ELEMENT

Fig. 4. Feature sequence logic for "command." Following the feature-abstraction logic, it is necessary to include sequence-recognition logic for each word in the AMU vocabulary. The elements labeled S_n in the sequence recognition logic shown above are cascaded to recognize the sequence of phonetic events comprising the word "command." The reset inputs are the class features which should not occur between the two events in the sequence.

THE VOICE CONTROLLER SYSTEM

The previous sections have described what functions are required to recognize the voice controller vocabulary. This section describes the manner in which these functions are obtained.

The block diagram of the basic voice controller system is shown in Fig. 5. In very general terms, the system functions in the following manner. Speech is transduced from acoustical energy to electrical energy by the microphone. The frequency spectrum of the speech input is shaped in a gross fashion in the equalizer. The preprocessor section which follows the equalizer performs a detailed analysis of the frequency-time content of the equalized speech signal. In particular, time-varying spectral shapes are abstracted and serve as inputs to the processor. In the processor, the time-varying frequency features are combined to achieve recognition of the speech elements required for the AMU vocabulary. Also in the processor, these abstracted speech elements are combined in proper sequence to give final word recognition. The final section, the display and interface, gives a positive indication of word recognition. The display function of course is not required in operational use, but only serves as a convenient method of monitoring system performance.

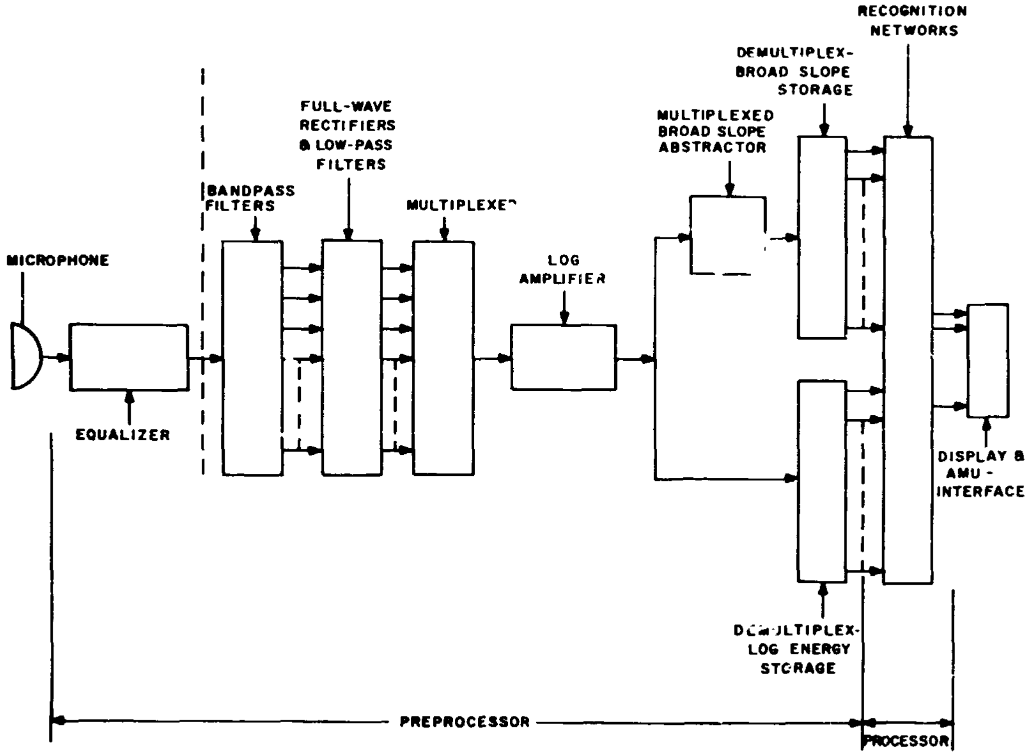


Fig. 5. Block diagram of AMU voice controller system.

In more specific terms, the functioning of the system is as follows: The electrical output of the microphone serves as the input to the equalizer. The equalizer is required to flatten the irregular frequency characteristics of the microphone and to provide preemphasis. The equalizer also provides an impedance match between the microphone and the filter bank as well as supplying the necessary voltage gain. From the equalizer, the signal is fed into a bank of filters. This bank of filters includes 14 low-Q bandpass filters which are used to separate the speech signal into its time-varying spectral components. In order to facilitate measurements of the time-varying spectra, the outputs of the bandpass filters are individually rectified and lowpass-filtered. (An additional full-wave rectifier and lowpass filter are used to process the speech waveform itself, resulting in a total of 14 filtered channels and one unfiltered channel at this point in the system.) All 15 channels of speech data are multiplexed into one common logarithmic amplifier. A logarithmic amplifier is used both to compress the dynamic range of the input signal to a range more manageable for the recognition networks and to provide very desirable amplitude normalization of the input signal. The amplitude normalization is accomplished by virtue of the fact that the basic recognition operation is a differencing operation. Taking the difference between two logarithmatized quantities is equivalent to obtaining the ratio of these quantities before the logarithmic operation. A ratio of two quantities is naturally invariant to simple amplitude changes such as those caused by a change in gain.

The speech signal is multiplexed into a single logarithmic amplifier in order to circumvent the impracticality of matching 15 logarithmic amplifiers over a wide dynamic range. The logarithmic amplifier feeds two parallel branches. The first of these parallel branches contains the multiplexed broad slope abstractor. On a shared-time basis, this multiplexed broad slope abstractor derives gross (broad) measurements of the first derivative (slope) of energy with respect to frequency and quantizes this measurement into three possible outputs with binary indications that a broad slope is or is not present. This broad slope information is subsequently demultiplexed and stored using only digital logic. In the second parallel channel emanating from the logarithmic amplifier which contains the 15 log-energy values in serial form, a simple but highly accurate demultiplex and storage operation is performed. The final stored outputs from the two parallel channels serve as inputs to the recognition networks. It is in the recognition networks that the elements of speech are finally recognized. This recognition is accomplished generally as follows: Products and ratios of channel energies (and their time variations) for a specific speech element are obtained by combining the outputs of the log energy storage networks in high-gain difference circuits with thresholds. The gain of the circuit is sufficiently high that the output is quantized, i.e., the output of the circuit is a high level when the input difference exceeds the threshold but the output is a low level when the input difference is below the threshold. These digital outputs are combined with the digital broad slope information in various digital logic functions to result in final recognition of a speech element. Several of these speech element recognitions are then combined in sequential digital logic resulting in the recognition of the desired word. Finally, indications that this recognition has been accomplished is given on the visual display

The experimental voice controller is built in a generally modular form with the module divisions shown in Fig. 5, i.e., microphone, preprocessor, processor and display. The circuitry contained in both the preprocessor and processor is built on 4-inch by 6-inch circuit boards, which plug into two prefabricated standard 19-inch circuit card nests. The circuits have been designed with the ultimate miniaturization and space qualification requirement in mind so that, wherever practical, designs which would preclude this miniaturization and space qualification have been discarded.

ACKNOWLEDGMENT

A substantial part of the design and experimental work on the AMU voice controller was performed by J. T. Martin and J. R. Barger, of RCA, Advanced Technology.

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UNMANNED RENDEZVOUS, STATION-KEEPING AND DOCKING FOR EXTRAVEHICULAR SPACE ACTIVITIES

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Summary: Rendezvous, station-keeping, and docking requirements are identified for a potentially important class of space mission, namely, refurbishment of passive (i.e., noncooperating) satellites in synchronous orbit. An unmanned system for performing this mission (called ARMS for Application of Remote Manipulators in Space) is described. Concepts, techniques, and computer simulation results for closed-loop unmanned rendezvous and station-keeping are presented. Soft docking and anchoring with noncooperating satellites, using a remotely controlled manipulator system, are discussed in terms of requirements for the baseline mission.

INTRODUCTION

The feasibility of performing simple work tasks in space has been demonstrated by both man and remotely controlled machines. The activities of the extravehicular astronauts, the success of the Surveyor Moon-Digger program, and the recently announced Russian achievements in effecting ground controlled rendezvous and docking provide credibility for postulating highly sophisticated space tasks for both man and machine.

Analyses of certain space missions have established the desirability of using man-equivalent systems for performing extravehicular work tasks on satellites previously placed in orbit. Where safety hazards exist or where work of a routine nature would make the cost of dedicated manned launches prohibitive, an unmanned system which uses remotely controllable manipulators and has a highly versatile maneuvering capability could be employed. The successful operation of such a system depends largely on the ability to rendezvous and dock with passive (i.e., noncooperating) satellites. In addition, station-keeping, or stand-off control will be a necessary capability for many applications.

It is the purpose of this paper to present a brief summary of the in-house effort conducted by General Electric Company in the areas of unmanned rendezvous, station-keeping and docking with passive satellites. This work was done as part of a broader activity, called ARMS (for Application of Remote Manipulators in Space), however the results are adaptable to many other designs and mission requirements. This paper describes the ARMS system concept and a suitable mission for establishing rendezvous, station-keeping and docking requirements. Preliminary designs for meeting these requirements are presented, along with computer analysis results.

ARMS SYSTEM CONCEPT

The ARMS system was conceived for the purpose of providing man's ability to do work in space without necessitating his presence directly at the scene. Figure 1 shows the principal elements of the ARMS system. These include a master station with human operators, one or more orbiting "slaves," a "tender" satellite and various work stations (satellites upon which work is to be performed).

The slave is essentially the mechanical counterpart of an untethered extravehicular man. It has a gimballed video camera subsystem, force feedback/position correspondence manipulators, a highly versatile maneuvering subsystem, plus necessary attitude control, communications, electrical power and thermal control subsystems. Figure 2 shows a stylized conceptual configuration of the slave vehicle.

The master station is located on the earth or in an orbiting space station. The operator, working in a shirtsleeve environment, has a head-aimed television system which allows him to move the slave's video camera in a natural manner and observe the space scene as the slave "sees" it. Force feedback/position correspondence manipulators. functionally similar to those on the slave, are also provided. The master operates these devices and the action is duplicated by the remote slave manipulators.[1,2]

The tender is essentially a maneuverable spacecraft which provides three basic functions as follows:

a. A communications relay between the master and slaves.

b. A space station "home" for the slaves when they are not in use.

c. Gross orbital changes beyond the capability of the slaves.

The concept of applying remotely controlled manipulator systems of the type described has been suggested for space work in the past. However, lack of development in the field of bilateral electric manipulators, notably for versions suitable for use in space, has impeded the development of these systems. Lately however, government and university interest has increased and encouragement to industry is being generated.[3,4] It is believed that increased interest and activity plus a background of more than 20 years in development of manipulators for hot labs, prosthetic devices and industrial mass transfer applications will ultimately lead to operational systems such as ARMS.

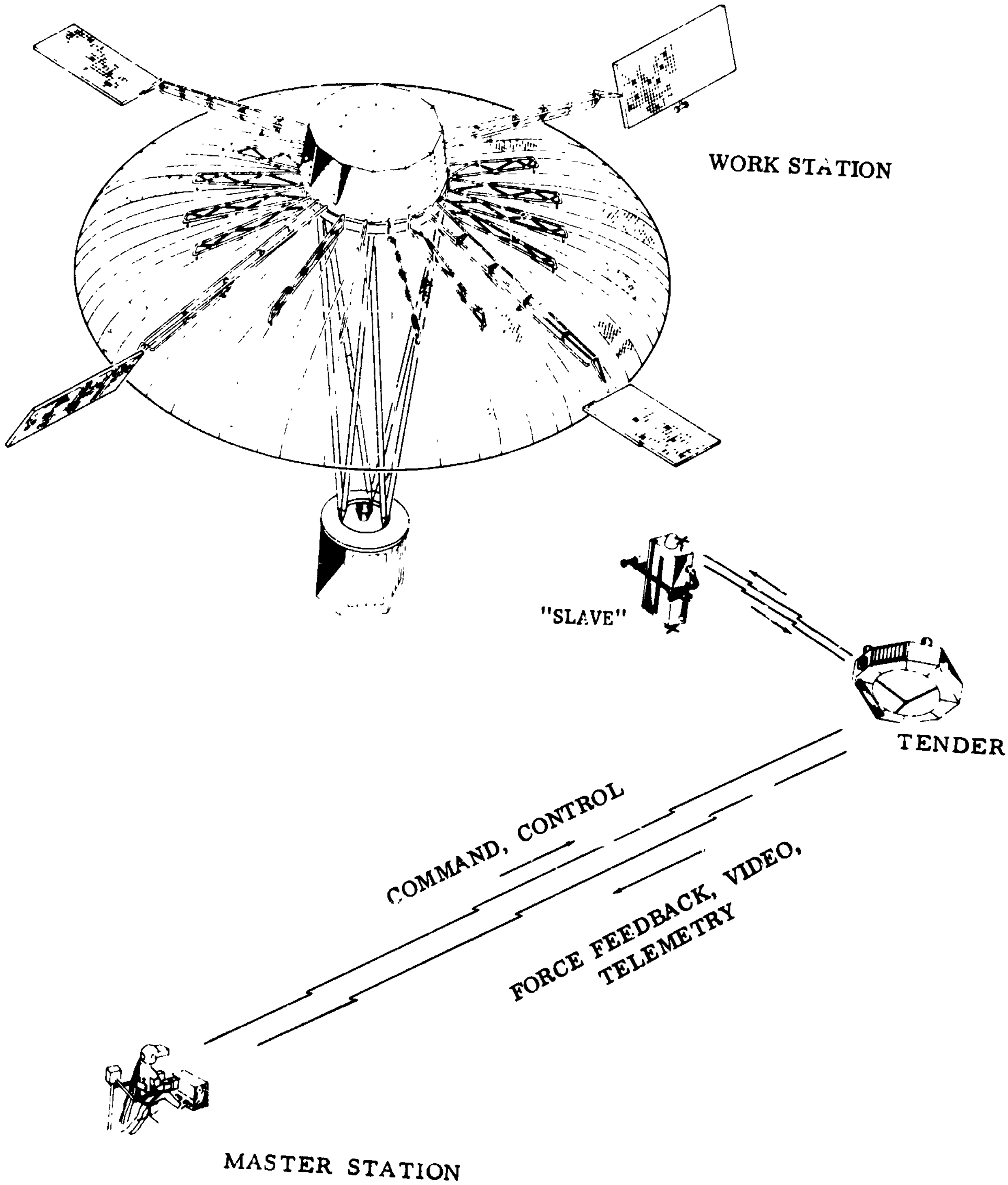

Figure 1. "ARMS" System Description

MISSION DESCRIPTION AND REQUIREMENTS

Potential missions for an ARMS system are probably as numerous and varied as manned extravehicular missions. Table 1 is a partial listing of such applicable missions.

Table 1. Partial Listing of Applicable ARMS Missions

- Satellite refurbishment
- Refuel, resupply and repair
- Maintenance of manned and unmanned observatories
- Salvage, retrieval and re-deployment of RTG fuel capsules, payloads, solar arrays, etc.
- Assembly of large structures
- Perform/support EVA and IVA tasks on manned space stations
- Astronaut emergency rescue
- Lunar and planetary exploration

A refurbishment mission was selected for detailed investigation of the ARMS concept because this mission could be readily planned, the nature of the manipulator tasks could be defined rather precisely in terms of their capability, system requirements and definition could be established, and the economics of the mission concept could be easily examined. The specific refurbishment mission chosen for investigation of rendezvous station-keeping and docking involves planned replacement of the mission payloads of unmanned satellites in synchronous, circular, equatorial orbit.

A stated goal for future satellites is long-life. However, achievement of operational lifetimes of say, 5 to 10 years can lead to a self-defeating situation caused primarily by payload obsolescence. Thus, in-orbit replacement of payloads (and expendibles replenishment, if required) could transform a healthy but obsolete satellite into a new and effective mission system at a significant cost saving. Synchronous orbit missions are selected because of (1) the probable high population density in this orbit regime in the post-1975 period (which significantly effects the economic justification of the system), (2) the low ΔV requirements for multi-satellite servicing and (3) the reduced logistics burden on ground support and communications systems.

Figure 3 illustrates the role of the ARMS system in the payload refurbishment mission. It is assumed that the space elements (slaves and tender) have previously been placed in orbit. Typically, the work station (particular satellite to be refurbished) is one of several synchronous communications satellites that form a network around the earth. The mission scenario is as follows: an unexpected breakthrough in communications technology, occurring after the satellite system had been established, will obsolete the entire network in about 2 years. It is, therefore, decided that rather than replace the existing, functionally operational, satellites or live with an obsolete system, a refurbishment program for replacing the outmoded payloads would be initiated.

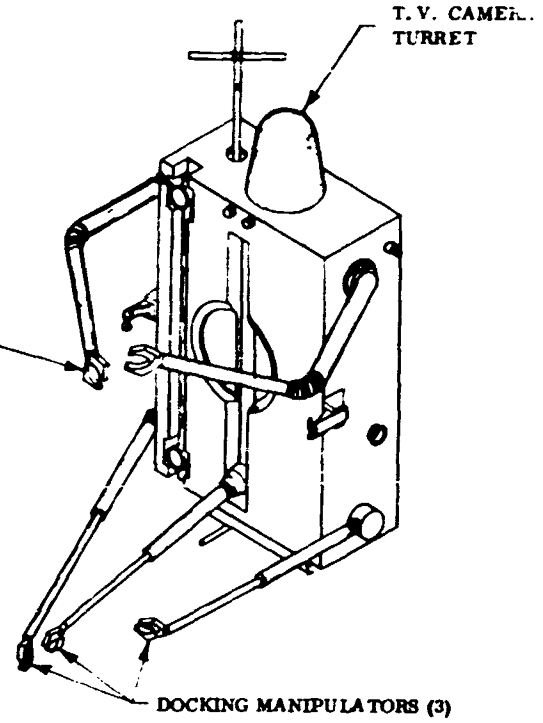

Figure 2. Slave Configuration

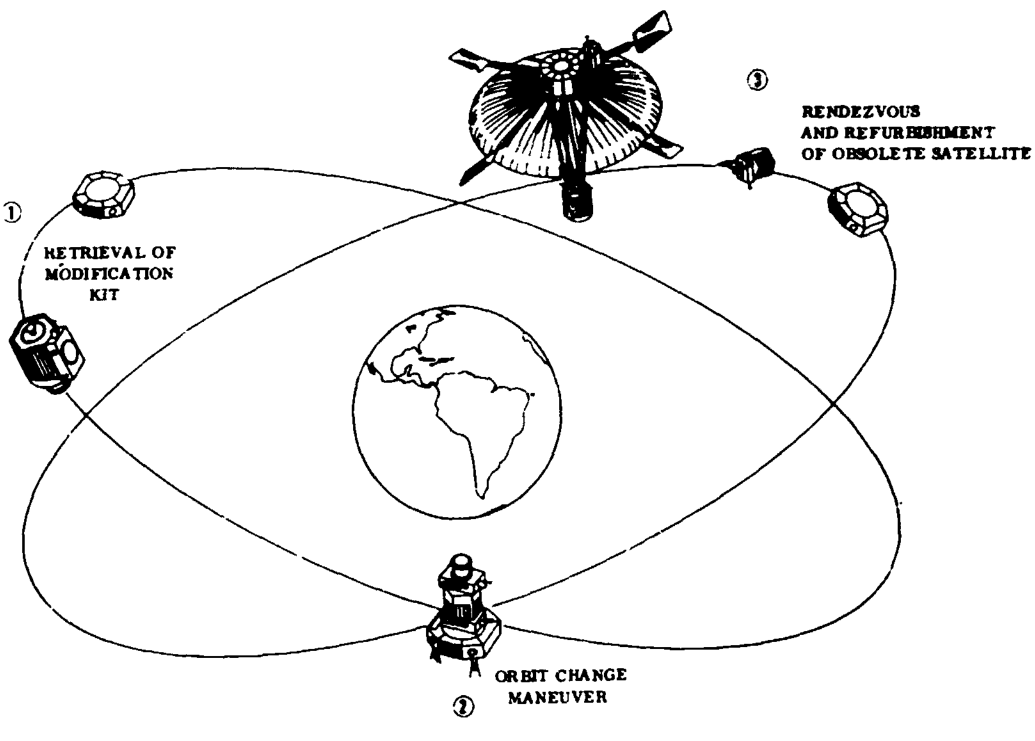

Figure 3. Refurbishment Mission

The mission sequence begins with the launch of a modification kit on a small launch vehicle and its injection into orbit in the vicinity of the ARMS system. The modification kit contains the new communications payload and fuel for replenishing both the work station and the ARMS system. The modification kit may be equipped with rendezvous and docking aids (in the form of beacons, docking collars, etc.) but the work stations are assumed to be not so equipped and are properly termed passive.

At point ① in Figure 3 the ARMS system is shown to have acquired the modification kit and completed its initial rendezvous. A slave has left the tender, docked with the kit and is about to return with it to the tender. When the slave and kit are secured to the tender, the initial phase of the mission is completed. The next phase, shown at point ②, consists of maneuvering ARMS plus the kit to the designated work station. This may involve an orbit plane change as shown in Figure 3 (exaggerated for clarity) as well as a large orbit angle change. Maneuver accuracy required for this phase of the mission is rather broad; a miss distance of a few thousand feet is acceptable.

The next phase is rendezvous to stand-off distance. The ARMS system is maneuvered to a range of between 200 and 500 feet from the work station to set up the conditions illustrated at point ③. After the required stand-off distance is achieved, the tender must maintain both its range and spatial relationship with the work station for extended periods (up to a few days, if necessary). A range variation between tender and work station of up to 10 percent of the nominal stand-off distance is permitted. An additional requirement is that the tender should be capable of being relocated. relative to the work station. both in range and aspect.

When the station-keeping mode is in operation. the slave departs from the tender and transports to the work station those portions of the modification kit required for replacing the mission payload and replenishing the fuel supply. Closure to within a maximum of three feet. but not touching the work station. is specified. The time to accomplish this is not critical for the baseline mission. but may be important for other missions. Thus. a nominal closure time of 2-5 minutes from a distance of 300 ft is indicated. The relative closing velocity at the terminal point is critical however. for docking to most work stations and is specified to be a maximum of 0.01 ft/sec within the rendezvous distance of 3 ft.

At the completion of the terminal rendezvous phase, the slave's docking manipulators are extended and the "hands" grasp suitable anchoring points on the work station. Analysis of typical large communications satellites indicate that the docking torques should be severely limited (to less than 1.0 in. -lb in some cases) to avoid damaging the delicate antenna structures. It is noted that three docking manipulators are specified for firmly supported the slave while the working manipulators perform their tasks.

The final phase of this mission is the return to the tender. Again, care must be taken to avoid damaging the work station upon departing. The slaves dock with the tender and the mission is completed.

GUIDANCE AND CONTROL STUDIES

Analytical guidance laws and implementation methods for terminal rendezvous and station-keeping are presented in this section. These are based on the ARMS system concept and typical mission described above, but may be readily applied to a broad range of configurations and mission requirements. Previous work in the area of terminal rendezvous guidance has been primarily involved with eliminating relative motion by arresting the line-of-sight rate and thrusting along the relative range. In most cases, throttleable engines are assumed. The rendezvous guidance philosophies and implementation schemes presented here, however, are based on constant thrust engines having multistart capability. The unique operational task of station-keeping has not been previously studied in detail comparable with rendezvous, so that effective models for the station-keeping dynamics are not available. Two approaches for achieving station-keeping are described here.

A. Terminal Rendezvous

1. Analytical Guidance Laws

The equations of motion are developed in terms most suitable for guidance and control implementation. Figure 4 shows the relative positions of the slave and work station.

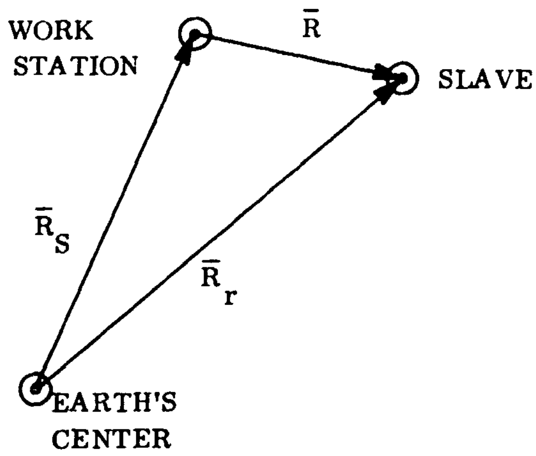

Figure 4. Position Relationships

For the mission parameters considered here, it is most reasonable to consider differential gravitational effects as small disturbances. Thus the resultant equations of motion are:

$$\frac{d^2\bar{R}}{dt^2} = \frac{\bar{T}}{m} \qquad (1)$$

where: T = Slave's thrust capability

m = Slave's mass

Equation 1 is quite useful for distances, R, of up to several thousand feet. but must be described in terms of coordinates suitable for guidance implementation. Therefore, let $\bar{r}$ be a unit vector in the range direction (line-of-sight) so that $\bar{R} = R\bar{r}$, $\bar{n}$ be a unit vector normal to the range direction, and define a third unit vector, $\bar{\sigma} = \bar{n} \times \bar{r}$, to complete the coordinate set.

If $\bar{R}$ is rotating with angular velocity $\bar{\omega}$ with respect to an inertial frame, it is readily shown that:

$$\frac{d^2\bar{R}}{dt^2} = (\ddot{R} - R\omega_n^2)\,\bar{r} + (R\dot{\omega}_n + 2\dot{R}\omega_n)\bar{\sigma} + (R\omega_n\omega_r)\,\bar{n} \qquad (2)$$

where the dot notation is used for scalar derivitives.

When $\omega_n = 0$, $\bar{R}$ and $\frac{d\bar{R}}{dt}$ are aligned and a collision course is assured. Therefore the variables of interest are R (relative range) and ω_n (line-of-sight rate). Combining Equations 1 and 2 results in:

$$\ddot{R} - R\omega_n^2 = (\frac{\bar{T}}{m})\cdot\bar{r} \equiv A_r. \qquad (3)$$

and

$$R\dot{\omega}_n + 2\dot{R}\omega_n = (\frac{\bar{T}}{m})\cdot\bar{\sigma} \equiv A_\sigma \qquad (4)$$

These are the general equations of motion for terminal rendezvous. Two guidance laws are now considered. The first is based on reducing ω_n to zero so that $R\omega_n^2$ decays exponentially. Two thrust engines (for A_σ and A_r) are used. The second employs only one engine (for A_σ) to build ω_n to a value such that $R\omega_n^2$ reduces the range R exponentially.

a. Two Engine Method

This method is based on the application of a control thrust, A_σ, such that:

$$A_\sigma = -K_\sigma R\omega_n \qquad (5)$$

If $(A_\sigma)_{max} \geq A_\sigma$ and if $K_\sigma + \frac{2\dot{R}}{R} > 0$.

Equation 4 reduces to:

$$R^2\omega_n = R_o^2\,\omega_{n_o}\,e^{-K_\sigma t} \qquad (6)$$

where R_o is the initial relative range, and ω_{n_o} is the initial line-of-sight rate.

Then $R^2\omega_n$ decays to zero with time constant $\frac{1}{K_\sigma}$ and $\omega_n \to 0$. With good ω_n control (i.e., $\omega_n \cong 0$) thus achieved. Equation 3 reduces to:

$$\ddot{R} = A_r \qquad (7)$$

b. One Engine Method

The absence of radial thrust modifies Equation 3 to:

$$\ddot{R} - R\omega_n^2 = 0 \qquad (8)$$

It is now assumed that A_σ is such as to bring ω_n to some constant value, ω_c, in a very short time. Then Equation 3 is further modified to $\ddot{R} - R\omega_c^2 = 0$. If ω_c is chosen such that $\omega_c = -\dot{R}_o R_o^{-1}$, it may be shown that $\dot{R}(t) = R_o e^{-\omega_c t}$.

Thus, R(t) and $\dot{R}$(t) decay, and rendezvous may be achieved. This may be obtained by the application of thrust A_σ given by:

$$A_\sigma = 2\dot{R}\omega_n - K_\sigma R(\omega_n + \frac{\dot{R}}{R}) \qquad (9)$$

Substituting Equation 9 into Equation 4 gives

$$\dot{\omega}_n = -K_\sigma (\omega_n + \frac{\dot{R}}{R}) \tag{10}$$

2. Thrust Profiles

A computation cycle, t_c, is defined such that information about range (R), range-rate ($\dot{R}$) and line-of-sight rate (ω_n) is updated every t_c seconds. Within the time period t_c, the appropriate engines are fired for variable durations ranging from 0 to t_c. Analyses have shown that line-of-sight control (A_σ thrusting) should be done with two thrust levels in order to achieve good control. Range control (A_r thrusting) can be adequately effected with a single thrust level.

a. Two Engine Method

The line-of-sight control thrust profile is expressed as:

$$A_\sigma(t) = -(\operatorname{sgn}\omega_{n_i})\, A_\sigma, \text{ for}$$

$$t_i < t \le (t_i + \Delta_i), \text{ and}$$

$$A_\sigma(t) = 0, \text{ for} \tag{11}$$

$$(t_i + \Delta_i) \le t < (t_{i+1})$$

where: $t_{i+1} - t_i = t_c$, the computation cycle time, and Δ_i, the time for which a thrust A_σ is applied for the i^{th} computation cycle, $i = 1, 2, 3, \ldots\ldots$

The thrust on-time at level A_{σ_1} is:

$$\Delta_i = t_c, \text{ if } \left| K_\sigma R_i \omega_{n_i} A_{\sigma_1}^{-1} \right| \ge 1$$

$$\Delta_i = t_c \left| K_\sigma R_i \omega_{n_i} A_{\sigma_1}^{-1} \right|, \text{ if} \tag{12}$$

$$\left| K_\sigma R_i \omega_{n_i} A_{\sigma_1}^{-1} \right| \ge \delta_1$$

The thrust on time at level A_{σ_2} is:

$$\Delta_i = t_c \left| K_\sigma R_i \omega_{n_i} A_{\sigma_2}^{-1} \right|, \text{ if}$$

$$\left| K_\sigma R_i \omega_{n_i} A_{\sigma_2}^{-1} \right| \ge \delta_2$$

$$\Delta_i = 0, \text{ otherwise} \tag{13}$$

$A_{\sigma_1} > A_{\sigma_2}$ are the two thrust levels, and quantities δ_1 and δ_2 (less than unity) are chosen from consideration of the minimum on-time of a thruster. It is apparent that the smaller the computation cycle time t_c, the more accurate will be the control.

The range control thrust profile is determined from consideration of factors such as fuel consumption and time-to-rendezvous. If the rendezvous time, T_f, is specified and if a minimum fuel path is desired, the thrust profile and resulting range versus range-rate profile solutions are shown in Figure 5 (a) and (b), respectively.

In Figure 5 (a), the period T_1 is equal to:

$$T_1 = \frac{1}{2}\left\{ (T_f - \frac{\dot{R}_o}{A_{r_m}}) - \left[\left(T_f + \frac{\dot{R}_o}{A_{r_m}}\right)^2 - \frac{4}{A_{r_m}}\left(R_o + \frac{\dot{R}_o^2}{2A_{r_m}}\right) \right]^{1/2} \right\} \tag{14}$$

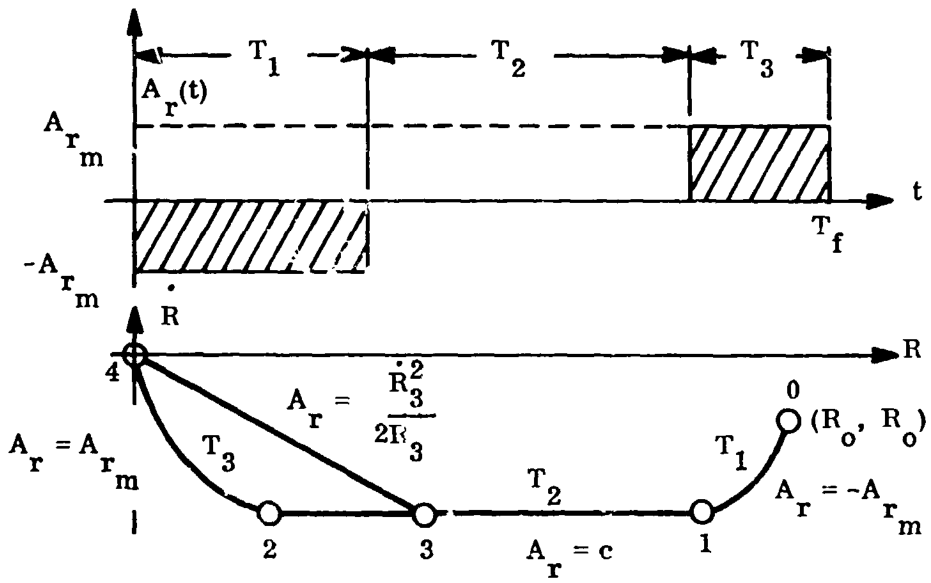

Figure 5. Thrust and Range Versus Range-Rate Profiles for Two Engine Approach

Coasting period, T_2, is equal to:

$$T_2 = T_f - 2T_1 - \dot{R}_o (A_{r_m}^{-1}) \quad (15)$$

and

$$T_3 = T_1 + \dot{R}_o A_{r_m}^{-1} \quad (16)$$

In practice it would be safer to apply positive thrust earlier than at point 2 (Figure 5 (b)) such as at arbitrary point 3. A thrust $A_r = \frac{\dot{R}_3^2}{2R_3} < A_{r_m}$ will effect rendezvous with the same amount of fuel but will take longer to accomplish. The scheme proposed, however, assumes that rendezvous time is not critical, and utilizes time modulation of the thrust-on period to obtain equivalent thrust levels $A_r \le A_{r_m}$ with constant level thrust engines. This method has the advantage that the rendezvous point is approached with a smaller acceleration, and hence, errors due to thrust level uncertainties are smaller. The approach is shown by the Path 0-1-3-4, and consists of applying maximum negative thrust until point 1 is reached, such that $\frac{\dot{R}_1^2}{2R_1} \le K A_{r_m}$. R_1 and $\dot{R}_1$ are the position and velocity at point 1, and K (less than unity) depends on the minimum on-time of the thruster. Thus the thrust profile is expressed as

$$A_r(t) = -A_{r_m}, \quad 0 < t \le T_1,$$
$$A_r(t) = 0, \quad T_1 < t \le T_2 \quad (17)$$

where: T_1 is given by Equation 14 and T_2 is found from:

$$\delta_3 A_{r_m} = \frac{\dot{R}^2(t)}{2R(t)} \quad (18)$$

If δ_3 is unity, the rendezvous time is T_f. For smaller values of δ_3, the rendezvous time is larger than T_f. From point 3 until rendezvous is achieved, the radial thrust profile is:

$$A_r(t) = A_{r_m}, \text{ for}$$
$$t_i < t \le (t_i + \Delta_i) \text{ and} \qquad (19)$$
$$A_r(t) = 0, \text{ for}$$
$$(t_i + \Delta_i) < t \le (t_{i+1})$$

where: $t_c = t_{i+1} - t_i$ and

$$\Delta_i = t_c \dot{R}_i^2 (2 R_i A_{r_m}^{-1})$$

b. One Engine Method

Implementation of the two-level thrust A_σ of the one engine scheme is as as follows. Let

$$A_i = (2\dot{R}_i - K_\sigma R_i)\, \omega_{n_i} - K_\sigma \frac{\dot{R}_i}{R_i} \qquad (20)$$

Then

$$A_\sigma(t) = (\operatorname{sgn} A_i)\, A_{\sigma_m}, \text{ for}$$
$$t_i \le t \le (t_i + \Delta_i) \text{ and} \qquad (21)$$
$$A_\sigma(t) = 0, \text{ for}$$
$$(t_i + \Delta_i) \le t \le (t_{i+1})$$

where:

$$\Delta_i = t_c, \text{ if } |A_i| > A_1$$
$$\Delta_i = t_c\, |A_i A_1^{-1}|, \text{ if}$$
$$|A_i A_1^{-1}| \ge \delta_4 \qquad (22)$$

If $|A_i A_1^{-1}| < \delta_4$, then

$$\Delta_i = t_c\, |A_i A_2^{-1}| \text{ if } |A_i A_2^{-1}| \ge \delta_5$$
$$\Delta_i = 0, \text{ otherwise} \qquad (23)$$

$A_1 > A_2$ are the two thrust levels and δ_4 and δ_5 (constants less than unity) are chosen from consideration of minimum on-time.

Sensor limitations give a minimum threshold region in the $R - \dot{R}$ plane (see Figure 6) and it is desirable to reach this region so that when the thrusters are shut off, both R and $\dot{R}$ are acceptably small. The slave is given an initial vel city $\dot{R}_0$ so that the starting poi.t is as close to the rendezvous path as possible.

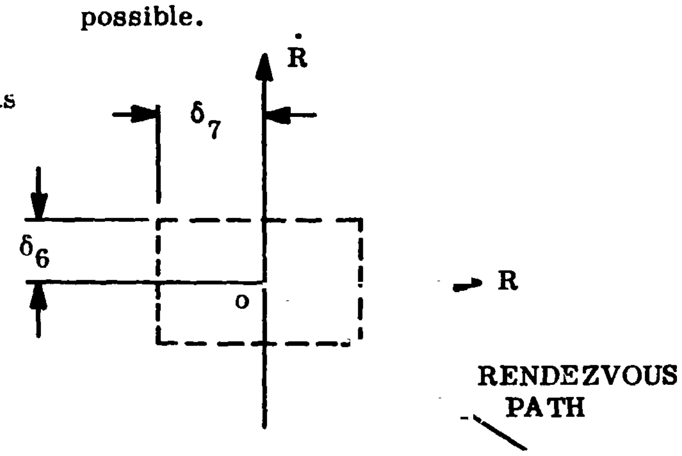

Figure 6. One Engine Rendezvous Profile

3. Implementation

A schematic view of the slave's rendezvous guidance implementation concept is shown in Figure 7. The translational thrust directions are computed from the rangefinder tracker's gimbal angles and rates if the slave is held inertially fixed.

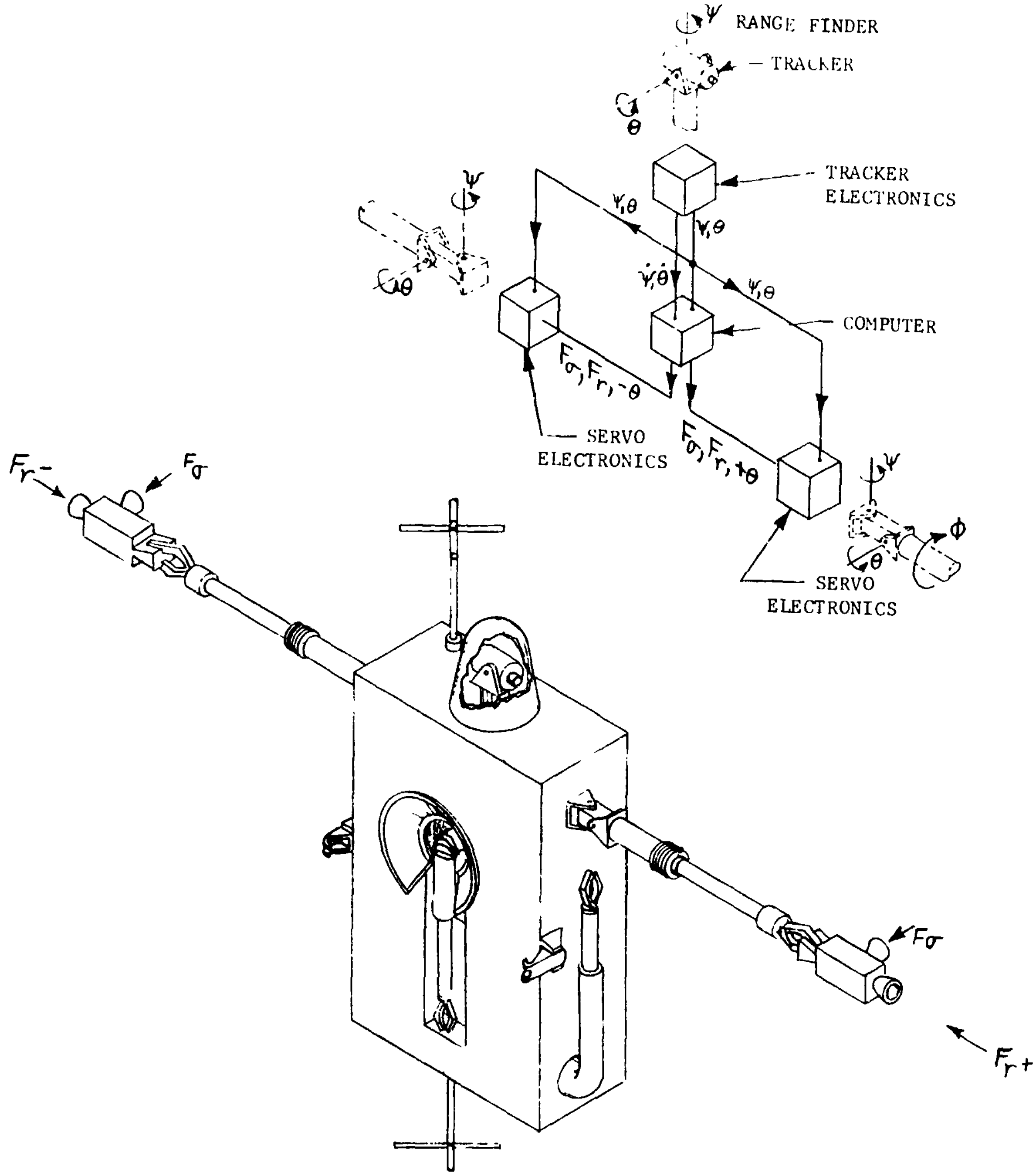

Figure 7. Terminal Rendezvous Guidance Implementation

The thrust engines can readily be vectored in the required directions by means of the working manipulators.

If the slave's attitude is held inertially fixed, the tracker's gimbal angles (θ, ψ) geometrically locate the range vector $\bar{r}$ with respect to the inertial frame. Furthermore the angular velocity vector, $\bar{\omega}_n$, can be expressed in terms of $\dot{\psi}$ and $\dot{\theta}$, as

$$\omega_n = (\dot{\psi}^2 \cos^2 \theta + \dot{\theta}^2)^{1/2} \qquad (24)$$

and

$$\beta = \text{Sin}^{-1} \left(\frac{\dot{\psi} \cos \theta}{\omega_n}\right), \ \dot{\theta} > 0$$

or

$$\beta = \pi \ (\text{sgn}\ \dot{\psi}) - \text{Sin}^{-1} \left(\dot{\psi} \ \frac{\cos \theta}{\omega_n}\right), \ \theta < 0$$

(25)

where β is the angle between $\bar{n}$ and the θ gimbal axis.

The direction of $\bar{\sigma}$ can be located by a rotation $\psi = (\frac{\pi}{2} - \beta)$ about the $\bar{r}$ direction. The control thrusts A_r and A_c will automatically be applied in the correct directions if the slave's working manipulators are slaved to the tracker's angles and the normal thrust engines servoed to the angle ψ.

B. Station-Keeping

Station-keeping is the continuous (or bounded) maintenance of the relative position vector between two bodies in space. Without station-keeping guidance, the gravity gradients between the two bodies will cause drift, even though the orbital parameters appear almost numerically identical. Although the distances involved for the A.tMS mission are of the order of several hundred feet, solutions are presented for the more general case of distances of several hundred miles. Thus, gravitational effects are not neglected; however second order perturbations (higher order harmonics) are not considered significant.

The linearized equations of motion are given by:

$$\frac{d^2\bar{R}}{dt^2} + \alpha^2 \left[\bar{R} - 3R_S^{-2} (\bar{R}_S \cdot \bar{R}) \bar{R}_S \right] = \frac{\bar{T}}{m} \qquad (26)$$

where: $\alpha^2 = \mu R_S^{-3}$

μ = GM = Gravitational Parameter

G = Universal Gravitational Constant

M = Mass of Force Center

A convenient coordinate system for bodies in circular orbits is shown in Figure 8.

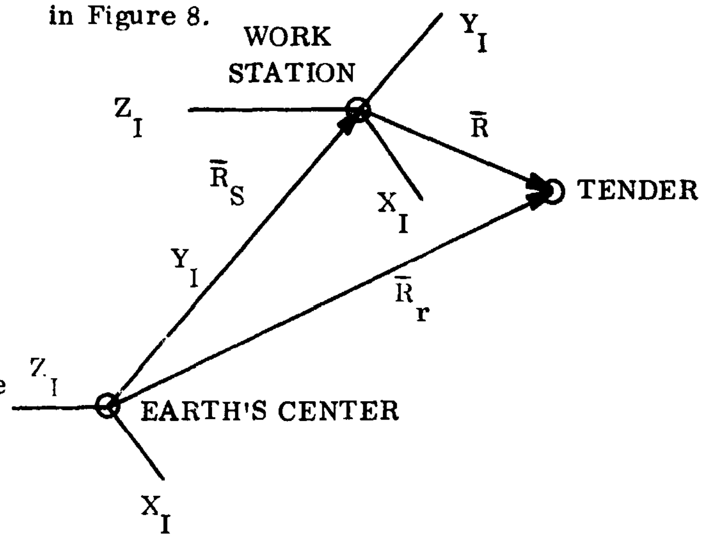

Figure 8. Station-Keeping Coordinate System

In this coordinate system the equations of motion reduce to:

$$\ddot{X} - 2\alpha\dot{Y} = A_X$$

$$\ddot{Y} + 2\alpha\dot{X} - 3\alpha^2 Y = A_Y \qquad (27)$$

$$\ddot{Z} + \alpha^2 Z = A_Z$$

where: $A_X = \frac{T_X}{m}$ $A_Y = \frac{T_Y}{m}$, $A_Z = \frac{T_Z}{m}$

The selected thrust laws for station-keeping are given as:

$$A_X = -K_x(X - X_S) - K_{\dot{x}}\dot{X}$$

$$A_Y = -(K_Y + 3\alpha^2)(Y - Y_S) - K_{\dot{Y}}\dot{Y}$$

$$A_Z = -K_z(Z - Z_S) - K_{\dot{Z}}\dot{Z} \qquad (28)$$

The various K's are chosen for stability and desired response characteristics, and X_S, Y_S, Z_S represent the station coordinates. Utilization of Equations 27 and 28 results in a stable system which will acquire the work station from any reasonable set of initial conditions to yield the steady-state results:

$$X = X_S,\ \dot{X} = 0$$

$$Y = Y_S,\ \dot{Y} = 0 \qquad (29)$$

$$Z = Z_S,\ \dot{Z} = 0$$

In order to conserve fuel. the choice of a dead-band thrusting approach seems ideal (especially where precise station-keeping is not essential). Thus Equations 28 and 29 are modified to give:

$$A_\epsilon = -\left[K_\epsilon(\epsilon' - \epsilon_S) + K_{\dot{\epsilon}}\dot{\epsilon}\right],$$

$$\left|K_\epsilon(\epsilon' - \epsilon_S) + K_{\dot{\epsilon}}\dot{\epsilon}\right| > \delta_8 \qquad (30)$$

and

$$A_\epsilon = 0,\ \left|K_\epsilon(\epsilon' - \epsilon_S) + K_{\dot{\epsilon}}\dot{\epsilon}\right| \le \delta_8$$

where: $\epsilon' = \epsilon$ for $|\epsilon| < \epsilon_S$

$\epsilon' = \epsilon_S(\mathrm{sgn}\ \epsilon)$, for $|\epsilon| \ge \epsilon_S$ and

where $\epsilon = X, Y, Z$ and ϵ_S is a positive constant determined from maximum allowable velocity. Quantity δ_8 is chosen from definition of allowable stand-off errors.

A second approach for development of station-keeping guidance laws is based on an analytical model of station-keeping guidance conceived by means of the state-space theory of astrodynamics. 5, 6, 7, 8, 9, 10, 11 Figure 9 illustrates the application of this method, involving the basic parameters R_{LS} (relative range along the lone-of-sight) and χ (direction angle referred to the local horizor of the powered vehicle).

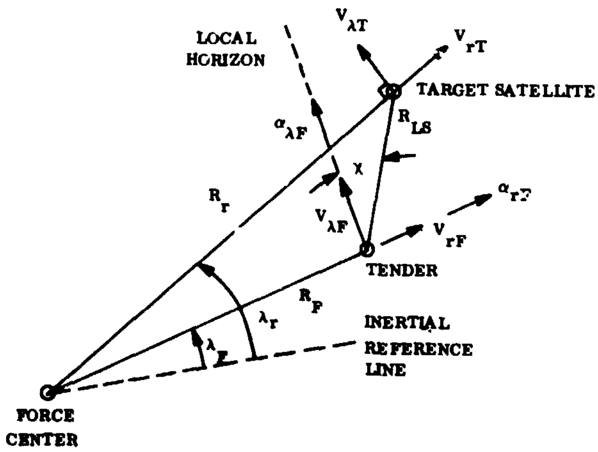

Figure 9. Alternate Station-Keeping Concept

Station keeping may be provided by the following guidance laws for two-dimensional motion in the orbital plane containing the target satellite and the power vehicle:

$$\alpha_{rF} = \dot{D} = \text{acceleration applied along radius } R_F, \tag{31}$$

$$\alpha_{\lambda F} = 2\dot{\lambda}_F D = \text{acceleration applied normal to the radius and in the orbital plane.} \tag{32}$$

$$\text{where: } D = \frac{d}{dt}\left(R_{LS} \sin \chi\right) \tag{33}$$

Sensed data consists of three inputs: the relative range (R_{LS}), the local direction angle (χ) of the relative range vector, and the inertial angular rate ($\dot{\lambda}_F$) of the powered vehicle about the force center.

COMPUTER RESULTS

Computer verifications of the proposed terminal rendezvous and station-keeping methods were performed.

Figure 10(a) shows range versus range-rate profiles for terminal rendezvous using the two-thrust engine approach. The initial range was assumed to be 300 feet, and initial rates of 10, 20, and 30 ft/sec were used. As shown in (b) of Figure 10, satisfactory rendezvous is achieved. The times to rendezvous are 56, 43, and 26 seconds, respectively, for the initial rates of 10, 20, and 30 ft/sec. Similar results are shown for the one-thrust engine approach in Figures 11 (a) and 11 (b). The times to rendezvous are 175, 71, and 31 seconds, respectively, for initial rates of 10, 20, and 30 ft/sec. For the same initial closing velocity, the rendezvous time for the two-engine scheme is smaller but fuel consumption is higher than for the one-engine approach. Figures 12 and 13 show the rendezvous paths in the orbital plane containing the work station and ARMS system.

Station-keeping results are shown in Figure 14, and indicate that the effect of cross-axis coupling and gravity gradient differences are negligible for the close-in (300 feet, nominal) standoff situation. The simulation shows that a dead-band of 5 feet in each axis is readily attainable. This is well within the specified 10 percent error of the nominal distance.

DOCKING WITH REMOTE MANIPULATOR SYSTEMS

The advantage of a bilateral, or "force-feedback," master/slave manipulator system is that it makes it surprisingly easy to perform tasks from a remote location. The human operator functions entirely within his natural frame of reference. Furthermore, no complex controls are required in the master station since the basic control system is the operator's own natural sensory system. Thus master/slave manipulators could be very readily applied to meet the requirements for docking and anchoring the slave vehicle to the mission work station.

The fundamental capabilities for a docking/anchoring system to meet the rather severe mission requirements implied earlier are listed in Table 2.

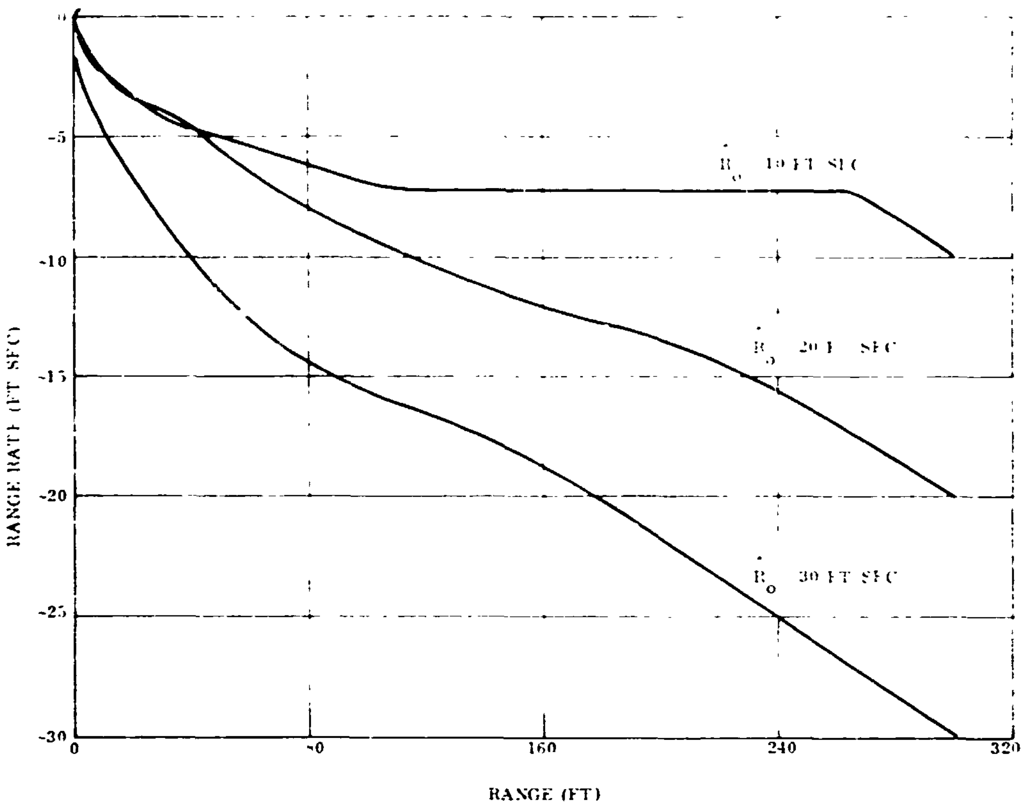

Figure 10a. Two Engine Rendezvous

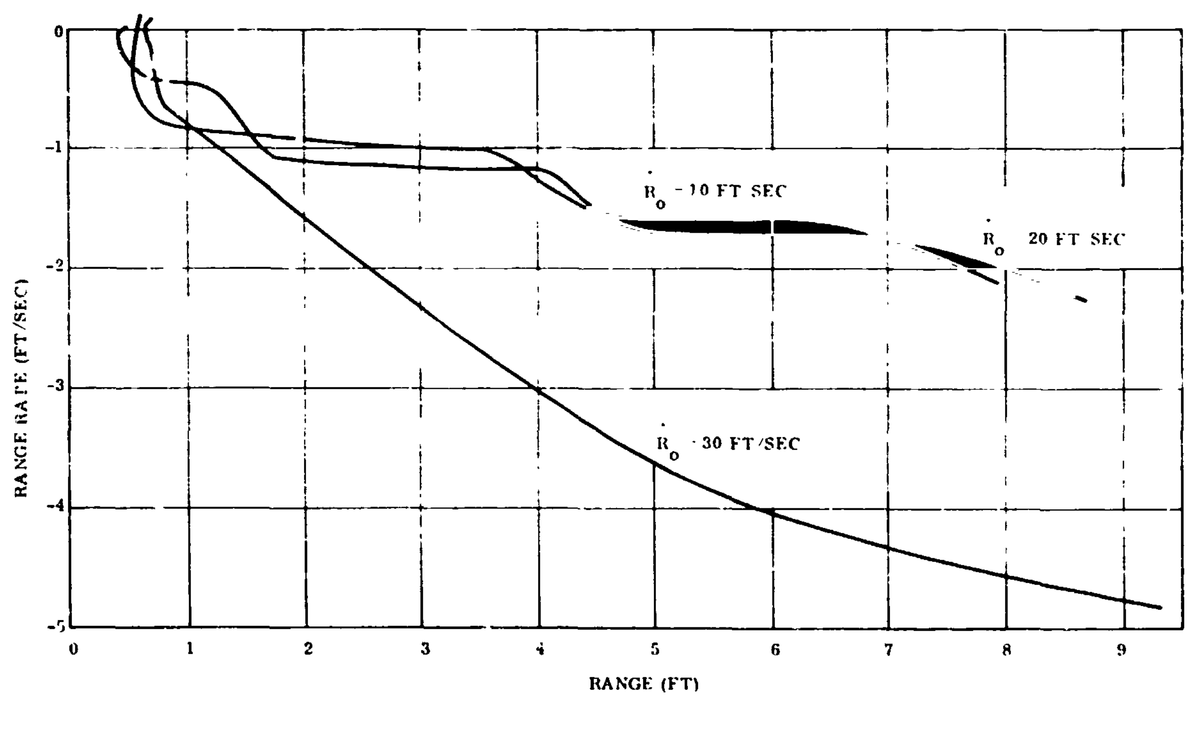

Figure 10b. Two Engine Rendezvous (Close-In Details)

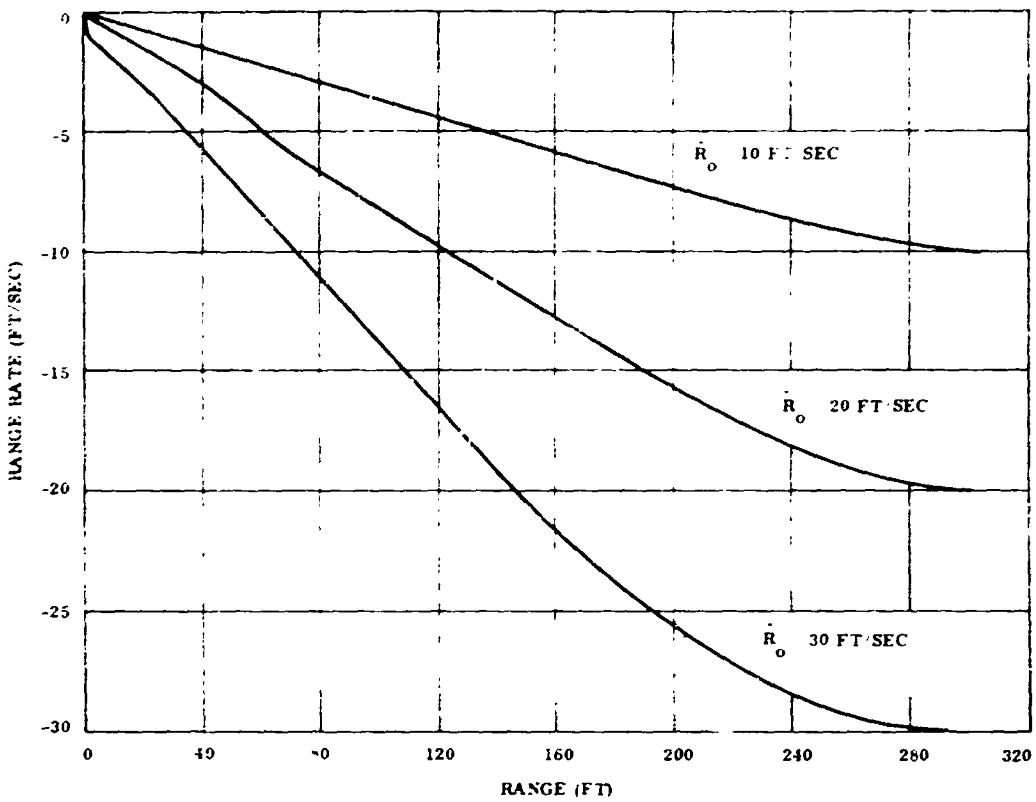

Figure 11a. One Engine Rendezvous

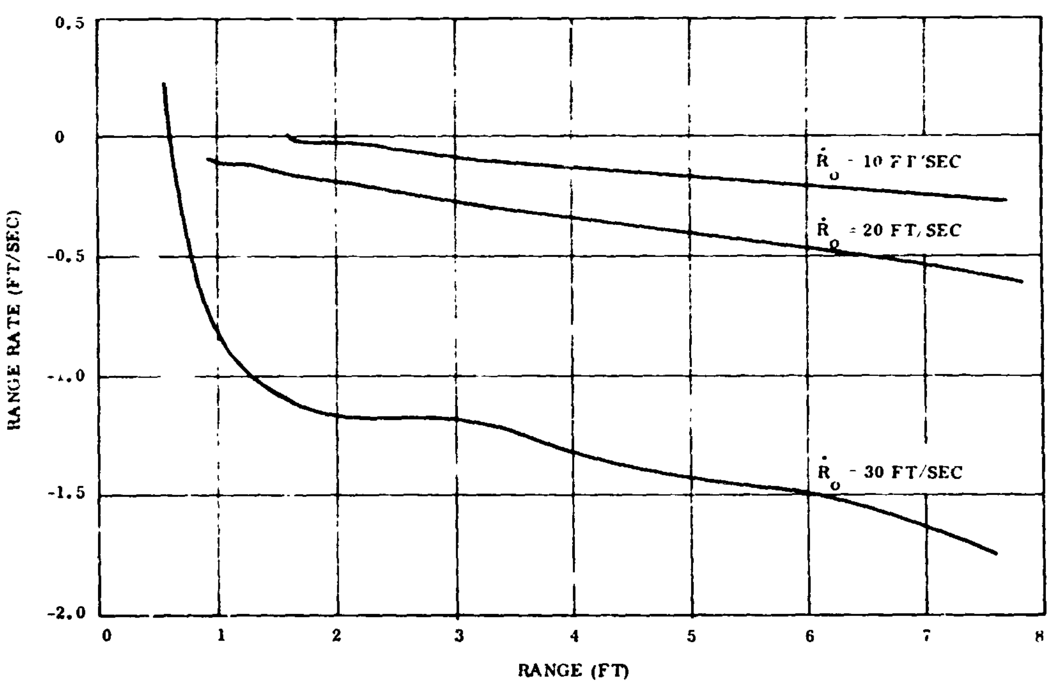

Figure 11b. One Engine Rendezvous (Close-In Details)

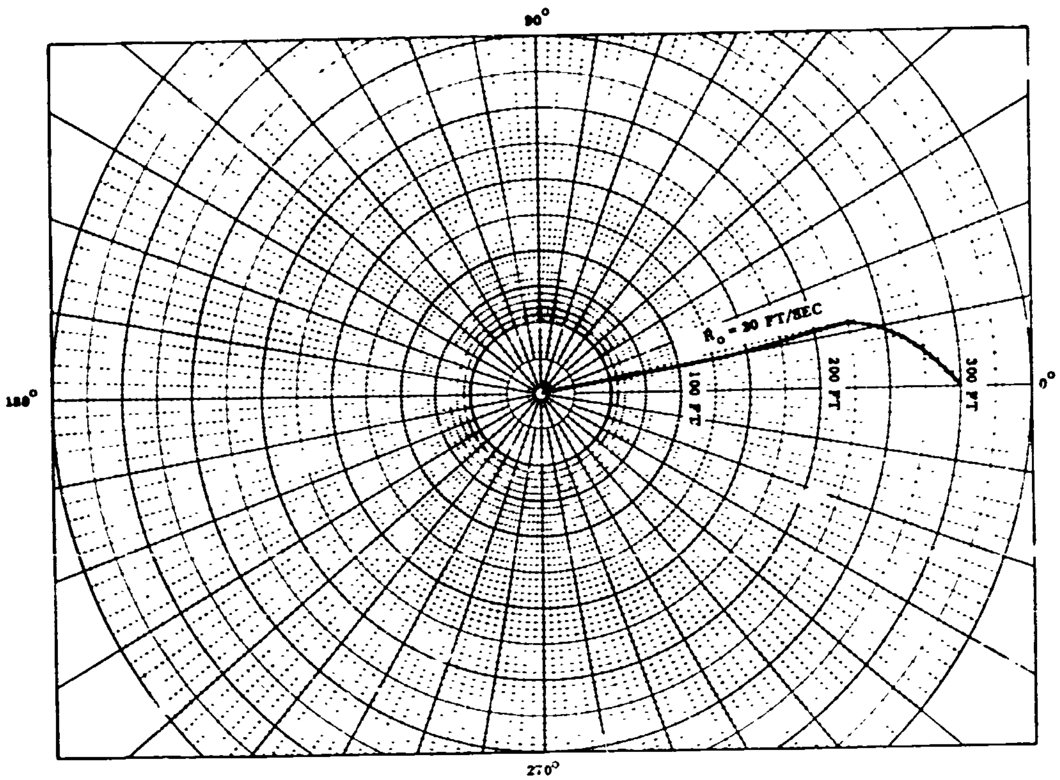

Figure 12. Two Engine Rendezvous-Range Versus Line of Sight Angle

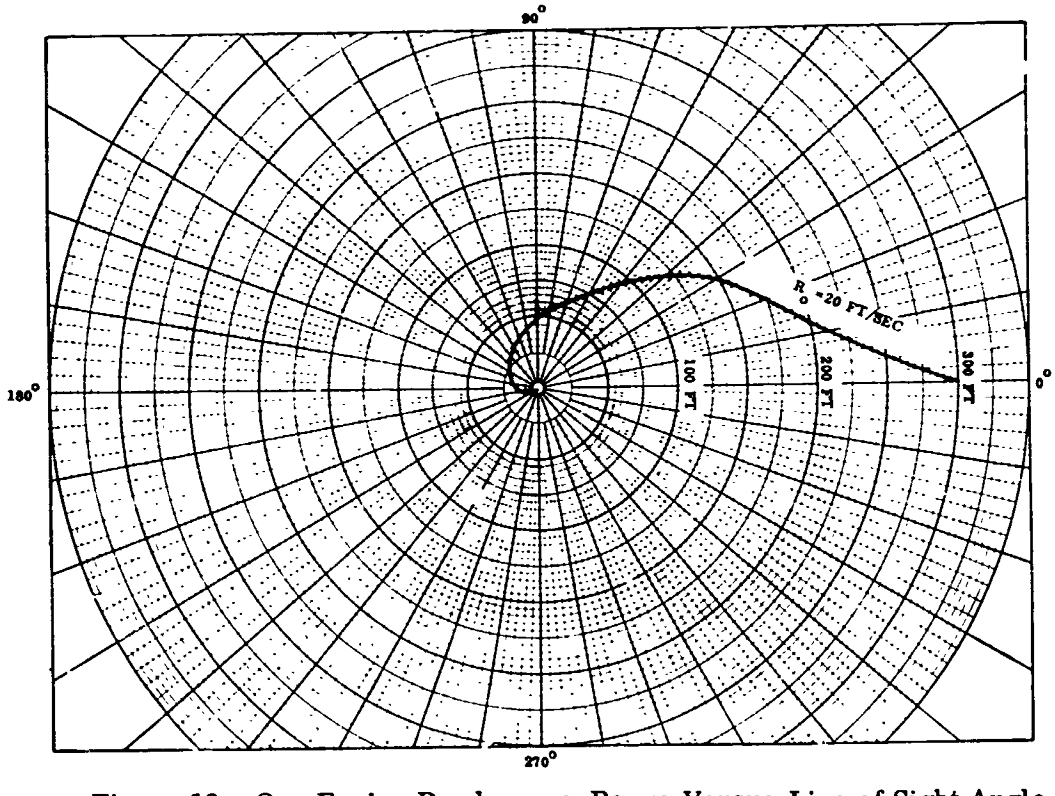

Figure 13. One Engine Rendezvous-Range Versus Line of Sight Angle

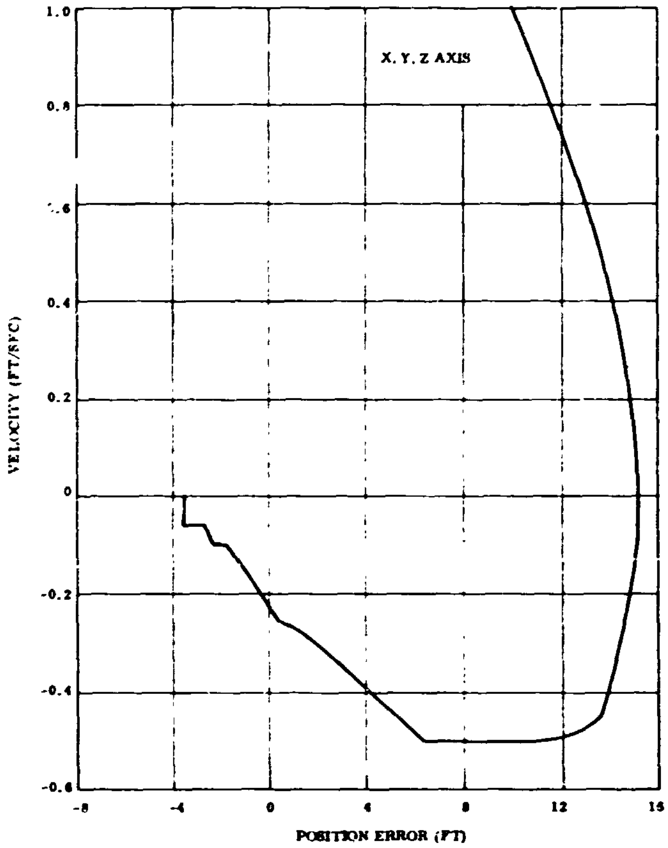

Figure 14. Station Keeping-Position Error Versus Velocity

Table 2. Required Capabilities for Docking/Anchoring

- Visual observation of activity
- Precise control of remote positioning
- Force feedback, or "feel"
- Suitable terminal devices for handling
- Compatibility with a wide variety of work station configurations and constraints

A man-equivalent android, such as the ARMS system's slave, comes quite close to being an optimum manipulation system for docking and anchoring.

The essential elements of the ARMS system for docking and anchoring to the work station are (1) a video system for steroscopic ranging and imaging, (2) bilateral multi-joint electric manipulators, (3) an RF link for two-way communications, and (4) a trained operator working under natural conditions. An attitude sensing and control subsystem is assumed.

Sensory feedback to the operator is essential to all remote operations. For performing the task of docking in space it appears that observation and force reflection are sufficient sensory information. Visual observation is achieved utilizing the slave's dual TV camera system. Control is effected by having the master operator equipped with a head harness so that as he moves his head the slave gimbals faithfully follow the motion. This has been physically demonstrated to be quite natural and results in a realistic feeling of visual presence.

Once having achieved closure with the work station, the master operator can now remotely attach the docking manipulators to the work station. Since the slave manipulators duplicate the motion of the master, it is only necessary for the operator to, in effect, "reach out and grab." By virtue of the ability to observe and to feel the forces involved, the operator could effect a very gentle docking. By properly setting the force feedback levels, the energy involved in the docking "impact" could be smoothly absorbed by the operator just as if he was directly involved. In addition, the structural design of the docking manipulators would be such as to assist in absorbing docking energy. The terminal devices for docking manipulators need not be significantly different from those of the working manipulators (i.e., parallel jaw tongs).

One facet of remote docking control which may be a potential problem is that of time-delay. The time-delay condition exists because of the spatial separation between the master and slave. Considerable study has been done, and is being continued, in determining suitable operating procedures. Two methods are receiving study emphasis. These are (1) a "move and look" method in which the master performs his motions in discrete steps, observes the corresponding motion of the slave, and then moves again, and (2) provide a high feedback gain which then limits the master to slow motion. In either case, operator training and ability are critical factors. Conclusions which may be made at this time are that time-delayed manipulation is feasible and that the use of feedback makes the task much easier to perform.

CONCLUSIONS

An unmanned remote manipulator system (called ARMS) for doing useful space work tasks has been described. All aspects of this system are essentially state of the art, although some components, such as the slave manipulators, are not presently available as qualified space hardware. The basic groundwork has been laid, however, and it is believed that an operational ARMS system of the type described could be flown within 4 or 5 years.

A baseline mission for the ARMS system, one of many possible missions, has been presented. This mission, refurbishment of obsolete payloads of synchronous orbit satellites, is considered to be feasible and cost effective. The baseline mission was used to formulate requirements for unmanned rendezvous, station keeping, and docking, although the resulting concepts and techniques are applicable to a wide variety of mission and system requirements. Terminal rendezvous and station keeping using closed-loop control techniques have been described considering alternate approaches. Preliminary results of computer simulations are encouraging and lead to the conclusion that the techniques can be demonstrated within the next few years. The unique feature is that of utilizing constant-level thrust engines, but time modulating the thrust-on times in order to approximate the effect of throttleable engines.

Soft docking and anchoring to passive satellites have been described utilizing man's natural sensing and motor ability to control a remote docking/anchoring manipulator system. The system provides visual and force feedback information to the ground operator for ease of docking. The problem of time-delayed operation is believed to be a minor one if proper operational techniques are employed.

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HANDS-FREE PRECISION CONTROL FOR EVA*

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SUMMARY: '.. ensive research into the use of the human foot-balanc...g reflex for control of vehicles in the one-g environment has led to an extrapolation of the concept to its use for Extra Vehicular Activity (EVA), the maneuvering of free-floating spacemen. An exploratory program in which zero-gravity was simulated for three degrees of freed(n in the horizontal plane has proved the basic utility of the idea and provided a model for the preliminary desi;.. of a prototype, EVA control system.

BACKGROUND

The use of the human balancing reflex for vehicular control was publicly propounded by Charles Zimmerman of the NACA in the early 1950's. His central thesis was that the learned pattern of reflexes used by a person in standing is essentially the same as that required to balance a force-vector supported platform, and hence should be directly applicable to the control of hovering type vehicles. This concept and its simple but dramatic demonstration by Zimmerman[1] piqued the imagination of a great many aeronautical engineers and led shortly to several experiments with free-flying platforms of various sorts. There were, for example, the ducted-fan machine of Hiller,[2] the stand-on helicopter of DeLackner (the "Aerocycle" tested by Princeton University[3]), and several research-oriented devices built by the NACA.[4,5]

Since that initial period of activity, engineering interest has waned, probably for lack of definitive information on optimum usage of the human balancing reflex, and the concept has made only sporadic appearances in one or another embodiment; for example, the "lunar scooter" studied by

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North American,[6] and the "Jet-Shoes" developed by NASA - Langley.[7,8] Grumman Research, however, has maintained a constant enthusiasm for the concept and has kept a small but steady effort going in the study of its application to various classes of vehicle and its significance to the fundamental understanding of human vehicular control behavior. This work, partially supported by the NASA, is described in Refs. 9 through 12.

A fairly extensive discussion of the advantages and potential applications of the balancing-reflex concept is given in Ref. 9. Of the items mentioned there, one of the most timely is the application to propulsion and control of the free-floating spaceman.

The difficulties encountered by a spaceman in attempting to do any significant amount of useful work outside his vehicle are by now well documented; they clearly stem from his inability to establish and maintain a required orientation of his body with respect to a "target" object without resorting to the use of clumsy restraining devices, dexterity preempting hand holds, and debilitating body contortions. Clearly, what the spaceman needs is a reasonably powerful and delicate means of controlling his body orientation that neither encumbers his hands nor requires him to fight his unyielding pressure suit. Adaptation of the natural, body-orienting responses of the feet and legs to the modulation of appropriately located thrusters appears to be a way to provide this means reliably, cheaply, and simply. The present document describes some preliminary work in this direction.

CHRONOLOGY OF THE DEVELOPMENT OF A SYSTEM

The development of a system for adapting natural, neuromuscular, body-orienting responses to the control of body-orienting thrusters for spacemen is, almost by definition, exploratory and experimental in nature. The particular problems and pitfalls likely to be encountered cannot be predicted and so the work must proceed in a stepwise manner, each step directed by the experience obtained from the preceding ones. The following discussion is a chronology of the steps that have led, in the present case, to a workable EVA control configuration.

Simulators

Many ways of simulating zero-g have been used or suggested, but of course all have drawbacks of one kind or another. Water immersion, for example, produces large viscous forces and is not completely free of gravity effects, cable suspension becomes involved with complicated pendulum dynamics, and so forth.

For the resources at hand, the most practical compromise with reality appeared to be a three-degree-of-freedom simulation based on frictionless motion in the horizontal plane. The particular combination of degrees of freedom obtainable in a plane (two translations and one rotation) is reasonably defensible for exploratory work in zero-g simulation. It does provide a logical sort of consistency, a representation of the complete job of "getting around" in space (albeit two-space rather than three).

Of the three possible configurations for planar motion of the human body, the one involving pitch rotation (see Fig. 1) appeared to be the most appropriate for initial exploration. Thus the simulator or "scooter," as it came to be called, took the form of an articulated bed, carried by two levapad (air-bearing) supported tripods, upon which a person reclines. Although designed primarily to accommodate a man lying on his side as shown in Fig. 2, the device can be adapted readily to the supine posi-

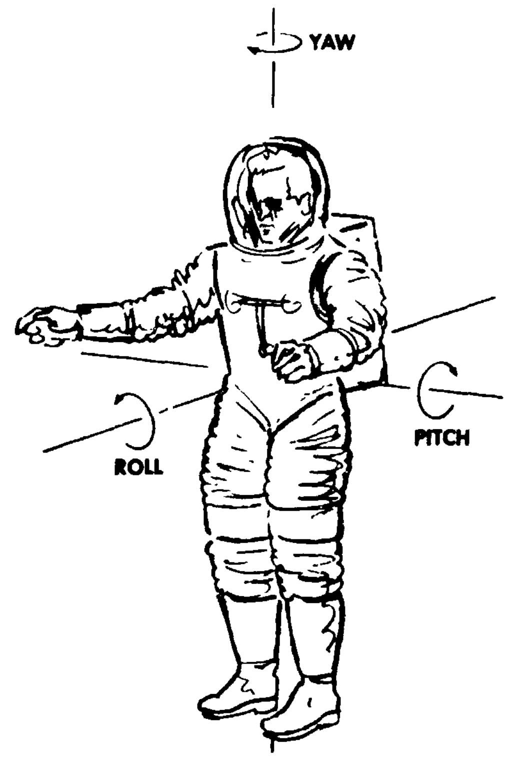


Fig. 1 The Rotational Axes

tion. The special floor on which the scooter glides is made of epoxy plastic poured over a concrete base, and is about 30 feet square, a more or less arbitrary compromise between desirability, availability, and expense.

Although the scooter could have been adapted to the standing position for examining yaw, it was not practical to do so. Therefore, a separate yaw simulator, a simple rotary device, was built for this purpose (see Fig. 3).

In all of the exploratory work carried out to date, the experimenters have served as the primary flyers and evaluators. Numerous

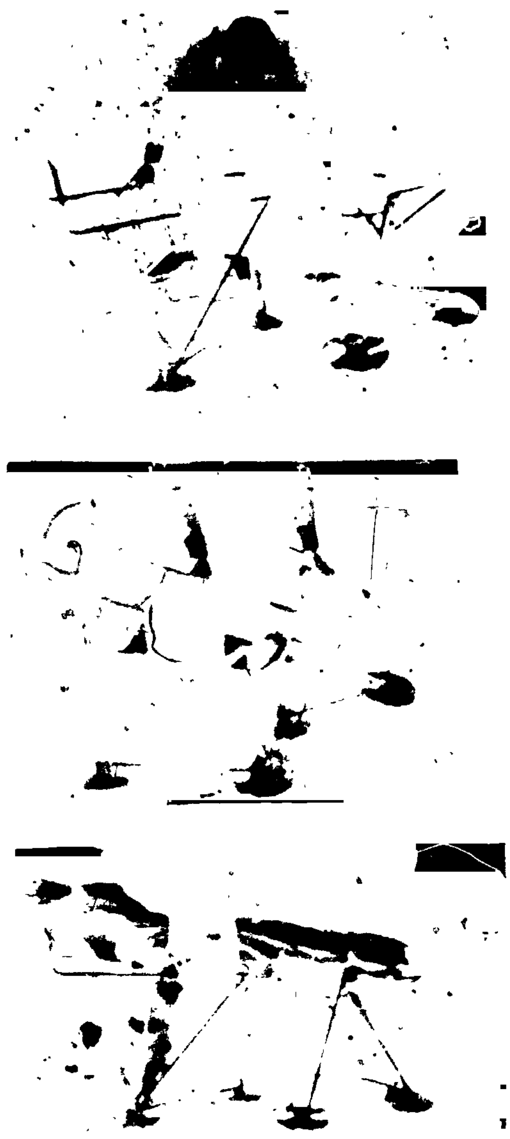


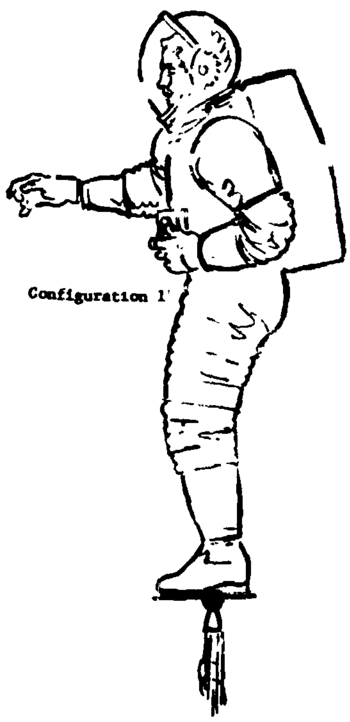
Fig. 2 The Basic Scooter

others, including experienced pilots, however, have flown the simulators in various control configurations, and their impressions coincide generally with those expressed in the following sections. No astronauts have as yet participated.

Fig. 3 Yaw Control Simulator

The Original Control Configuration

The one-g, balancing-reflex concept, in its most elemental form, makes use of a single, supporting thruster which, with the aid of gravity, gives the flyer control of five degrees of freedom. It is the very essence of elegant simplicity. Thus it is not at all surprising that extrapolation of the idea to zero-g applications should center on basically the same configuration. This was in fact the case for the initial effort at Grumman, and the idea still prevails in the NASA Jet-Shoes work.[7,8]



Unfortunately, the very first simulator trials demonstrated quite clearly that the simple configuration could not provide what the Grumman research philosophy had established as a design goal: natural (unconscious), precise control of the body in space. An immediate and clear symptom of the problem was a complete absence of any feeling of "balancing," in the automatic sense which is typical of one-g jet-platform flying. Consequently there was no delicacy of control. The reasons for this (obvious in retrospect) also became quite clear. First, the amount of thrust needed for fairly spirited maneuvers was very small (less than five pounds), hence the system gain, i.e., angular acceleration per degree of ankle deflection, was extremely low, orders of magnitude below the optimum for one-g balancing (as established by Ref. 9). Second, thrust was required only for brief periods, hence pitching responses did not inexorably follow ankle motions, as in the one-g jet platform, and there could be no sustained "feel" of the system.

Besides the basic balancing problem demonstrated by the brief series of experiments with the jet-platform configuration, a more subtle difficulty began to come to light. The original thinking had been that, in the absence of gravity (combining vectorially with thrust for forward motion;

"walking" mode), translations would be accomplished primarily in a "swimming" mode (head or feet first), with up-and-down thrust controlled by knee flexing. It began to be apparent, however, that people have a natural inhibition against traveling any distance head-first or feet-first; a flyer insists that he must be able to <u>look</u> in the direction of motion, and if he cannot, as when he is inside a space suit, he becomes not only apprehensive, but faulty in his judgment of direction and speed.

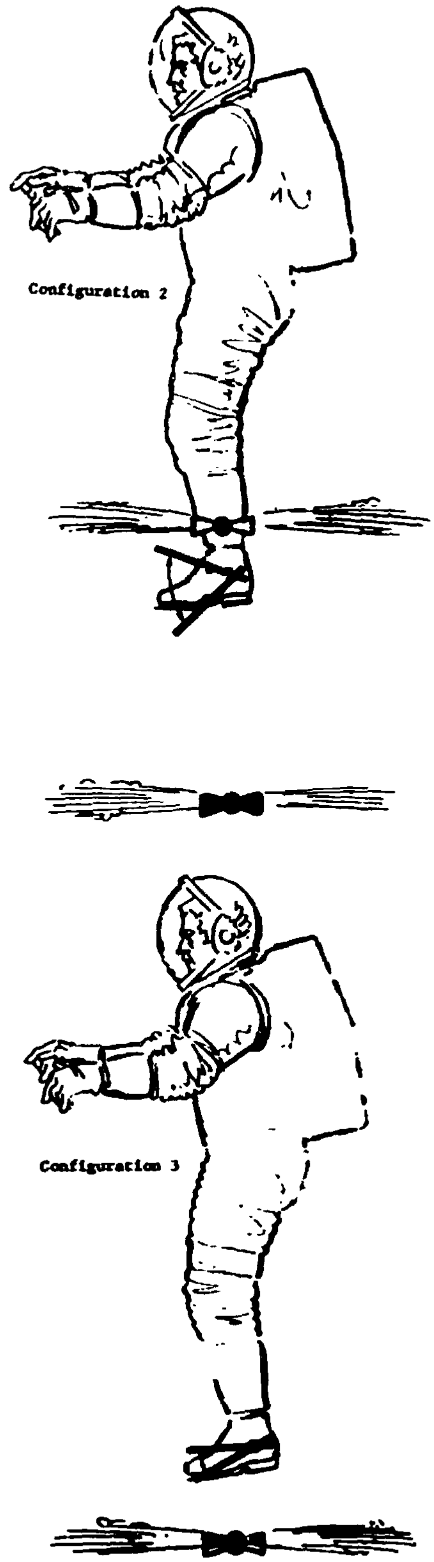
In light of the clear and inescapable conclusion regarding adherence to the Grumman objectives, some commentary on the apparent success of the Jet-Shoes concept[7,8] is in order. As far as can be determined, the NASA personnel have adopted a quite different, but equally valid, set of ground rules. They, too, appear to have uncovered the same basic problem early in their experimentation, but they have chosen to sacrifice the high degree of control finesse inherent in natural balancing in favor of the extreme simplicity of Jet-Shoes. Their objective has become simply to provide the spaceman with a cheap and reasonably effective way of getting from one place to another, not to give him precision control when he gets there. As far as is known, they have not concerned themselves with the swimming-mode visual problem.

Control Configurations Two and Three

Following such abject but eye-opening failure of the simple concept to behave in zero-g even vaguely according to objectives, a certain amount of backtracking seemed to be necessary. The thinking <u>had</u> been along the lines that the simple jet, somewhat elaborated, might serve the complete control and propulsion function, as it does in one-g. It now appeared, however, that control of the various degrees of freedom would have to be separated and, perforce, evaluated one at a time. Pitch control, which is the most closely associated with balancing, seemed to be the appropriate function to look at first, and the scooter was therefore reworked to provide for a pair of crosswise (fore-and-aft) thrusters, located near the feet, and controlled, roughly proportionally, by a valve actuated mechanically by ankle deflection. Photographs of the configuration are shown in Fig. 4.

The previous experiments had clearly brought out the need for higher system gain, but just how high

Fig. 4 Ankle Pivot and Thruster Arrangement for Configuration 2



it should be was moot. For one-g flight Ref. 9 had established an optimum gain in the vicinity of .1 g acceleration at the feet per degree of ankle deflection, but conceivably this value might not be in any way related to the requirement for zero-g flights. A simple side experiment using the research apparatus of Ref. 11, suitably modified (Fig. 5), indi-

Fig. 5 One-g Simulator as Modified for "Zero-g" Trials

cated that the .1 g per degree value was probably valid. It turned out, however, that achievement of this value on the zero-g simulator, without the introduction of inordinate amounts of friction and backlash, was almost impossible. Therefore a compromise value of about .01 g per degree was set up. Results were encouraging; a feeling of balancing, though weak, was now clearly evident. But it was also evident that the gain was still far from satisfactory, and that there was a maneuvering problem in which the unbalanced forces produced by the thrusters during moderate rotational maneuvers built up a disconcerting spurious translation.

The lessons learned from the second configuration led to trial of Configuration 3 in which the single force was replaced by a couple, and the system gain was quadrupled by increasing the thruster moment arm and altering the control-valve linkage. The results of these changes, measured in terms of prior experience, were spectacular; pitch attitude control became entirely natural and effortless, permitting angular displacements to be made with precision, and "tumble" recoveries to be executed smartly. Roll control, briefly investigated with the flyer lying on his back, looked equally good. Friction and dead zone in the linkage, however, had been increased by the gain-changing alterations, and the dramatic elimination of other faults now caused these to stand out very clearly, especially dead zone, which had never really been encountered before in any of the one-g balancing experiments of Refs. 9 and 11.

The Fourth Control Configuration

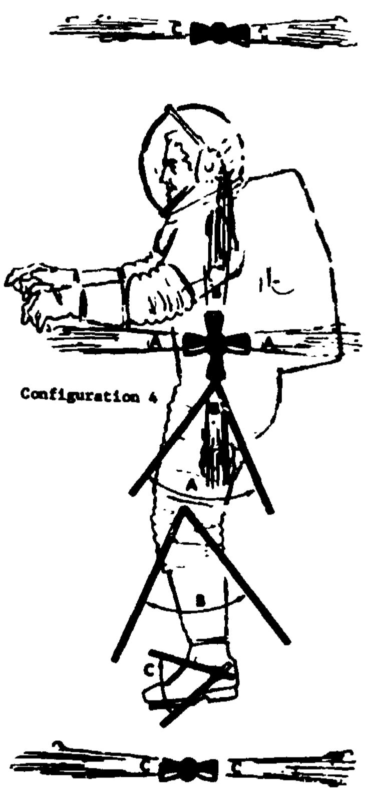
With the encouraging results achieved for pitch control alone, it seemed appropriate to turn attention to the two translational degrees of freedom: fore-and-aft and up-and-down.

There has been general agreement, dating back to the one-g jet platform work of Ref. 9, that "squatting" might be an appropriate mechanism for control of up-and-down thrust. Here, upward acceleration would be the natural and expected response to extension of the legs, and downward acceleration to retraction; the proper direction of response is clear and unambiguous. There is, however, a question about how the body deflection should be measured for transferral to a thruster control valve. The simplest arrangement seemed to be to pick up knee flexing at the appropriate joint in the simulator bed.

In an analogous fashion, waist-bending appeared to be an appropriate mechanism for the control of fore-and-aft thrust, but in this case the choice of direction of the response depends strongly on one's point of view. If one thinks in terms of leaning the upper body (buttocks fixed to the ground), then forward bending should produce forward motion. But if one adopts a "baby-walker" point of view in which the feet are fixed to the ground and the torso is propelled back and forth by the legs, then backward bending (backward thrust of the legs) should produce forward motion. The former arrangement seems to have a more elemental psychological appeal, and certain forms of human behavior can be pointed to in its support, e.g., the tendency of a highly involved observer of some action to "urge" an object toward a desired goal by leaning. The latter arrangement, on the other hand, is an exact analog of the clear-cut, vertical motion case, where the legs also propel the torso in the desired direction.

This philosophical controversy is perhaps resolved by considering that even in the baby-walker case the motion that <u>initiates</u> an action is a lean in the desired direction. It is this unconscious, precursor type of muscular response that would be expected to provide the most natural mechanism for control of the body. For Configuration 4, then, the body-lean philosophy was adopted. Waist flexure, measured between the thigh and torso, was picked up for transferral to the air valve mechanism by a lever extending between the upper and lower halves of the simulator bed. A system gain of about $1\frac{1}{2}$ pounds of thrust (or 1/300 g) per degree of body deflection was selected for both translational control modes on the basis of practical valve-linkage considerations.

Simultaneous operation of all three control modes



became fairly successful after a little practice, but a single, glaring deficiency interfered with natural control. The manner of picking off waist bending required that thigh motion be reserved exclusively for fore-and-aft control, thereby precluding the use of true squatting for up-and-down control. Unfortunately, pure knee flexing turned out to be a highly unnatural substitute for squatting; unless the flyer put his mind to it, he invariably squatted for up-and-down commands, causing a most disconcerting, concomitant, fore-and-aft response. An occasional tendency to become confused in the use of the translational controls can probably be attributed to this cross-coupling effect, and it was interesting to note that dead zone (detrimental in the prior experiments) now seemed to be helpful for reorientation after a period of momentary confusion, raising the question of whether some sort of tangible neutrals might be desirable.

It was quite clear that pitch control remained good or perhaps even improved a bit when the flyer became preoccupied with his translational controls, which plainly demonstrated the value of "natural" neuromuscular mechanisms in this application.

Although very little body motion could be seen, the translational control gains were judged to be far too low, even lower than the rotational gain, and there was a distinct feeling of disharmony between the modes.

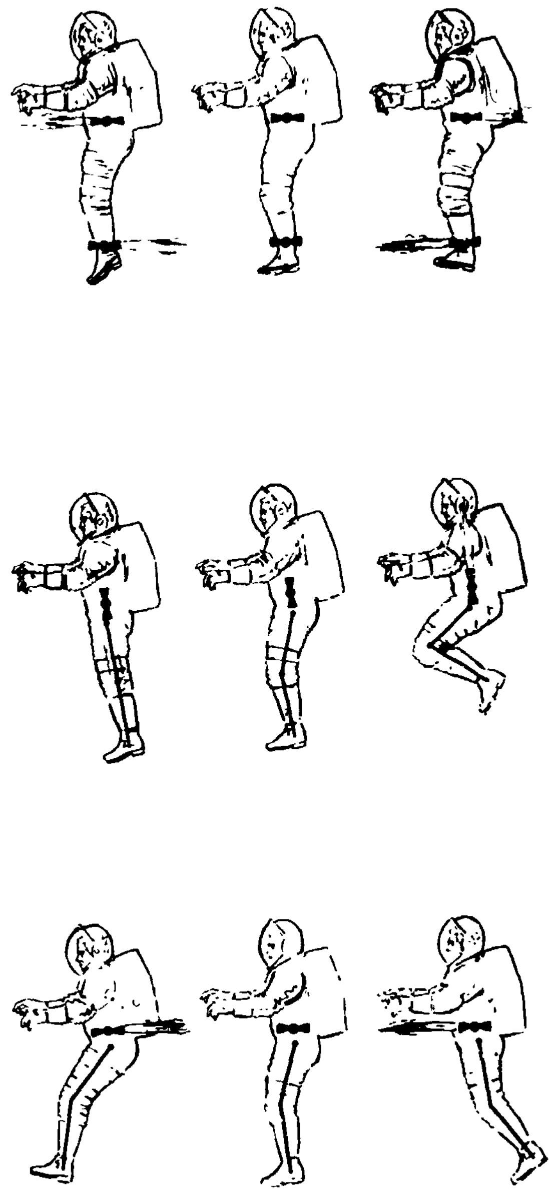
Configuration Five:
A Success

Configuration 5 might be considered a kind of culmination, because it represented for the first time, a truly workable system for spaceman maneuvering. On the simulator, the control valve linkage geometries had been modified to eliminate cross coupling between squatting and waist bending, and provision had been made for centering springs and detents on all three controls. Mechanical considerations did not permit any appreciable increase in the system gains over those used in Configuration 4, so the same questions concerning gain and gain harmony remained, but it turned out that the elimination of translational control cross coupling provided such a dramatic increment in naturalness that the gain problem lost much of its immediacy; the system, even with its low, inharmonious gains, became workable.

The scooter as shown in Fig. 6 was fairly extensively

Fig. 6 The Scooter Arranged for Proportional Control

flown in simulated space task maneuvers, and a number of impressions about its flyability under various conditions emerged:



Configuration 5

1) All three modes of control (ankle deflection, squatting, and waist bending) can be handled quite nicely, but with varying degrees of apparent naturalness. The relatively low gains of the translational modes almost certainly contribute to their lower quality, but there is a powerful experimental artifact that must raise serious doubt about any hasty judgment of control naturalness. This has to do with the sound of the control jets, which is loud, disconcerting, and often downright confusing. Because maneuvering is typically slow and deliberate, the motion cues (visual and proprioceptive) by which a flyer should operate, are weak and easily swamped by strong aural cues. Unfortunately there is a very strong urge, especially in the novice, to try to use the jet noise cues for flying. This can, in fact, be done for very simple maneuvers, but the sounds become hopelessly confusing in complex situations, and the flyer who has begun to rely on them often finds himself in a panic, unable (momentarily) to figure out what to do. It requires a strong effort of will for the novice to ignore the sound and attend only to the proper signals. Once he has learned to do this, however, his flying becomes much more instinctive.

2) Centering springs on the controls are, in general, beneficial; they make it easier for the flyer, especially the novice, to find neutral. Detents, in the form of preloads on the springs, are also useful. A certain amount of care in the selection of spring rates and detent loads must be used, however, lest the flyer's natural coordination of hip and knee flexing in squatting be upset and, more critically, lest the subjective values of system gain be reduced.

It turns out, in this respect, that a flyer's interpretation of gain seems to be based on some over-all feeling of "effort" required to obtain a given response. Thus gain ought really to be expressed in terms of "acceleration per unit of effort," but it is not clear just how a flyer senses accelerations or how he defines "effort." Apparently, "effort" represents some combination of force and displacement, but just what combination is quite unknown. Its mathematical describing function undoubtedly is one in which the relative contributions of force and displacement to the subjective impression of gain change drastically with the spring rate, ranging from all-displacement at zero spring rate to all-force at infinite spring rate. A determination of this function for the various control modes could become the objective of some interesting additional experimentation.

Comparison of the flying characteristics of the scooter with and without centering springs is of some interest. It turns out that the novice is much more comfortable, and maneuvers more skillfully, with springs, but the experienced flyer apparently does equally well either way, and, in fact, if there is appreciable dead zone in the control system, may actually prefer no springs. Probably, as previously discussed, this is because the expert is able to ignore the sound of small residual jet flows resulting from his imprecision in neutralizing the valves. Such flows, though of negligible effect on maneuvering, are quite audible, hence difficult for the novice to ignore, and likely to cause him to go through a great deal of unnecessary struggle to eliminate them. Thus he prefers the springs, which permit him to shut off his jets completely simply by relaxing. The expert, on the other hand, tends to be annoyed by the springs because they demand more effort, particularly if there is a large dead zone to be pushed through before the jets come on. This line of thought now returns closely to the previous discussion about the meaning of "effort" in the operation of the control system. The expert's objection to the effort required to manipulate the springs appears to be based not so much on muscular "laziness" — forces (a pound or two) are, after all, far lower than people handle routinely without complaint — as on some sort of "control quickness" factor; in other words, "effort" seems to refer more to subtle difficulties with the response characteristics of the system (including the neuromuscular part). If this is in fact the case, the general study of gain previously suggested becomes all the more intriguing, and possibly quite important to the design of optimum systems.

3) Control power levels required for useful maneuvering are remarkably low. Maximum thrusts and torques typically used on the scooter (although more is available) are about 5 pounds and 15 foot-pounds, respectively which translate to about 2 pounds and 4 foot-pounds in the real spaceflight situation, where the thrusters do not have to move the considerable mass of the scooter. Such low values are certainly significant to the design of a practical system.

On-Off Control

There are two, potential, major advantages to the use of on-off operation in the present application: thruster control may be simpler, and fuel specific impulse may be greater. Thus the flying qualities of on-off control systems are of some importance to the overall picture.

The simulator was modified for on-off control experimentation by the addition of a solenoid-operated air valve behind each thruster nozzle, and short throw, low force, snap switches at each body motion pickoff point. Nozzles of various diameters were provided for each thruster to permit examination of the effect of thrust level. Views of the scooter as it was thus set up are shown in Fig. 7.

Fig. 7 The Scooter in Its Final Configuration

Initial trials of the on-off system used thrust levels of $10\frac{1}{2}$ pounds for the translational modes, and a torque level of 15 foot-pounds for the pitch mode. Centering springs and detents as in the proportional control experiments were used, and the "off" zones of the controls were made fairly large. The flyability of this arrangement turned out to be much better than expected, but several deficiencies stood out quite clearly. For one, the "off" zones were far too large, giving a subjective impression resembling unduly low gain in the proportional system. Secondly, there was an annoyingly large hysteresis in the switching arrangement, which created the effect of requiring a positive effort to shut off a thruster once it had been turned on. Because of the flyer's neuromuscular time lag this put a noticeable lower limit on the minimum duration of a thrust burst (perhaps $\frac{1}{4}$ second), resulting in constant overcontrolling and "limit-cycle" type of behavior during attempts at delicate maneuvering. And thirdly, $10\frac{1}{2}$ pounds of thrust was much too high, clearly aggravating the hysteresis problem and essentially precluding precision control. This thrust level also caused a peculiar dynamic instability, characterized by a high frequency (2 cps), limit cycle type of oscillation in the waist-bending mode whenever the flyer arched back against the spring just to the edge of switch closure. This phenomenon was not particularly debilitating because it occurred only rarely and could be stopped by simply relaxing, but it does illustrate a potential problem with on-off systems that could very well dictate such factors as thruster location, centering spring sizes, and "off" zone minima.

Following these experiments, the "guilty" parameters were readjusted to the levels shown in Table I. Flight with this configuration turned out to be remarkably good. Delicate maneuvers could be made with precision, and the flying, though done in a style noticeably different from that of the proportional control system, was quite natural.

As in the proportional control experimentation, configurations with and without centering springs behaved quite differently. As before, springs benefitted the novice more than the expert and called for reduction of the dead zones (in this case the "off" zones). But, unlike the proportional case, springs seemed to be preferred by both expert and novice. A strong tendency toward limit-cycle type of operation without springs is the probable explanation.

Although the basic control parameters (thrust, "off" zone size, and control-centering strength) have admittedly not been optimized, on-off control has nevertheless been shown to be practical.

Table I

NOMINAL PHYSICAL CHARACTERISTICS

	Ankle	Knee	Waist
Off Zone	± 1½ deg	± 1½ deg	± 1 deg
Friction	Nil	Nil	Nil
Turn-On Torque	± 16 in.-lb	± 45 in.-lb	± 40 in.-lb
On-Off Differential	4 in.-lb	12 in.-lb	18 in.-lb
Detent Torque	± 8 in.-lb	± 30 in.-lb	Nil
Thruster Effort	± 15 ft-lb	± 2½ lb	± 2½ lb
Mass Scooter & Man	15 Slugs		
Mom. of Inertia Scooter & Man	42 Slug-ft^2		

Several subjective impressions regarding the relative behavior of on-off and proportional control systems have evolved:

1) The character of the flying of the two systems is clearly different. The proportional system seems to promote simultaneous operation of the various controls with a consequent feeling of continuity and smoothness during complex maneuvers. On-off controlling, on the other hand, seems to be done primarily sequentially, so that maneuvering becomes a series of discreet operations. (Of course, the actual flight path is smooth and essentially as precise as that of the proportional system.) The feeling of smooth continuity in proportional flying is particularly striking and pleasant immediately after transition from an extended period of practice in on-off control.

This may, however, result as much from the character of the jet sounds — which change from a cacaphony of brain stabbing blasts to a modulated hissing — as from actual motion effects.

2) Fast maneuvering is done more confidently with the proportional system. This undoubtedly stems from the availability of larger thrusts that can be used as "safety margins" to compensate for any misjudgments in speed. With the on-off control only one level of "braking" is available and the flyer must therefore be more skillful in his selection of braking points, particularly if he is trying to operate as smoothly as possible. Of course if the maximum proportional thrust were not larger than the on-off value, this conclusion would be invalid, and in fact the proportional flyer might have <u>more</u> trouble with fast maneuvers if "running out of control power" comes as a surprise.

The whole question of the desirability of fast maneuvering is complicated by the fact that velocity is equivalent to fuel increment, and it is therefore desirable from the economy standpoint to keep all motion as slow as possible. On the other hand, factors such as the limits of human patience or the need to get a job done quickly may overbear economy at some point. Thus the parameters that govern fast maneuverability ought eventually to be examined in detail. It is clear, here, that control power is a strong parameter up to a point, but that human factors such as ability to judge and predict, and neuromuscular lags must enter the picture sooner or later.

On balance, proportional control appears to be generally better than on-off control, but not so much better that some engineering consideration such as simplicity of thruster actuation might not specify the use of an on-off system.

Yaw Control and the Current Design Thinking

For some time the Grumman idea has been that body-twist is the appropriate natural motion for controlling yaw. It could not be proved, however, until the recent completion of the yaw control simulator (Fig. 3). To use this device the pilot stands on the platform and is strapped to the "T" bar. Body-twist, which commands motor output torque, is measured as the angular displacement between the platform and the bar, and the motor drives either the pilot's feet (via the platform), or his body (via the metal bar).

Two important results were dramatically demonstrated during preliminary experiments with this simulator. First, yaw control is just as natural as pitch and roll control. In fact, the pilots who have "flown" the simulator have not required any learning. The other important result is that driving the feet provides the pilot with more natural force feedback than driving the body, and thereby results in a much more instinctive and precise control. This result led to a brief reevaluation of pitch control on the scooter, with pitching torques applied to the feet. Here again, applying torques to the feet was found to be superior. The results of these preliminary experiments indicate that a free floating spaceman's control mechanism should apply forces and torques directly to the feet and legs.

This philosophy has been applied to the preliminary design of a prototype flight system. An artist's conception of the system as it is currently envisioned is shown in Fig. 8. It provides the five separate modes of control that have been discussed (pitch, roll, yaw, and fore-and-aft and up-and-down translations). The rationale for excluding lateral translation is, basically, that

Fig. 8 Design for EVA Control System

lateral translation would be needed only for close-in work and in small amounts, and therefore could be adequately effected by use of a "backing and filling" technique involving yaw and fore-and-aft control. This idea is admittedly a speculation that would have to be demonstrated, but in any case lateral translation <u>could</u> be added to the design at a certain cost in complexity.

An interesting feature of the design shown is that all thruster valving functions are carried out in the compact mechanism between the feet, and that, essentially, the feet become the agents for all control. This arrangement, besides being appealingly simple, eliminates some of the control harmony problems that ensue from picking off body deflections higher up.

QUESTIONS AND SPECULATIONS

The experimentation carried out to date has proved a basic concept, but there remains a number of possibly crucial, unanswered questions. Some speculative discussion of these follows.

Are More Than Three Degrees of Control Freedom Practical?

This is the crucial question, and it is not likely to be answered with any finality until a complete system can be tried, either in flight or in a complete-motion simulator. There are some encouraging signs, however. For instance, there is the clearly demonstrated naturalness of pitch, roll and yaw control alone in one-g and "zero-g", and there is Zimmerman's demonstration that pitch and roll can be combined without upsetting their instinctive operation. These lead easily to the speculation that control of all rotations simultaneously can be just as natural and instinctive as control of one alone. If this can indeed be shown, there is room for a good deal of optimism that control of at least five degrees of freedom will be little, if any, harder than the presently demonstrated three. Thus it seems that the crucial experiment for the near future must demonstrate the simultaneous use of the three rotational control modes.

Are Six Degrees of Control Freedom Necessary?

This question can be asked in connection with ideas not only of human capacity, but of mechanical complexity. Under the assumption that complete control of rotation is vital to the performance of space tasks and is relatively easy to accomplish, the question becomes, "Are three degrees of translational control freedom necessary?" At one point during the experimentation described in this report, the question was phrased, "Could, for instance, control of vertical translation be successfully eliminated?" The answer turned out (not too unexpectedly) to be an unqualified "No;" the mechanical process of "backing and filling," or "tacking," (using pitch rotation), to effect a change in vertical position proved to be unacceptably clumsy. But it might be speculated that the same process using yaw rotation to effect a lateral translation might not be at all clumsy, because yawing (as in body twisting) is quick and easy, and requires little space. This philosophy has, in fact, dominated the preliminary design thinking to date. Definite proof of the concept must be obtained, however, before any serious, detailed designing of a prototype system can proceed.

Does a Space Suit Interfere?

One of the principal artifacts of space suit technology today is stiffness. Therefore, any activity of a spaceman that requires extensive flexing of his body must be looked at askance, and it is only natural that doubt should arise in this respect concerning a control system that requires flexing of the hips, knees, and ankles. The present experimentation has shown, however, that the gains preferred in this system are so high that there is very little visible flexing of the body, even during spirited maneuvering. The speculation here, therefore, is that the foot and leg control concept, far from being incompatible with space suit operation, is in fact particularly appropriate to it.

What About System Safety?

Two kinds of unwelcome system failures are conceivable: one in which the system dies, leaving the spaceman stranded, and one in which the system goes berserk. Of course, the latter would usually lead to the former.

For the stranding situation, one can think in terms of a simple, emergency backup system (such as the present "space gun"), or in terms of retrieval of the stranded spaceman by his buddy in the mother vehicle. A certain amount of training in the use of a space gun could be required, however, since the spaceman might well be left with a rotation to be gotten rid of before he could attempt to return to his vehicle.

For the berserk-system case one thinks primarily of automatic and manual system cutoffs. A rotation cutoff would most likely have to be automatic, because very nasty spin rates can be built up in fairly short times. It should be possible to devise some sort of rotation sensing mechanism, perhaps based on centrifugal or Coriolis effects, which would respond to the emergency but not to ordinary operations. Translation cutoff could probably be done manually.

SUMMARY OF MAJOR CONCLUSIONS

1. The basic concept of precise, hands-free control of spaceman maneuvering by exploitation of instinctive muscular responses of the feet and legs is practical.

2. Accurate, natural control of gravity-free motion

in a plane has been demonstrated.

3. A control system should include separate and uncoupled control of the individual degrees of freedom, but control of all six may not be necessary.

4. Ankle deflection for pitch control, differential foot lifting for roll control hip twisting for yaw control, squatting for vertical control, and waist bending for fore-and-aft control are instinctive responses.

5. Control mode gains (acceleration per unit of body deflection) should be high, resulting in little or no body flexure noticeable to an observer.

6. Both proportional and on-off control are practical. Proportional control is slightly preferable to the flyer.

7. The most natural, instinctive, and precise control is achieved when control forces and torques are applied as feedback to the appropriate "controllers" (e.g., pitching torque applied to feet). If control forces are not applied as feedback, mild centering devices on the control pickoffs are generally desirable.

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SESSION VI

PROTECTIVE SYSTEMS

Session Chairman: E. L. Hayes
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NASA PROGRAMS FOR ADVANCED SPACE SUIT DEVELOPMENT

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SUMMARY: A space suit development philosophy has been formulated through which a unified, coordinated NASA suit development program will yield hardware design concepts capable of supporting any desired mission involving orbital or lunar surface activities in either an intravehicular (IVA) or extravehicular (EVA) mode.

These missions will require at least three different, distinct space suit configurations, including: (a) an emergency, intravehicular suit for use inside the orbiting spacecraft; (b) an EVA suit optimized for orbital use; and (c) an EVA suit optimized to meet the stringent demands of lunar surface exploration.

INTRODUCTION

During the latter phases of the Gemini missions, adequate in-flight operational data had been accumulated to permit a thorough analysis of the man/suit system, its strong points and its weak points.

This data, along with experience gained in ground-based simulations, and mission analysis, formed the basis for establishment of the present NASA space suit development philosophy.

Up to and including the Apollo program, primary consideration has been given to providing a single space suit which is configured to "accommodate" the crewman for both the intravehicular and extravehicular portions of the mission. From experience gained during the Mercury and Gemini flights and from a review of space suit state-of-the-art technology to properly accommodate the dual requirements, it has been determined that future development programs should consider the use of special-purpose suits, tailored specifically to the intended mission requirement; i.e., intravehicular, orbital EVA, and/or surface exploration.

Generally, comfort and low-bulk characteristics desirable for IVA use, are divergent requirements from those associated low torque high range characteristics for optimized mobility. For the Gemini mission, the prime consideration was given to providing a suit capable of long-term comfortable, unpressurized wear.

This concept required a suit (Figure 1) with minimum number of "hard" components which normally result in body pressure points and discomfort. Hard components, which are generally associated with optimized mobility and minimum torque frequently result in a comfort compromise for long-duration, unpressurized wear.

A space suit development philosophy has been formulated through which a unified, coordinated development program will yield hardware designs capable of supporting any desired mission involving orbital or lunar surface activities in either an intravehicular or extravehicular mode. These missions will require at least three different, general space suit configurations which include:

a. Intravehicular space suits

b. Orbital EVA space suits

c. Surface exploration space suits

Consistent with this space suit development philosophy, the NASA has established and is presently pursuing the development of a suit technology base in the three mission categories. Development attention is also being devoted to ancillary equipment requirements and needs, such as glove optimization, body cooling techniques, closures, etc.

INTRAVEHICULAR (IV) SPACE SUIT

Space suit systems for use specifically inside a spacecraft may be designed to a significantly different group of development objectives than those used for EVA suits and combination EVA/IV suits. The IV space suit is intended to function primarily as an item of survival equipment for use during inflight emergencies. For this use the system must provide adequate unpressurized comfort to permit a crewman at least 8 hours continuous wear, while on an alert or standby status and while performing such critical maneuvers as lift-off, rendezvous, docking, and reentry.

General design objectives for IV suit development include the following:

a. System should approach shirt-sleeve mobility and comfort in unpressurized mode.

b. Must be acceptable for 8 hours continuous wear while on an alert or standby status.

c. System must provide adequate mobility to permit proper control and operation of a depressurized spacecraft under emergency conditions.

d. Suit must provide a habitable environment while pressurized and operating at pressures up to 5.0 psig.

e. Suit design shall permit

Figure 1.

intermittent donning and doffing for completion of a mission of one year duration.

To meet the prime design emphasis being placed on suit reliability, comfort for long-term wear, low weight, low bulk and quick donning characteristics, two intravehicular concepts are presently being pursued.

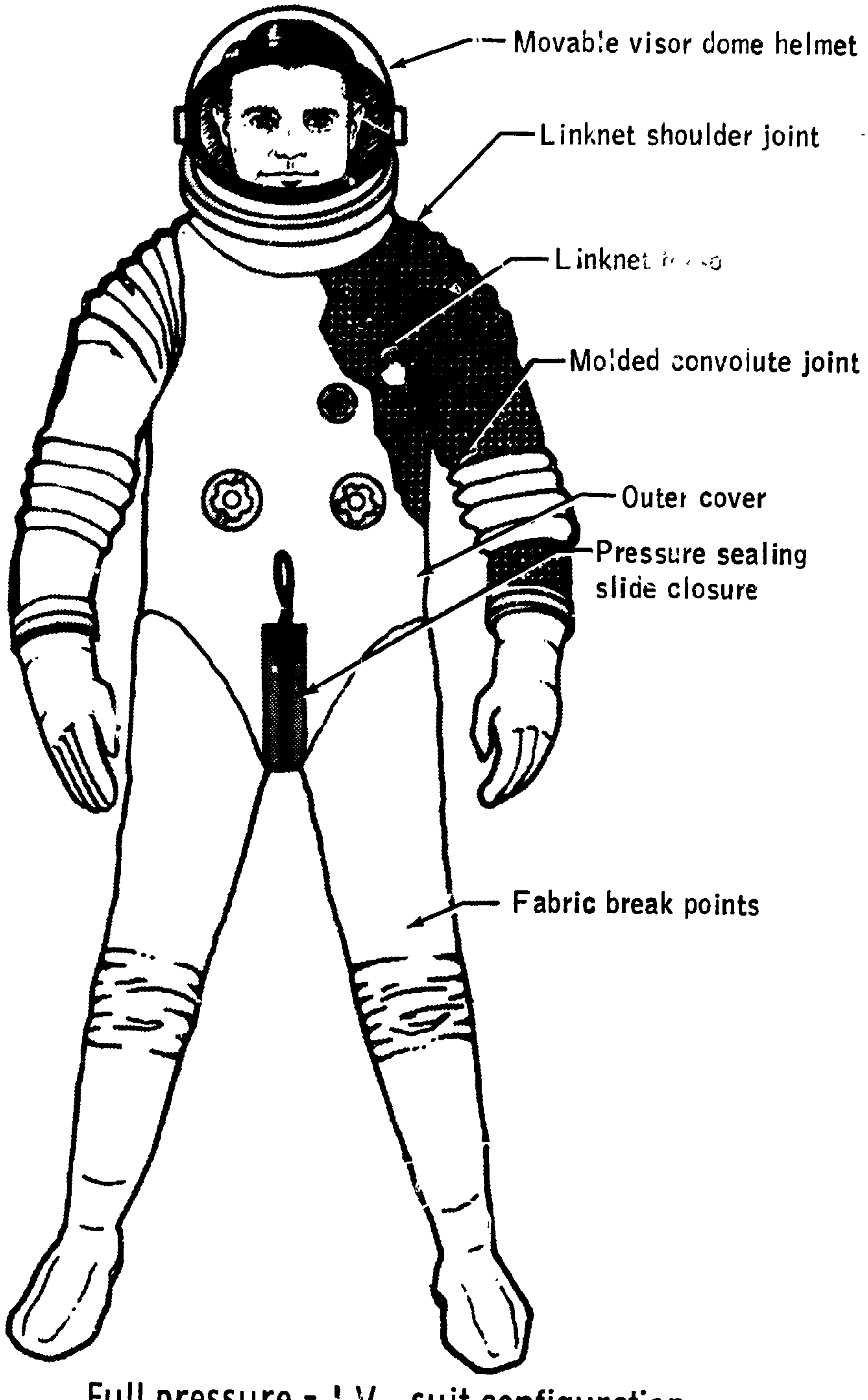
Full Pressure Suit System:

As an extension of a concept developed for the Gemini VII manned mission, developments are in process for a light weight, low bulk, full pressure space suit system. Using state-of-the-art design and fabrication concepts, an IV space suit (Figure 2) is being configured which makes maximum use of fabric or soft suit design techniques and eliminates, where feasible, hardware components. The torso from the groin line up to the shoulder area consists of a link net restraint layer over a neoprene coated bladder layer. The l- and segments of the arms are constructed using an outer covering of high strength, high temperature, nylon restraint fabric over the coated bladder layer. Elbow and knee mobility is obtained through the use of convolutes and fabric break points. Shoulder mobility is obtained through the use of the link net restraint/mobility layer.

Mechanical Pressurization Suit System:

In an effort to advance the state-of-the-art in IV space suit design, considerable attention is presently being given to a mechanical pressurization space suit system. The concept (Figure 3) is one in which the required pressure loading is applied to the surface of the skin through the use of a combination mechanical-pneumatic system. The force application is achieved through the use of inflatable tubes located against the skin and by an outer, porous restraint layer. Under normal operating conditions within a pressurized cabin, the tubes are deflated and retracted away from the body to provide a very comfortable loose fitting constant wear garment. During depressurized operation of the spacecraft, the tubes are inflated by gas supplied from the spacecraft ECS. The pressurized tubes fill the void between the restraint garment and the skin and apply a mechanical force to the surface of the skin. The skin at this time se[illegible] as the gas barrier to contain body fluids.

Breathing gas [illegible] supplied to the crewman through the use of a full pressure helmet, sealed from the [illegible] by a neck seal. In order to provide a balanced system of forces between the lungs and the mechanical external forces, the breathing air to the helmet and the pressurization gas for the inflation tubes are referenced to a common pressure source.



Full pressure - I.V. suit configuration.

Figure 2.

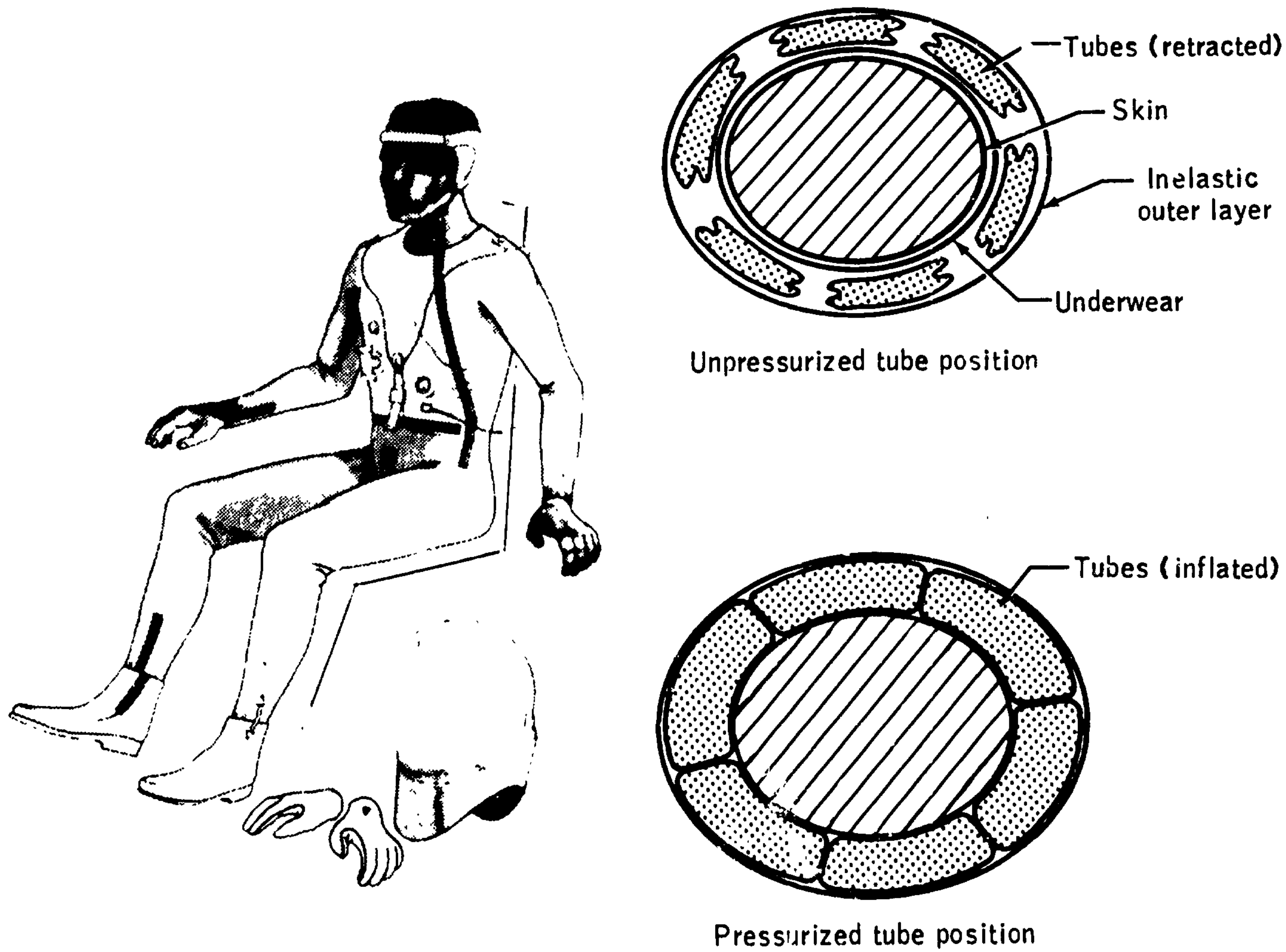


Figure 3. - Mechanical pressurization suit (unpressurized).

Although the concept is not entirely new, there are still numerous unknowns relative to long-duration use of the system. In particular, the hazard of long-duration exposure of the body surface to the vacuum of space is not fully understood as it relates to the physiology of the man.

ORBITAL EVA SPACE SUIT

To date, all extravehicular activities have been performed under near earth orbital conditions. Much has been observed relative to the performance capabilities and limitations of suits used for these missions. A review of potential flight plans for earth orbital EVA operation indicates the need to provide the capability for supporting a number of different types of EVA missions from those of the past. Some typical types of operations which may be required include:

a. Fabrication and erection of space stations

b. Vehicle inflight maintenance

c. Crewman rescue

d. Retrieval of experimental test panels

e. Inspection, capture, and repair of orbiting satellites

f. Emergency vehicle transfer

It is observed from the types of potential missions for an orbital EVA space suit, that a suit with a very refined mobility system will be a firm requirement. Complicating the achievement of this goal will be a requirement for minimum stowage volume consistent with the use of spacecraft with very limited pressurized stowage space. In order to achieve minimum stowage volume, it is felt that "soft" construction techniques will be necessary. Soft systems will permit the suit to be folded back into itself in a flat pattern to facilitate stowage.

Several candidate systems are presently under development through sponsorship of various Government agencies. Advanced suits under development by NASA are full pressure fabric systems utilizing constant volume mobility joint design techniques. The system shown in Figure 4 is representative of a general configuration believed to be necessary to meet the mission requirements. Specific features include:

a. Constant volume convolutes located in knees, ankles, elbows, and between bearings in shoulder and hip joints

b. Constant volume "stove pipe" multiple bearing shoulder and hip mobility joints

c. A two-axis waist joint in assembly with a single plane rigid waist disconnect.

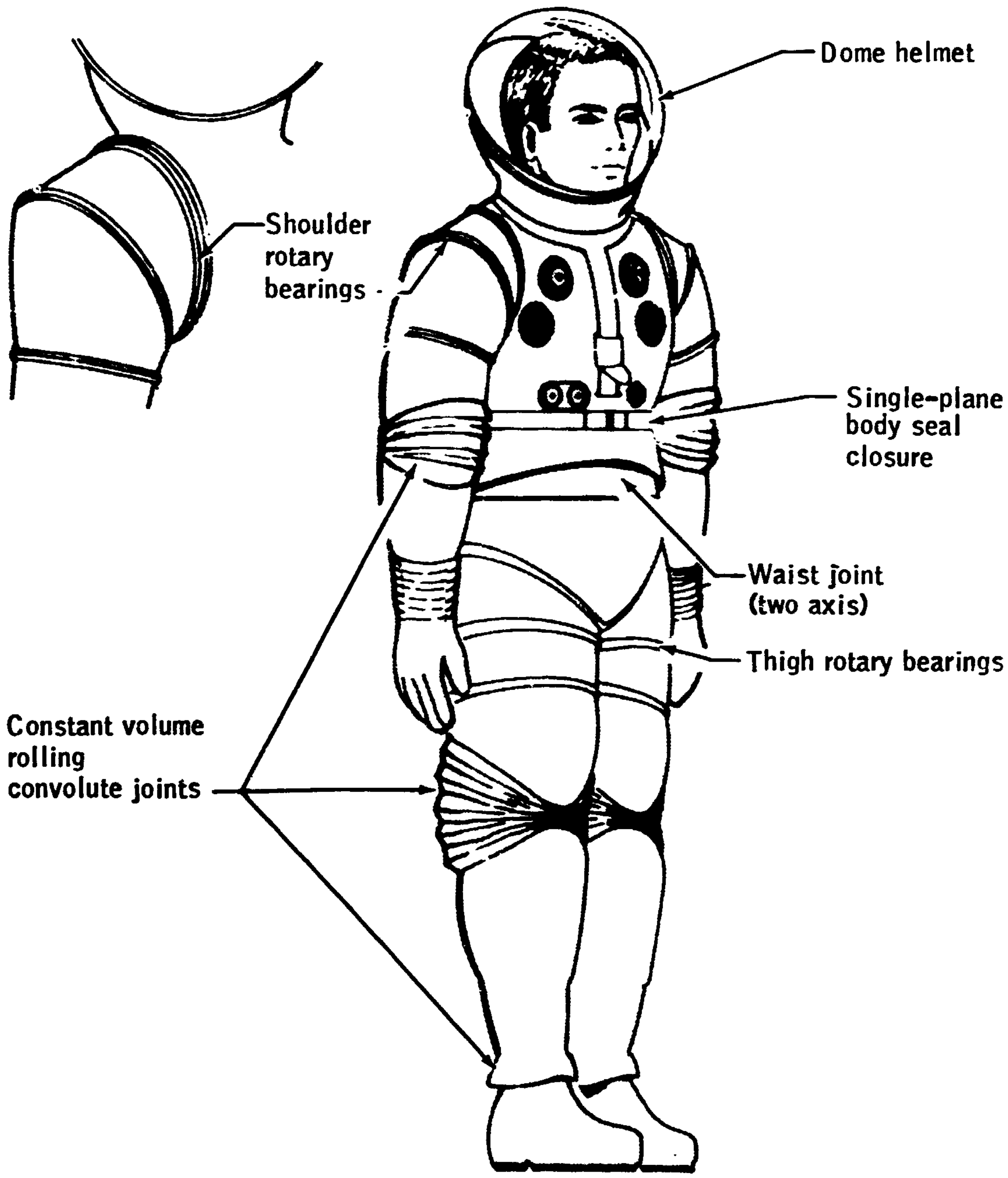


Figure 4. - Eva constant volume-space suit.

Developments are also being pursued to provide a single-layer restraint bladder system. Recent technology advancements in the use of metal fabrics and metallic yarns provide optimistic hopes for the system. The use of elastomer impregnated metal fabrics and yarns may permit the use of entirely new techniques for attaching hardware items to the gas bladder. Early developments indicate the feasibility for welding hardware to the pressure vessel, providing reduced weight and improved reliability.

SURFACE EXPLORATION SPACE SUIT

To accommodate long-duration lunar exploration EVA missions, a suit providing the maximum capabilities in mobility, reliability, impact resistance, and service life must be provided. In that much of the exploration equipment will possibly be delivered to the lunar surface, aboard unmanned equipment transport vehicles, increased weight and storage volume can be tolerated.

The state-of-the-art in suit design presently dictates the use of a hard structure suit assembly for long-duration missions. The capability to produce an effective, reliable mobility system for all segments of the body in combination with a high-impact resistance pressure vessel makes a hard structure space suit attractive for multiple, excursion, long duration exploration missions on the rough, abrasive lunar surface. Exceptionally long service life is a demonstrated characteristic of hard structure space suit designs.

Design goals for the surface exploration space suit encompass such general characteristics as:

a. Multiple don/doff capability to accommodate large numbers of excursions.

b. Suit must be don/doffed by wearing crewman, unassisted.

c. Abrasion resistant pressure vessel which will not degrade when in direct contact with the abrasive lunar surface during a fall or during kneeling operations.

d. Body mobility system must approach that for nude range and have less than 1.0 foot-pound torque for each joint with negligible spring back forces.

e. Items of high wear must be repairable during the mission.

f. Stowage volume should be minimized.

g. Suit should be capable of operating at pressures from 3.5 to 7.0 psig.

To achieve these design goals, two series of hard structure space suits are currently under NASA sponsored development.

The AX series hard structure space suit is a detailed study in multiple bearing joint technology with primary emphasis on a requirement that the total suit system be produced without the use of fabrics. The mobility system consists entirely of rotating pseudoconic (stove pipe joints) and metal bellows.

Considerable study has been made relative to stove pipe joints and techniques for optimizing the location and orientation of the various bearing components and associated hardware items. Following fabrication and evaluation of the initial suit AX-1, work has begun on design of the second generation system.

The RX series hard structure space suit has reached the final phases of research and development activities and is now considered to be flight qualifiable. Should the system be programmed for use in a manned mission, there are a few desirable modifications which would be made. These needs have been identified and evaluated during the suit system design verification testing. The latest model RX-5 (Figure 5) utilizes several combinations of constant volume joint mobility technologies.

Single axis rolling convolutes are utilized in the knee and elbow joints. Two axis joints using rolling convolute techniques are provided for the waist and ankle joints. Multidirectional joints for the shoulder use rolling convolute techniques and the hip joints make use of the stove pipe rotary bearing design concepts.

The suit as designed appears to offer one of the most durable and effective mobility systems presently available for long duration lunar surface exploration.

Figure 5.

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ADVANCED PORTABLE LIFE SUPPORT CONCEPTS

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INTRODUCTION

Manned space programs for the next half century will likely be aimed at such objectives as:

a. Development and establishment of orbiting research and applications laboratories.

b. Continued exploration and exploitation of the moon following the initial Apollo landing.

c. Manned exploration of one of the near planets, probably Mars.

The complexity of such future manned space missions will make ever-increasing demands upon the crewman and his equipment. In particular, the field of extravehicular activity (EVA) will consist of many diverse missions, each imposing its own unique set of requirements.

EVA equipment, as presently defined, consists of the following systems:

1. Life Support System

2. Suit System

3. Maneuvering System

This paper addresses itself primarily to the performance and operational characteristics of EVA life support systems for future missions. As an indication of present status, the Apollo Extravehicular Mobility Unit (EMU) is described and its intended applications discussed.

Candidate concepts for "next generation" EVA life support systems are then presented and trade-off studies conducted. Finally, conclusions are derived relating to projected state-of-the-art advancements in future EVA life support systems.

PRESENT STATUS

The Gemini Program, through the accumulation of 12 hours and 25 minutes of EVA, established the basic feasibility of extravehicular activity. While most of the Gemini EVA operations were successful, limitations such as the inability to perform EVA tasks without the proper body restraints, the mobility restrictions imposed by the design of the space suit, and the limited cooling capacity of life support systems using gaseous cooling, were identified.

The knowledge of EVA acquired through the Gemini program has been incorporated into the operational procedures and equipment design of the Apollo program.

Figure 1 is a schematic representation of the Apollo Extravehicular Mobility Unit (EMU) which is a functionally integrated system designed to allow the Apollo crewmen to perform such extravehicular activ-

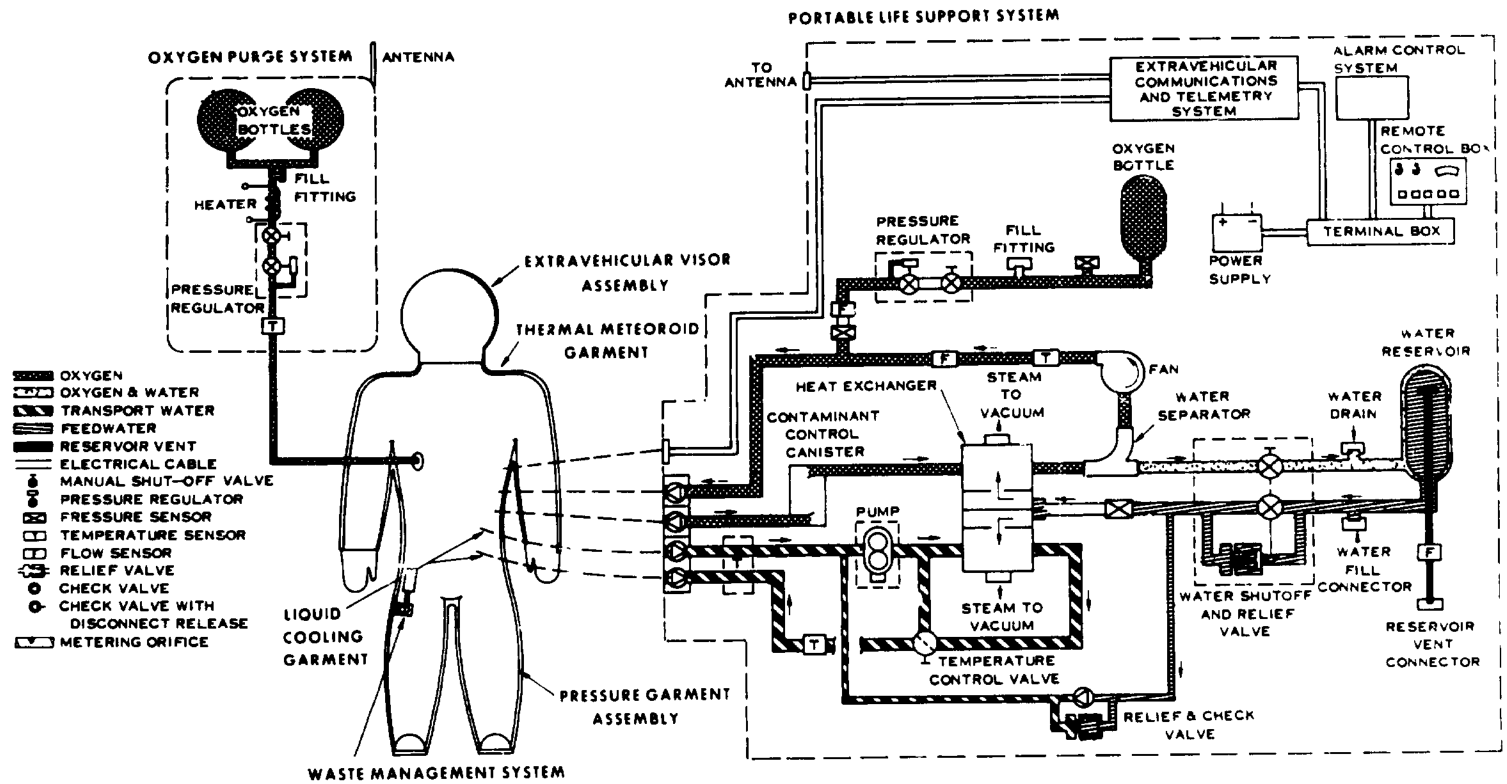

FIGURE 1. APOLLO EXTRAVEHICULAR MOBILITY UNIT SCHEMATIC

ities as exploring the lunar surface, gathering samples for return to earth, and emplacing scientific instruments on the lunar surface for gathering and transmitting information to earth.

The EMU life support system is comprised of the Portable Life Support System (PLSS) and the Oxygen Purge System (OPS), shown in figures 2 and 3, which were designed and developed by the Hamilton Standard Division of the United Aircraft Corporation.

The primary purpose of the PLSS is to condition and replenish the atmosphere inside the space suit and to cool the suited crewmen. The PLSS maintains space suit inlet oxygen pressure at 3.85 ± 0.15 psia and controls inlet gas temperature, carbon dioxide, odor, particulate contamination, and moisture levels inside the suit, for average metabolic rates of 1,200 to 1,600 Btu/hr and short term peaks of up to 2,000 Btu/hr. In addition, the PLSS cools the suited crewman by supplying and circulating cool water through a network of tubes built into the space suit undergarment in such a way that the tubing comes into contact with the skin. The skin is cooled by direct conduction, and the mean skin temperature is lowered to a level where little, if any, perspiration occurs.

Each of the two PLSS units carried to the lunar surface is rechargeable from spacecraft supplies to allow multiple excursions. The PLSS is designed to operate for periods of up to four hours without recharging, with three hours for nominal missions and one hour reserved for contingency operations.

The high pressure oxygen subsystem consists of a primary oxygen tank, oxygen fill fitting, pressure regulator assembly, primary oxygen pressure transducer and oxygen flow sensor. The primary oxygen tank is a cylindrical stainless steel vessel which operates at a nominal pressure of 900 psia and contains 1.12 pounds of oxygen for metabolic consumption and a nominal leakage of 200 scc/min from the EMU. The pressure regulator is a single-stage device maintaining suit inlet pressure at 3.85 ± 0.15 psia. The primary pressure transducer provides a telemetered signal of primary oxygen tank pressure and the oxygen flow sensor provides a warning signal in the event of excessive oxygen flow.

Gas circulation is provided by a centrifugal fan which produces a constant ventilation flow of 6 cfm. The contaminant control canister, which contains a filter, three pounds of lithium hydroxide and five ounces of activated charcoal, maintains acceptable carbon dioxide, odor and particulate contamination levels. The canister is a radial flow device in which gas is introduced to the center of the cylindrical cartridge, flows first through the charcoal, then through the lithium hydroxide and last through the filter material and is collected at the periphery.

Thermal control is maintained by a porous plate sublimator. Water is supplied to the sublimator at 3.7 psia from a water reservoir. The water enters the sublimator between the surfaces to be cooled and the porous nickel plates. The other side of the porous plates are exposed to ambient space vacuum conditions. The water flows into the pores in the porous plates, is exposed to the vacuum, and freezes when the vapor pressure approaches the triple point. Heat is then rejected by sublimation of the ice directly to space.

PORTABLE LIFE SUPPORT SYSTEM (PLSS)
OXYGEN PURGE SYSTEM (OPS)

FIGURE 2. BACK VIEW

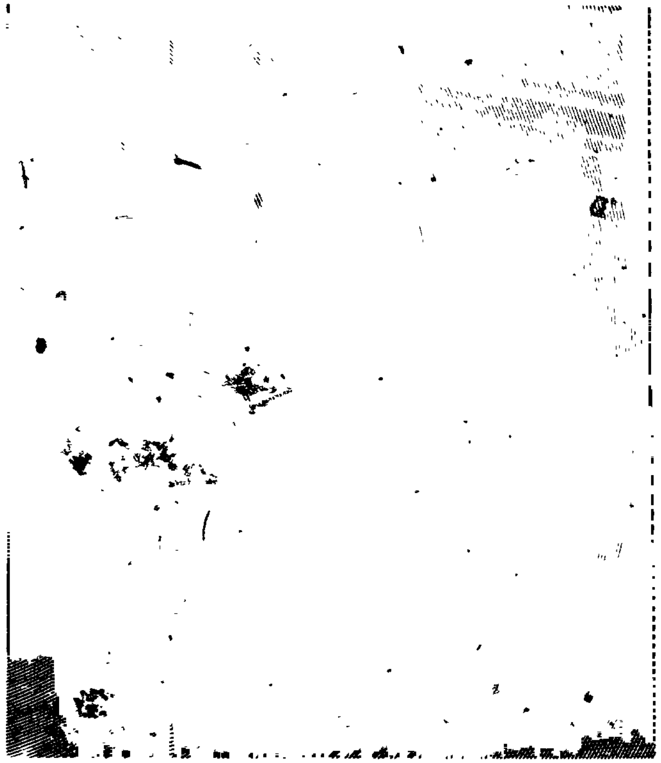

FIGURE 3. FRONT VIEW

Moisture collected by the ventilation stream is condensed on the gas side of the porous plate sublimator. As the condensate leaves the sublimator, it passes through an elbow which throws the droplets against the duct walls. As the condensate flows along the duct wall, it is trapped by a wick-type water separator and collected.

The prime mover for the liquid transport loop is a positive displacement diaphragm pump which circulates a constant flow of 4 lbs/min. Comfort control is accomplished with a sublimator bypass loop and a manual diverter valve. The diverter valve allows the suited crewmen to select one of three inlet water temperature settings, according to the amount of effort he is expending.

The PLSS contains numerous sensors and transducers to monitor PLSS performance and status of the suited crewman. The following parameters are monitored and telemetered:

a. Primary oxygen pressure

b. Pressure garment assembly (PGA) pressure

c. Liquid cooling garment (LCG) inlet water temperature

d. LCG differential temperature

e. Battery current drain

f. Battery terminal voltage

g. PGA ventilation flow

h. Primary oxygen flow

i. Sublimator gas outlet temperature

j. Feedwater pressure

k. PGA inlet CO_2 partial pressure

l. EKG

Two-way communication between two astronauts on the lunar surface, between one astronaut on the lunar surface and one in the lunar module, and telemetry of critical parameters is provided by the PLSS Extravehicular Communication System (EVCS). The PLSS power supply is a silver-zinc battery consisting of eleven cells in series. Battery capacity is 240 watt-hours with a nominal terminal voltage of 16.8 volts. The PLSS weighs 78 pounds and has a volume of 5,300 cubic inches.

The Remote Control Unit (RCU) is chest-mounted to provide adequate astronaut accessibility and visibility. The RCU contains all PLSS electrical controls and displays. The RCU weighs 4 pounds and has a volume of 60 cubic inches.

The Oxygen Purge System (OPS) is an emergency system designed to provide backup protection for the suited crewman in the event of a PLSS functional failure or for crew transfer in the event of a lunar module docking failure. The OPS is an open loop flush flow system which provides 30 minutes of oxygen at a flow rate of 8 pounds/hour to the crewman's helmet. This flow rate provides sufficient oxygen for breathing, respiratory product washout, visor defogging and EMU external leakage.

The OPS consists of two spherical tanks which hold a total of 4 pounds of useable oxygen at 5,880 psi, a single-stage pressure regulator which maintains suit inlet pressure at 3.7 ± 0.3 psi, a heater to maintain suit inlet oxygen temperature between 30 and 80°F, a temperature sensor connected to an automatic heater controller, and a silver-zinc battery to provide power.

The OPS weighs 40 pounds and has a volume of 1,400 cubic inches. For normal use, it is mounted on top of the PLSS; when being used independently, for vehicle transfer, it is worn as a chestpack by the crewman.

In summary, the Apollo EMU PLSS provides oxygen supply and pressurization, thermal control, humidity control, contaminant control, and communications and telemetry for EVA missions up to four hours in duration. The OPS provides emergency oxygen supply and pressurization for periods up to thirty minutes. While future EVA life support systems must still supply these basic functions, future mission requirements will demand that these life support systems be less encumbering, require less vehicle storage volume and weight as the number of EVA missions increase, and permit increased operational flexibility in planning and performing EVA missions.

FUTURE LIFE SUPPORT SYSTEMS

The well being of the suited crewman and the success of the mission will depend to a large extent upon the ability of the life support system to maintain a predefined environment throughout the EVA mission. Performance requirements for next generation EVA life support systems are projected as follows:

a. Suit pressure	3.7 - 5.0 psia
b. Suit inlet CO_2 partial pressure	4.0 mm Hg max
c. Ventilation flow	7 cfm
d. Suit inlet vent flow temp	55° to 85°F
e. Suit inlet dewpoint	50°F max
f. EVA system external leakage	200 scc/min
g. Metabolic expenditure range	400 - 3,500 Btu/hr
h. Average metabolic rate	2,000 Btu/hr
i. EVA equipment thermal load	1,000 Btu/hr
j. Environmental heat leak in	250 Btu/hr
k. Environmental heat leak out	350 Btu/hr

Although there are many combinations of processes and components that would satisfy the performance requirements outlined above, we have selected for consideration in the trade studies only those appropriate for "next generation" EVA life support systems.

The three life support subsystems that have the greatest impact on the EVA equipment requirements and vehicle constraints are the thermal control, oxygen supply, and carbon dioxide control subsystems. Therefore, each of these areas will be discussed in detail. Concepts within each of the areas will be compared, considering both life support system and space vehicle parameters; the resultant curves of subsystem equivalent weight and volume versus EVA mission duration and vehicle launch weight and volume penalties versus number of EVA missions are presented at the conclusion of each subsystem discussion. In addition to future concepts for life support subsystems, present technology items are identified for reference and to indicate possible improvements.

Thermal Control

Figure 4 shows schematically three candidate thermal control subsystems. Figure 4A shows a subsystem very similar to the existing Apollo EMU PLSS thermal control subsystem. The crewman is cooled by supplying and circulating cool water through a liquid cooling garment which is in direct contact with the crewman's skin. Thus the skin is cooled by direct conduction. The coolant is circulated by a positive displacement pump which is powered by a lithium halide battery. Heat is rejected to vacuum by a porous plate sublimator. Expendable water is stored in a water reservoir and supplied to the sublimator by back pressuring the bladder in the water reservoir. Comfort control is accomplished with a sublimator bypass loop and a manual temperature control valve. The temperature control valve permits the crewman to select any desired temperature setting (within a predetermined range), according to the amount of effort he is expending.

Figure 4B shows a heat pump/deployed radiator system which utilizes the crewman as an evaporator. A coolant (such as Freon F-113) is circulated through a cooling garment which is in direct contact with the crewman's skin. The crewman's metabolic heat production is removed by forced evaporation of the coolant. The vapor is then compressed by a high-speed centrifugal compressor which is powered by a lithium-halide battery. The hot vapor passes through a radiator which rejects heat to deep space. The vapor then passes through an expansion valve and is condensed. As in the previous system, comfort control is accomplished with a bypass loop and a manual temperature control valve.

The radiator itself is physically a separate component from the remainder of the life support system. It is deployed at the EVA worksite by the suited crewman and is connected to the life support system by umbilicals. It is easily transportable and may even be carried to and from the work site by the crewman prior to and after each EVA mission.

Figure 4C shows a system employing forced evaporative cooling within a water vapor permeable suit. Expendable water is fed into a cooling garment composed of a network of wicking material. The crewman's metabolic heat production forces evaporation of the water in the wicking material which is then evacuated to vacuum through the water vapor permeable suit. Expendable water is stored in a water reservoir and supplied to the wicking material by back pressuring the bladder in the water reservoir.

Trade-off studies for thermal control subsystems equivalent weight and volume versus mission duration, and vehicle weight and volume penalty versus number of EVA missions are shown in figures 5 and 6 respectively.

O_2 Supply/CO_2 Control

Due to the dual nature of some of the candidate subsystems, O_2 supply and CO_2 control subsystems have been combined for joint evaluation. Candidate O_2 Supply/CO_2 control subsystems are shown schematically in figure 7. Figure 7A depicts a subsystem similar in concept to the exist-

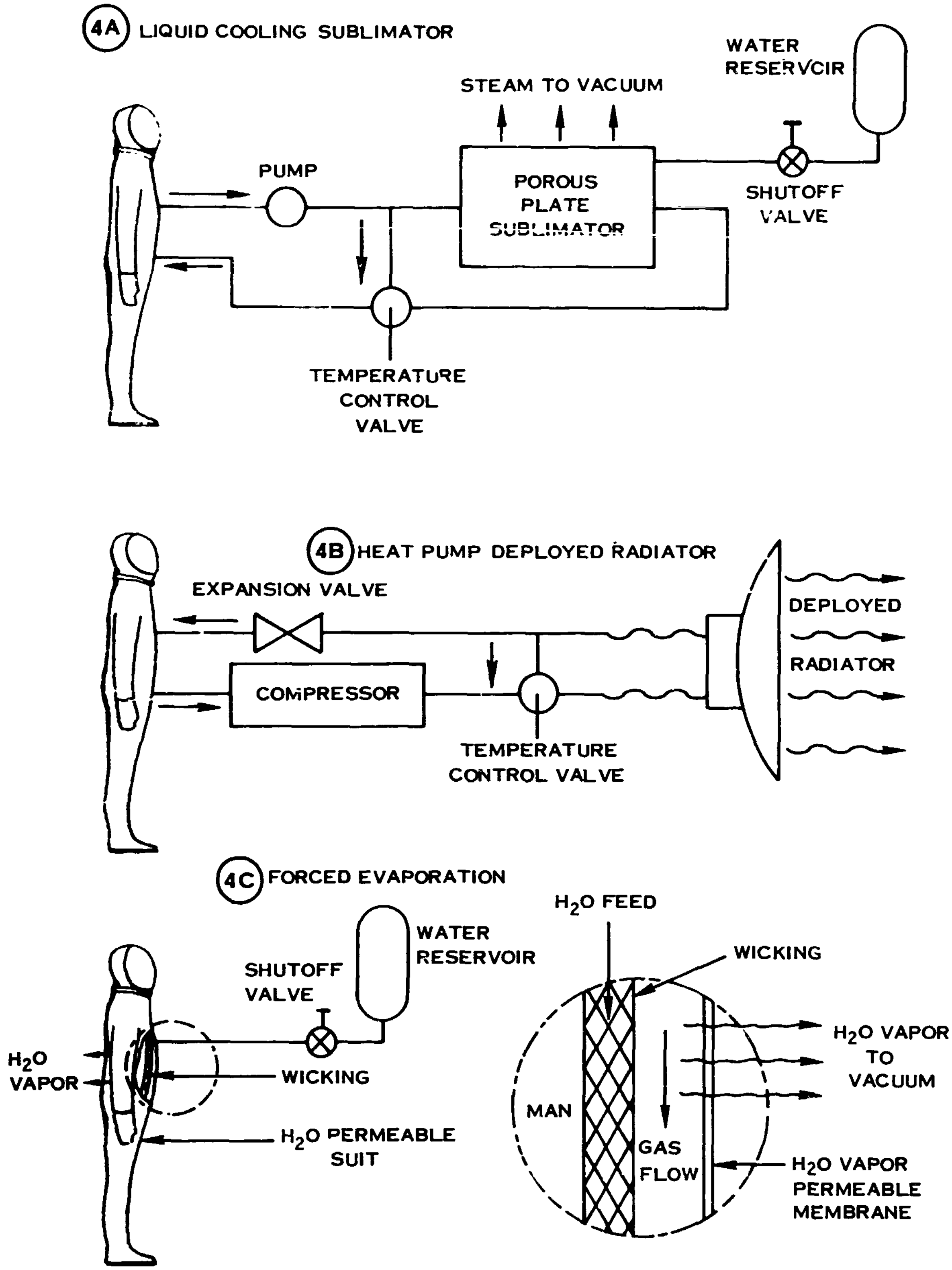

FIGURE 4 THERMAL CONTROL SUBSYSTEMS

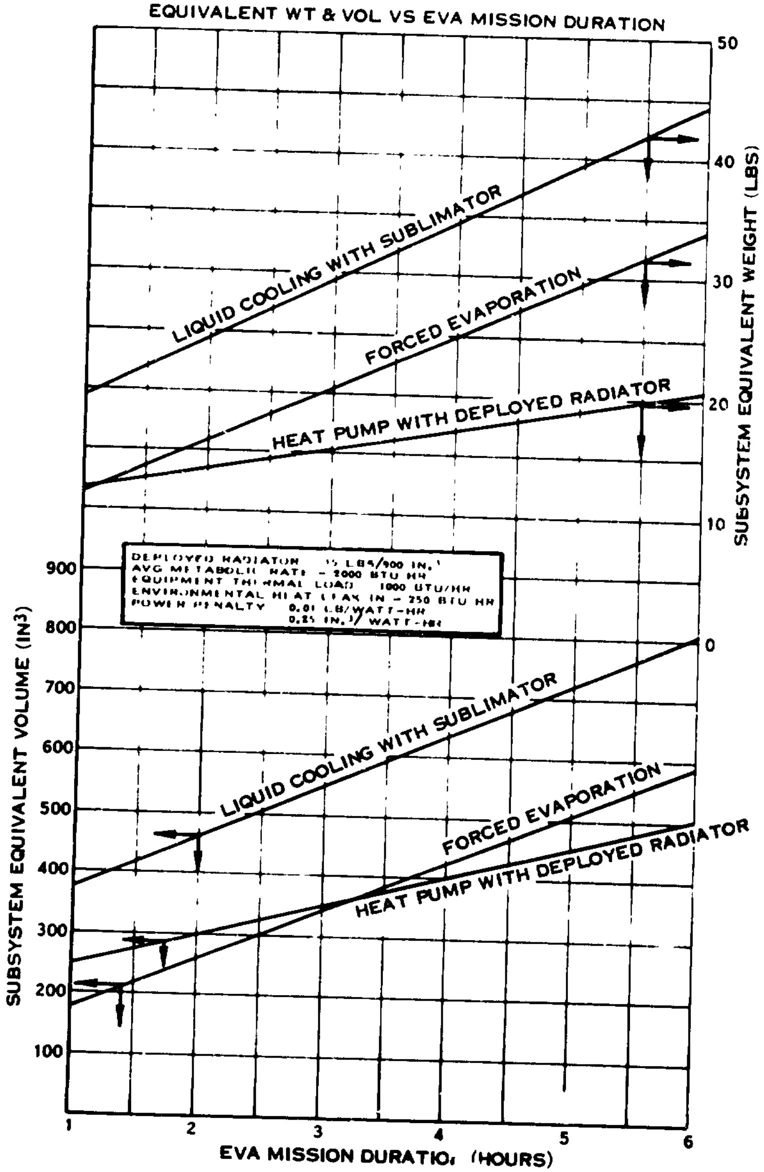

FIGURE 5 THERMAL CONTROL SUBSYSTEMS

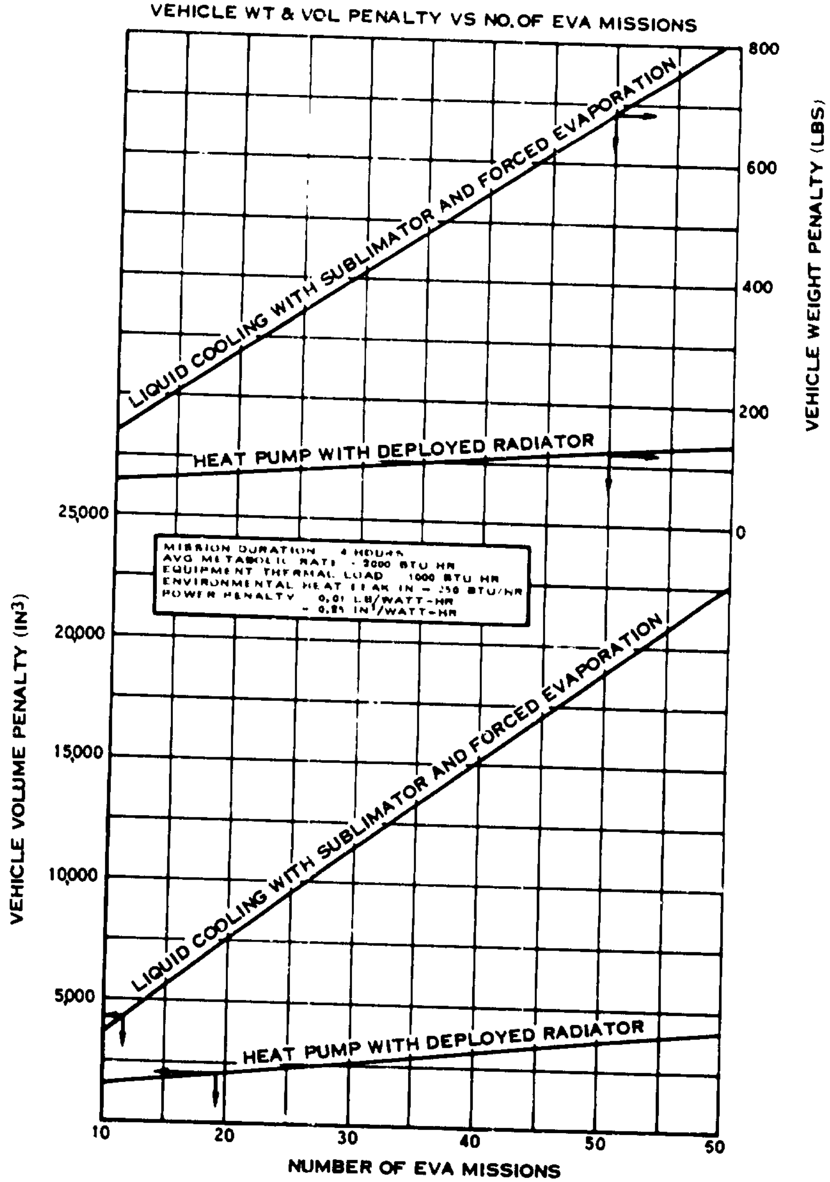

FIGURE 6. THERMAL CONTROL SUBSYSTEMS

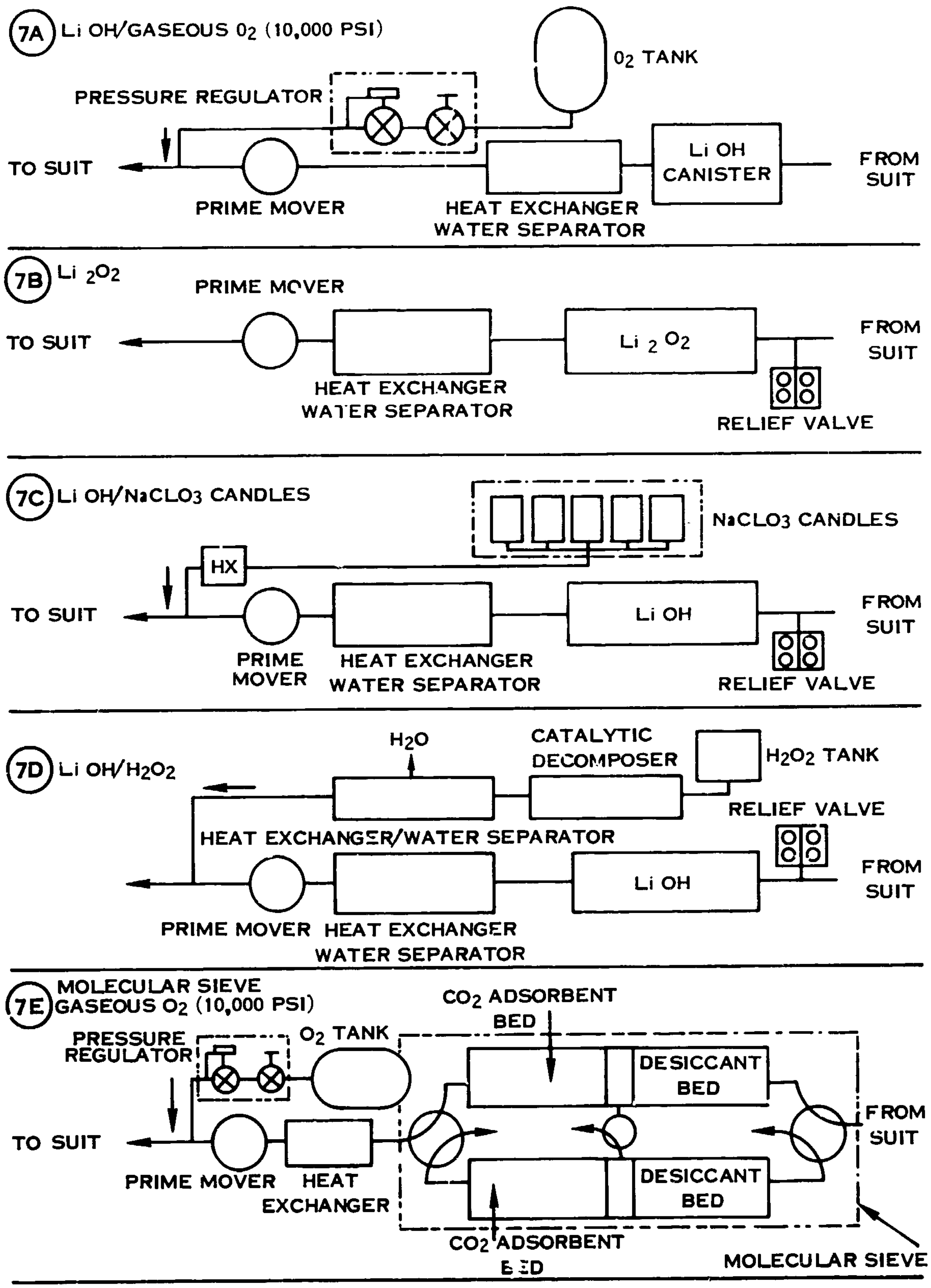

FIGURE 7. O_2 SUPPLY/CO_2 CONTROL SUBSYSTEMS

ing Apollo EMU PLSS O_2 supply/CO_2 control subsystem. Gaseous oxygen is stored at 10,000 psi and pressure regulation is accomplished by a single-stage pressure regulator which maintains suit inlet pressure within a predetermined range. Suit ventilation flow is introduced into a contaminant control canister containing lithium hydroxide. Carbon dioxide is absorbed by the lithium hydroxide thus maintaining acceptable CO_2 partial pressure levels.

An all-chemical O_2 supply/CO_2 control subsystem is shown schematically in figure 7B. Suit ventilation flow is introduced into a canister containing lithium peroxide. The carbon dioxide and water vapor react with the lithium peroxide bed resulting in the absorption of the CO_2 and the generation of gaseous oxygen. Pressure regulation is accomplished by bleedoff of the excess oxygen produced, thus maintaining suit pressure within a specified range.

Figure 7C shows a subsystem utilizing LiOH for CO_2 control and sodium chlorate candles for O_2 supply. CO_2 control is maintained in the same manner as described for the LiOH/gaseous O_2 subsystem (figure 7A). Sodium chlorate candles are a high density solid oxygen source which provides gaseous oxygen by initial ignition and a controlled burn rate. Pressure regulation is accomplished by bleedoff of the excess oxygen generated.

Figure 7D depicts a subsystem utilizing LiOH for CO_2 control and hydrogen peroxide for O_2 supply. CO_2 control is maintained as described previously. Hydrogen peroxide is stored in a storage tank in liquid form. H_2O_2 is supplied to a chamber where it is catalytically decomposed into water vapor and oxygen. The products of the reaction pass through a heat exchanger where the water vapor is condensed and separated. The separated water is fed into the thermal control subsystem and the oxygen is supplied to the suited crewman. Pressure regulation is accomplished by bleedoff of the excess oxygen generated.

The final O_2 supply/CO_2 control subsystem concept is shown in figure 7E. Gaseous oxygen is stored at 10,000 psi and pressure regulation is accomplished by a single-stage pressure regulator. CO_2 control is maintained by a molecular sieve which utilizes a regenerable solid adsorbent. Suit ventilation flow is introduced into the desiccant bed where water vapor is removed by adsorption. Flow continues into the CO_2 adsorption bed and adsorption continues until the bed is saturated (a predetermined time). At this time, automatic valving isolates the saturated beds from the process stream. The saturated beds are then exposed to vacuum resulting in desorption of the water vapor from the desiccant bed and CO_2 from the CO_2 adsorbent bed. At the completion of the desorption cycles, the automatic valving returns the beds to an "on line" condition and the cycle is repeated.

Two operating sets of beds are provided so that continuous removal of CO_2 occurs. The adsorption cycle is equal to the desorption cycle. After completion of each EVA mission, the molecular sieve is also desorbed thermally.

Trade-off studies for O_2 supply/CO_2 control subsystems equivalent weight and volume versus mission duration, and vehicle weight and volume penalty versus number of EVA missions are shown in figures 8 and 9 respectively.

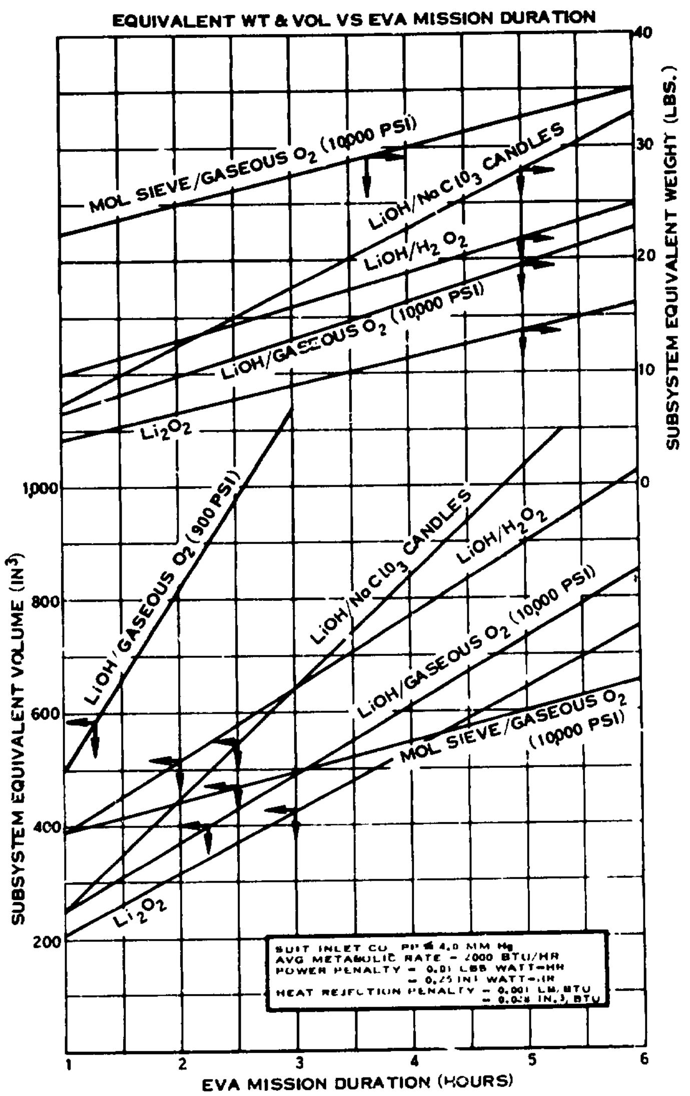

FIG. 8. O_2 SUPPLY / CO_2 CONTROL SUBSYSTEMS

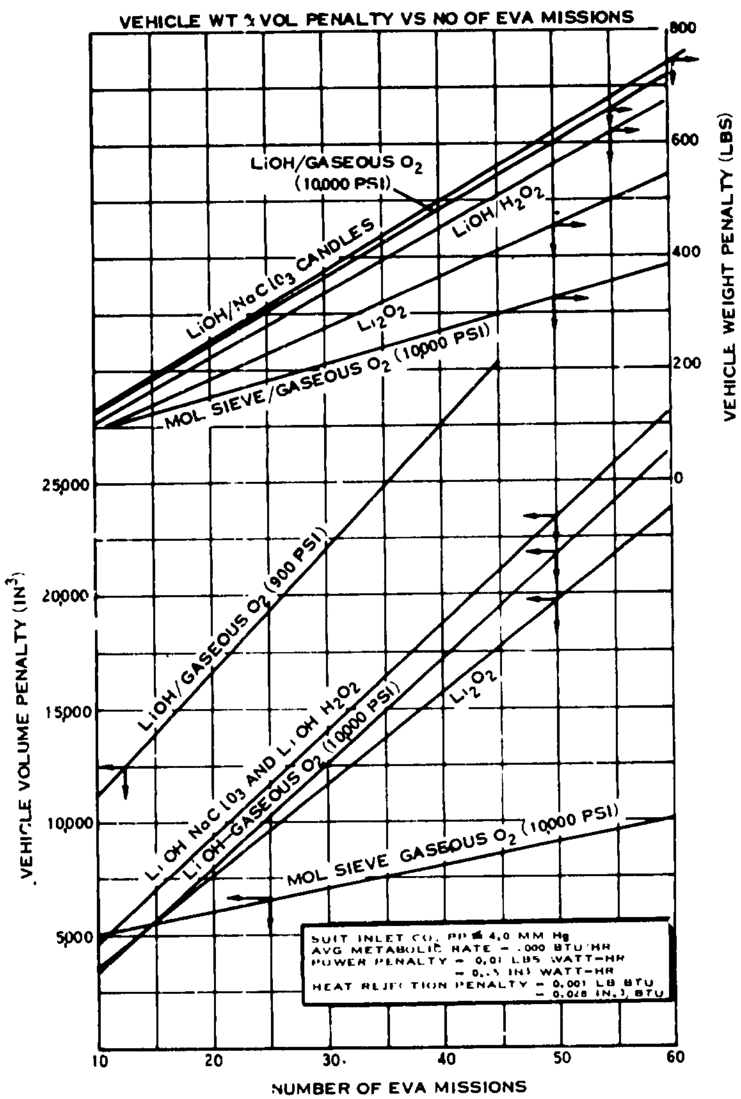

FIG. 9. O_2 SUPPLY/CO_2 CONTROL SUBSYSTEMS

CONCLUSIONS

The design of an EVA system involves a wide spectrum of requirements and considerations. Life support, as a fundamental element of the EVA system, is a major contributor to the overall evaluation of competitive system concepts. Life support comparisons, as presented in this paper, are based on both system and vehicle considerations. Selection of the life support subsystems is dependent mainly on conformance with the EVA mission performance requirements when the planned number of EVA missions are relatively low. Once program requirements dictate many EVA missions, total vehicle constraints such as launch weight and stowage volume constraints may become the major determinant in selection of an EVA life support system.

For space programs involving a low number of total EVA excursions, the forced evaporation and the heat pump/deployed radiator subsystems offer the optimum candidate thermal control subsystems. The heat pump/deployed radiator subsystem is only applicable for EVA missions in which there is a discrete worksite where a radiator may be deployed successfully. The optimum candidate O_2 supply/CO_2 control subsystem for programs with a low number of total EVA excursions is an all-chemical subsystem utilizing lithium peroxide.

As the total number of EVA excursions increase, regenerable subsystems (molecular sieve for CO_2 control) and subsystems utilizing a minimum of expendables (heat pump/deployed radiator subsystem) become more advantageous.

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SPACE-SUIT THERMAL CONDITIONING TECHNIQUES FOR FUTURE EXTRAVEHICULAR MISSIONS

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SUMMARY: Battelle Memorial Institute - Columbus Laboratories is conducting a conceptual study on new approaches to space-suit thermal conditioning. This study, which involves the application, combination, and extension of known conventional and unconventional techniques, has the objective of identifying the most promising techniques that can be foreseen today, irrespective of current hardware practicability.

INTRODUCTION

Space suits used for extravehicular activities serve to protect the astronaut from hostile environments, while permitting him to perform relatively complex functions with minimum physiological stress. The current-generation extravehicular suit is essentially an extension of the aircraft full-pressure suit, which is comprised of a helmet, overall garment, gloves, and boots, with metabolic heat removed by flowing, ventilating gas over the surface of the body. However, for the comparatively high metabolic heating rates that normally accompany extravehicular maneuvers, the effectiveness of cooling by gas ventilation alone has been found to be marginal at best. Extravehicular activities associated with future extended lunar and interplanetary missions are expected to increase the thermal performance requirements of space suits well beyond those for presently planned earth-orbit and lunar missions. Therefore, it will be necessary to provide increased thermal isolation from the environment for longer periods of time, and in some cases to extend the capability for rejecting heat to a higher ambient temperature. Consequently, the thermal-conditioning schemes that will be employed in future extravehicular space suits will most likely require the development of new technology to ensure that the needed advances in the state of the art do occur. The extent to which such advances can be made will undoubtedly be a major determinant in the type of extravehicular activities that will be feasible for future lunar and interplanetary missions.

To aid in establishing guidelines for future research-and-development activity, Battelle-Columbus is conducting a conceptual study aimed at generating new approaches to space-suit thermal conditioning. This study involves the application, combination, and extension of known conventional and unconventional techniques, with the objective of identifying the most promising schemes that can be foreseen today, irrespective of their current hardware practicability. The three primary areas of investigation are (1) physiological response, (2) body-heat transfer, and (3) system-heat rejection. A premise fundamental to this study is that the totally effective conditioning system must include and utilize the normal thermoregulatory mechanisms of the human body. This deviates somewhat from the more conventional approach to environmental control-that of treating the human body as a "black box" influenced by, but separated from, the environmental control system. This integrated approach, although not totally new, adds an important dimension to thermal conditioning of the astronaut in the extravehicular operating mode. Closing the loop through continuous feedback of physiological reaction to a conditioned preaction appears to be a necessary prerequisite for the development of the advanced thermal-conditioning techniques that will be required for future extravehicular activities.

This paper outlines the approach being used to conduct the current Battelle study and reviews the progress to date. Since the subject study is continuing, any conclusions in this paper, either stated or implied, are preliminary at this time, and, thus, are the subject of further analysis and evaluation. It is hoped that, although the subject study has not yet been completed, this paper will serve to stimulate further thought and interest toward developing the advanced space-suit thermal conditioning concept that will be required for the ambitious extravehicular activities envisioned for the future.

INVESTIGATION GUIDELINES

For this study the primary functions to be performed by a typical extravehicular space-suit thermal-conditioning system are to:

- Comfortably carry away metabolic heat and any attendant surface moisture from the astronaut
- Thermally protect the astronaut from his environment
- Provide appropriate heat sinks and/or heat sources.

In view of the simplicity, high degree of reliability, light weight, and compactness of electrical-resistance heating devices, it is assumed that only resistance heaters would be employed where auxiliary heat sources are required. Consequently, for the subject study the main emphasis is on investigating schemes for cooling the astronaut.

As previously noted, the primary goal of the study is to identify promising concepts that appear to warrant further research and development, without regard to present hardware practicability. Thus, the initial phase of the investigation is not concerned with the problem of how to incorporate the candidate thermal-conditioning schemes into a given space-suit design. For the most part, it is anticipated that space-suit design will have changed significantly by the time that any of the candidate systems are reduced to hardware. In addition, it is highly likely that advanced thermal-conditioning-system configurations will be instrumental in determining how future-generation space suits will be designed.

In view of the broad nature and long-range objectives of this study, specific mission requirements are not considered per se. Instead, each of the candidate thermal-conditioning concepts is being evaluated on the basis of its particular performance limitations as well as its most favorable performance range. This should ultimately permit classification of these concepts in such a manner that those showing the most potential can be selected for a given extravehicular mission for the specified environmental conditions.

The three primary areas selected for investigation are (1) physiological thermal response, (2) body surface heat and mass transmission, and (3) system-heat rejection. Although this study treats each of these areas separately, it must be recognized that they are closely related, so that the eventual evolvement of potentially useful advanced thermal-conditioning techniques requires integration into total-system concepts.

PHYSIOLOGICAL CONSIDERATIONS

The three primary physiological considerations for space-suit thermal conditioning are (1) metabolic heating rate, (2) skin surface temperature, and (3) sweat- or moisture-removal rate. A fourth consideration of utmost importance is the natural thermoregulatory mechanism, which is essentially the integrating factor for the first three.

Metabolic heating rates for performing physical tasks in a pressure suit are significantly higher than for the same tasks in a natural, unconstrained environment. For example, previous research studies have shown ratios as high as 5 or 6 to 1 with current-generation space suits. Considering that advances in the art of space-suit design expected for the future will most likely reduce the physical encumbrance and, thus, the metabolic heating rate, this ratio for a given work task should be lower for future extravehicular activities. However, for this investigation a maximum metabolic heating rate of 3750 Btu/hr for strenuous extravehicular activity is used for the upper limit, without regard for possible space-suit structural design improvements. This value of 3750 Btu/hr, which represents the maximum cooling load for the thermal conditioning schemes being considered, was selected as a representative value through review of the literature and discussions with other researchers currently active in this field. For nominal sedentary activity the lower limit was taken to be 500 Btu/hr.

A number of previous studies of human performance in a space environment have established the acceptable-comfort mean skin-temperature range to be from 86 to 94 F. These thresholds were defined on the basis of induced metabolic heat generation by shivering at 86 F and excessive sweating at 94 F. Consequently, this study is concerned with thermal conditioning techniques with the potential for maintaining a mean skin temperature between 86 and 94 F.

The third primary physiological consideration is the sweat rate, both active and insensible. It has been determined that high active sweat rates in conjunction with high body temperatures induce undesirable physiological stresses in humans performing in a space environment. One major consequence is that high continuous latent cooling loads will in time dehydrate the body. On the other hand, a zero sweat rate requirement for the body thermal conditioning appears to be unnecessary, and relatively low active sweat rates are generally considered to be desirable. With regard to sweat rate as a thermal-conditioning performance criterion, the fundamental concern is that there is provision for adequate cooling capacity to remove skin surface moisture at a comfortable body temperature for any reasonable sweat rate. The maximum sweat rate, or conversely, the maximum moisture-removal rate, has been estimated to be 3750 Btu/hr for this study, which corresponds to an equilibrium heat dissipation of 3750 Btu/hr, with total latent evaporation. It is recognized, of course, that the actual sweat rate varies within and from individual to individual, so this maximum value represents a composite.

Table 1 summarizes the major thermal performance criteria dictated by physiological considerations for the thermal conditioning schemes being considered in this study.

TABLE 1. THERMAL-CONDITIONING-SYSTEM PERFORMANCE CRITERIA

Heat Removal Rate-500 to 3750 Btu/hr

Mean Skin Temperature-86 to 94 F

Moisture Removal Rate-0 to 3750 Btu/hr

BODY-HEAT TRANSFER

Any thermal-conditioning techniques ultimately selected for space-suit environmental control must, of course, employ one or a combination of the basic modes, i.e., conduction, convection, or radiation, to transfer heat to and from the astronuat's body. Many of the thermal-control schemes now envisioned would be configured with the major system components external to the suit, most likely back pack, with fluids cycling to and from the body region to effect conductive or convective heat transfer. Radiation heat transfer to the suit wall may also be employed in conjunction with either of these modes.

The heat-transfer modes normally available for human comfort conditioning in the terrestrial environment will be either absent or constrained in the extraterrestrial environments considered for this study. Radiative heat transfer from the surface of the body is restricted when the body is enclosed in space suits having the wall temperature approximately equal to body surface temperatures. Convective heat transfer is reduced both by the absence of gravity-induced buoyancy effects and by the comparatively low pressure of the gas normally used to surround the body. Therefore, it does not appear feasible to provide thermal conditioning by attempting to reproduce the normal terrestrial environment.

The environmental-control technique that has been used for previous manned extravehicular activities is ventilation cooling, whereby heat and mass transfer are effected by forced convection. With this scheme, the majority of the cooling load is latent, resulting from sweat evaporation. Ventilation cooling has the advantage of being able to utilize the natural thermoregulatory mechanism for body thermal control. However, as previously noted, this dependence on the perspiration mechanism can be a severe disadvantage at high metabolic heat rates if high body temperatures are required to produce excessive sweat rates.

Results from previous studies indicate that environmental control by ventilation cooling alone is restricted to metabolic heating rates below 1200 Btu/hr with suit pressures of 3.5 psia. Considering that metabolic heating rates may well approach 3800 Btu/hr for future extravehicular activities, ventilation cooling by conventional means appears to be impractical. However, even when the body is maintained at conditions that do not induce appreciable active sweating, some perspiration removal (including insensible) will normally be required. Consequently, techniques that rely on a primary mode of heat transfer other than forced convection may still require some gas ventilation for sweat removal. Ventilation-gas circulation may also be employed on a localized basis to the high-sweat areas such as armpits, groin, hand palms, and foot soles.

Cooling the body by conduction can be induced by the use of a coolant loop with elements in contact with the skin. This approach establishes comfort conditions essentially by maintaining the skin temperature within a specified range through nearly constant heat-transfer rates. One technique now being developed uses a liquid circulating from regional body-heat exchangers to an external heat sink. The main advantage of such a scheme is the potential for obtaining comparatively high heat-transfer efficiencies. However, since the conduction approach does not make use of the sweat mechanism for body thermal control, the thermoregulatory mechanism is in effect inhibited, and, therefore, has little influence in controlling internal body temperature. Consequently, although the absence or reduced rate of sweating may offer a decided physiological advantage from the standpoint of low physiological stresses at comparatively low metabolic heating rates, the complete obviation of the sweating condition could be a serious disadvantage for the high metabolic heating rates normally produced by extravehicular activities. In this regard, the liquid-loop conductive-cooling technique presently appears to be limited to heat removal rates in the range from 2000 to 2500 Btu/hr.

Cooling the body passively by radiation to the space-suit wall appears to offer an inherent design simplicity. However, it has been determined that for a radiation gap existing between the body and the wall of the space suit, the maximum theoretical heat-transfer rate is about 1100 Btu/hr, the limiting condition being potential ice buildup on the inner wall of the suit. Therefore, the use of a passive radiation gap alone appears limited to comparatively low metabolic heating rates.

From the above considerations one might assume that the most practical extravehicular-activity thermal-conditioning techniques must rely on a combination of the basic modes for effective body-heat transfer. For example, it is reasonable to expect that the most workable system will be one that combines sensible and latent heat removal on a localized basis at various points on the body, in which case the judicious zoning of the conductive and the convective heat-transfer modes would be in order.

HEAT-REJECTION CONSIDERATIONS

Under most conditions, it will be necessary to reject heat from the space-suit thermal-conditioning system directly to the environment or to a heat sink. Ultimately, the heat must be removed by radiation, conduction, convection, sublimation, or evaporation. For survival in environments having an equivalent ambient temperature higher than those normally encountered on Earth, the use of a powered refrigeration system must be considered. Refrigeration techniques selected must provide extended-temperature-range capability and adaptability, low heat-rejection requirements, compactness, light weight, low power requirements, and zero-g operational capability.

For the current study, the identification and preliminary assessment of candidate refrigeration systems was preceded by a brief examination of the basic physical processes that can potentially be employed to reject heat. A brief discussion of these basic processes is presented in the following section preparatory to the discussion in succeeding sections of some of the refrigeration approaches being considered in this study.

Basic Cooling Processes

The basic physical processes considered for this study are those that, when incorporated into a system, are capable of effecting a reduction in the system temperature with respect to its surroundings, thereby placing the system in a position to absorb energy (i.e., heat) from these surroundings. These basic cooling processes can be catagorized under three major headings, each of which describes the principal change of state responsible for inducing a system to lower its temperature below that of its surroundings. These major headings, along with appropriate examples and subdivisions are as follows:

1. Mechanical Changes of State
    a. Single-component-system phase changes, including vaporization, melting, sublimation, and solid state
    b. Binary-component-system phase change, principally vaporization
    c. Expansion with transfer of external work, including gaseous phase, liquid phase, and solid phase
    d. Expansion without transfer of external work, including vortex action and gas throttling

2. Chemical Changes of State

   a. Dissociation reactions

   b. Solution effects

   c. Sorption effects

3. Electrical and/or Magnetic Changes of State

   a. Thermoelectric

   b. Thermoelectric-magnetic

   c. Thermomagnetic.

It is generally accepted that any refrigeration cycle suitable for heat rejection in a space-suit thermal-conditioning system will employ one or a combination of these basic cooling processes. Consequently, they essentially represent the rudiments for all advanced thermal-conditioning schemes.

On the basis of a preliminary assessment, the following cooling processes appear to warrant the primary consideration for space-suit heat rejection. It is emphasized however, that at the present time no single process has been totally ruled out.

- Vaporization (single-component systems)
- Sublimation (single-component systems)
- Vaporization (binary-component systems)
- Expansion of a gas with transfer of external work
- Expansion of a liquid with transfer of external work
- Vortex action (Ranque-Hilsch tube)
- Throttling of a gas (Joule-Thompson effect)
- Thermoelectric (Peltier devices).

Conventional Refrigeration Cycles

The cycles being investigated are grouped under two headings according to the degree of availability of the energy that is directly supplied to motivate the cycle. Cycles receiving motivational energy in a form that is totally available for doing work, such as mechanical or electrical energy, are grouped under the heading "Explicitly Driven Cycles". Cycles receiving motivational energy in the form of heat that is not totally available for doing work, are grouped under the heading of "Implicitly Driven Cycles".

The discussion of each of the conventional cycles in this section centers around the efficiency of the particular cycle. This is because cycle efficiency offers a convenient means for performing a preliminary assessment, in that efficiency directly relates heat-rejection capability and power requirement to a given cooling requirement. It is important to note, however, that efficiency is but one of the standards that must ultimately be used to compare candidate systems and that other salient features may actually contribute more toward determining the long-range potential for a particular system.

Explicitly Driven Cycles

Vapor-Compression Cycle. Because the explicitly driven vapor-compression cycle is the basis for most air-conditioning systems (and also for the vast majority of existing refrigeration systems), it becomes for this investigation a practical basis for comparison with other cycles. The comparisons should then reveal which cycles from a thermodynamic viewpoint appear to offer possibilities for significant improvements.

To be concise, the vapor-compression refrigeration cycle forms the basis of the refrigeration and air-conditioning art, because it is the closest practical approach to Carnot-cycle requirements yet devised. This close approach is due primarily to the employment of a condensable working fluid that inherently provides for heat-absorption and-rejection processes that are for the most part isothermal.

In considering this cycle specifically for use in a space-borne thermal conditioning system, it is noted that the mechanical vapor-compression system, which is very efficient under normal terrestrial conditions, may lose capacity rapidly as the heat-rejection temperature is extended above the normal range. Also, many of the refrigerants currently being used in vapor-compression machines have unacceptable long-term chemical stability at temperatures above 400 F. Therefore, further evaluation may show this cycle to have limited application to space-suit thermal conditioning.

Stirling and Ericsson Cycles. The Carnot cycle is not the only combination of processes that can provide for maximum efficiency of thermal devices. The requirement for attaining Carnot-cycle efficiency is simply that the heat-rejection and- absorption processes must be isothermal, with connecting processes that are isentropic or the cyclic equivalent. The connecting processes can be made effectively isentropic through regenerative heat exchange within the cycle. This principle is the basis for the Stirling and Ericsson cycles, which ideally operate at Carnot-cycle efficiencies. The basic difference between the Stirling- and Ericsson-cycle processes is that, for the Stirling cycle, the regenerative heat exchanges are considered to occur at constant volume, whereas, for the Ericsson cycle, these exchanges take place at constant pressure.

The application of these principles to an actual refrigeration machine has been limited, primarily because of technological problems. However, it is considered highly unlikely that this regenerative-cycle approach will continue to be rejected merely because present technology has not yet solved all the problems connected with this type of machine. Although this approach is presently considered to be inferior to that of the vapor-compression cycle, it is conceivable that future technological developments in mechanisms, lubricants, and heat exchangers will improve this situation to a point where the regenerative Stirling cycle can equal or possibly surpass vapor-compression equipment on an efficiency basis. Furthermore, when it is considered that Stirling-cycle machines maintain capacity substantially better than do vapor-compression machines at elevated heat-rejection temperatures, they may well hold promise for future space-suit thermal conditioning.

Brayton Cycle. This type of refrigeration system is based on a thermodynamic cycle of processes known variously as the Brayton cycle, Joule cycle, or air cycle. Although the vast majority of commercial and military aircraft depend either entirely or to a large extent on air-cycle refrigeration systems for personnel and equipment cooling, applications of this cycle to other air-conditioning practices have been rare. This is primarily because the Brayton cycle, when reduced to actual practice, is subject to certain irreversibilities that drastically reduce its potential. The most significant irreversibility is that accompanying the expansion process. During expansion, turbulence and, therefore, irreversible heating effects first reduce the work extracted from the gas and, second, reduce the refrigerating effect, since the gas leaves the expander at a temperature higher than ideal. The compression processes also suffer irreversibilities which require that the actual compression work be larger than the ideal quantity. The reduction in refrigerating capacity and increase in net work required naturally reduce the coefficient of performance and, therefore, increase power requirements.

Consequently, the comparatively low efficiency and resultant large power requirements of the cycle prevent it from competing with the vapor-compression cycle for normal terrestrial air-conditioning applications.

However, for space application, and particularly for extravehicular systems where low weight and compactness are of the utmost importance, closed Brayton-cycle cooling using small, high-speed equipment may eventually offer considerable promise.

Joule-Thomson Cycle. Perhaps the best way to present the Joule-Thomson refrigeration cycle is to make a comparison between it and the Brayton cycle. As shown in Figure 1, both cycles require a compressor, expansion device

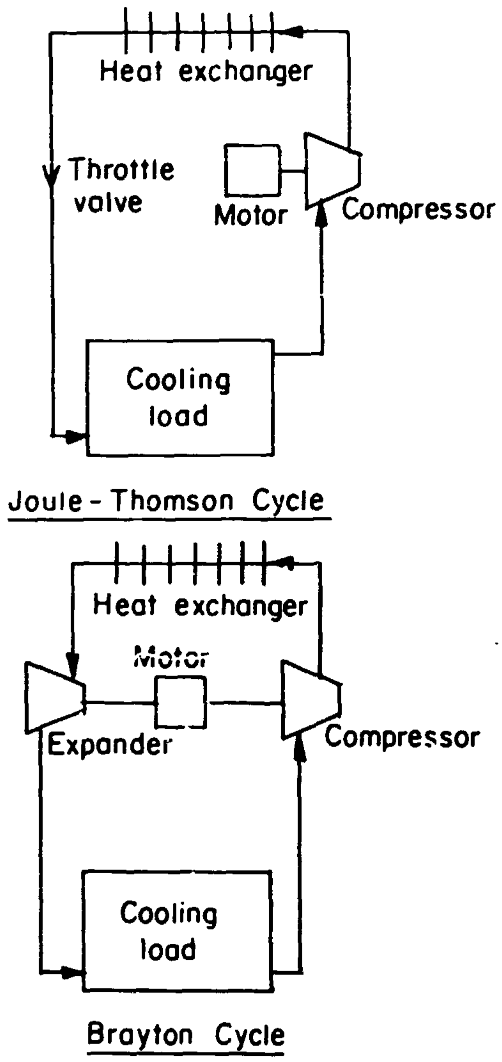

FIGURE 1. ARRANGEMENT OF COMPONENTS FORMING THE JOULE-THOMSON AND BRAYTON REFRIGERATION CYCLES

and two heat exchangers. The basic difference between the two cycles is, then, the means provided for expanding the working fluid from the high- to low-pressure regions in order to facilitate a temperature decrease. In the case of the Brayton cycle the expansion is accomplished in a machine that extracts some, but not all of the work that is ideally available. A Brayton-cycle expansion process is, then, partially irreversible, and the work that is extracted is considered to be returned to the compressor, thereby reducing the net work input required to motivate the cycle. The expander in a Joule-Thomson cycle is a throttling valve; therefore, the expansion process is totally irreversible and no work is extracted. Assuming for the moment that both cycles operate over the same pressure range, it becomes obvious that the Brayton cycle is the more efficient of the two.

To be at all effective a Joule-Thomson cycle must operate over a much greater pressure range than is normally required for a Brayton cycle. Higher pressures mean higher temperatures and, therefore, further deviations from Carnot-cycle requirements. It is quite evident that a temperature decrease obtained by a totally irreversible process does not provide an effective means for rejecting heat. Therefore, in spite of its apparent simplicity, the Joule-Thomson cycle may not be competitive with the Brayton cycle for space-suit thermal conditioning.

Peltier Devices. Thermoelectric devices are certainly mechanically attractive as means for rejecting heat in space-suit thermal conditioning systems, and therefore appear to be worthy of continued interest. However, it must be recognized that the reason thermoelectric cooling devices do not presently find wide application in air-conditioning practice is their inefficiency of operation when compared with other types of equipment.

Whereas a figure of merit of presently available thermoelements is approximately $(3.2 \cdot 10^{-3})(C)^{-1}$, a figure of merit of $(10 \cdot 10^{-3})(C)^{-1}$ is often quoted as being the value needed to make the Peltier device competitive with vapor-compression equipment. This obviously will require a major breakthrough in advancing the state of the art. This is not meant to imply that a figure of merit of $(10 \cdot 10^{-3}(C)^{-1}$ must be attained before the thermoelectric device appears attractive for space-suit thermal conditioning. Considering other attributes of these devices, it seems reasonable to assume that somewhat lower figures of merit will make them competitive in the future.

Implicitly Driven Cycles

The preceding discussion dealt with what has been termed explicitly driven cycles, that is, those refrigeration cycles that are considered to be directly motivated by energy in a totally available form such as shaft or electrical energy. The discussions to follow are concerned with what has been termed implicitly driven cycles, or those refrigeration cycles that are directly driven, or motivated by energy in the form of heat. Since heat energy is not totally available for doing work, a heat-motivated refrigeration cycle is actually a rather subtle combination of refrigeration cycle and heat engine. Expressed another way the available energy needed by the refrigeration section to pump heat from a lower temperature to sink temperature is derived from the heat-engine section, which absorbs heat at a temperature above sink temperature and ultimately rejects the unavailable and unused available portions to the sink.

Absorption Cycle (Two-Fluid Type). For the purpose of this discussion, it should be remembered that an absorption refrigeration cycle is in effect a vapor-compression cycle. The distinguishing feature between them is the means employed to change the state of the refrigerant vapor from evaporator to condenser conditions. In a mechanical vapor-compression cycle (that is, an explicitly driven cycle), the refrigerant vapor is removed from the evaporator and is delivered to the condenser at a higher pressure and temperature by a mechanical compression device, such as reciprocating- or centrifugal-type compressor. The absorption cycle accomplishes the same increase in pressure and temperature by first absorbing the refrigerant vapor in solution with a second liquid having an affinity for the vapor. The absorbing solution, rich in refrigerant, is then delivered to a region of higher pressure where it is subsequently heated to drive the refrigerant out of solution and into the vapor phase at condenser conditions.

Figure 2 shows a schematic arrangement of the components forming the basic absorption refrigeration cycle. The condenser, throttling valve, and evaporator appearing to the left of Line AA in Figure 2 perform the same functions as in a mechanical vapor-compression cycle. The components to the right of Line AA, i.e., the absorber, heat exchanger, generator, liquid pump, and liquid-pressure-reducing valve, all help to perform the vapor-compression process.

In general, an absorption system is not an efficient refrigeration device, primarily because it is forced to absorb its motivation heat energy at a relatively low temperature, a condition that does not allow for a high thermal efficiency of its engine section. Absorption systems do have the advantage, however, of operating with low-temperature heat sources that would normally be considered as waste heat. In this regard, it is conceivable than an absorption system used in an extravehicular-activity space suit might be solar motivated.

Recent conceptual studies at Battelle-Columbus have produced an absorption-system cycle capable of rejecting heat at substantially higher temperatures than current state-of-the-art systems. There is reason to believe that a system of this type may hold promise for future space-suit thermal conditioning.

Absorption Cycle (Three-Fluid Type). Figure 3 is a simplified diagram for an absorption cycle that eliminates the need for a refrigerant throttling valve and a mechanical circulation pump, although the basic operation of this cycle is identical to the preceding absorption cycle. The feature that distinguishes this cycle is the use of an inactive gas in the evaporator-absorber section. The gas is termed inactive because it neither acts as a refrigerant working fluid nor as a thermal-engine working fluid. The primary purpose of the inactive gas is to depress the vapor pressure of the refrigerant in the evaporator-absorber section while permitting the entire system to operate at essentially the same total pressure.

Thermodynamically, this cycle does not offer advantages over the two-fluid absorption cycle previously discussed.

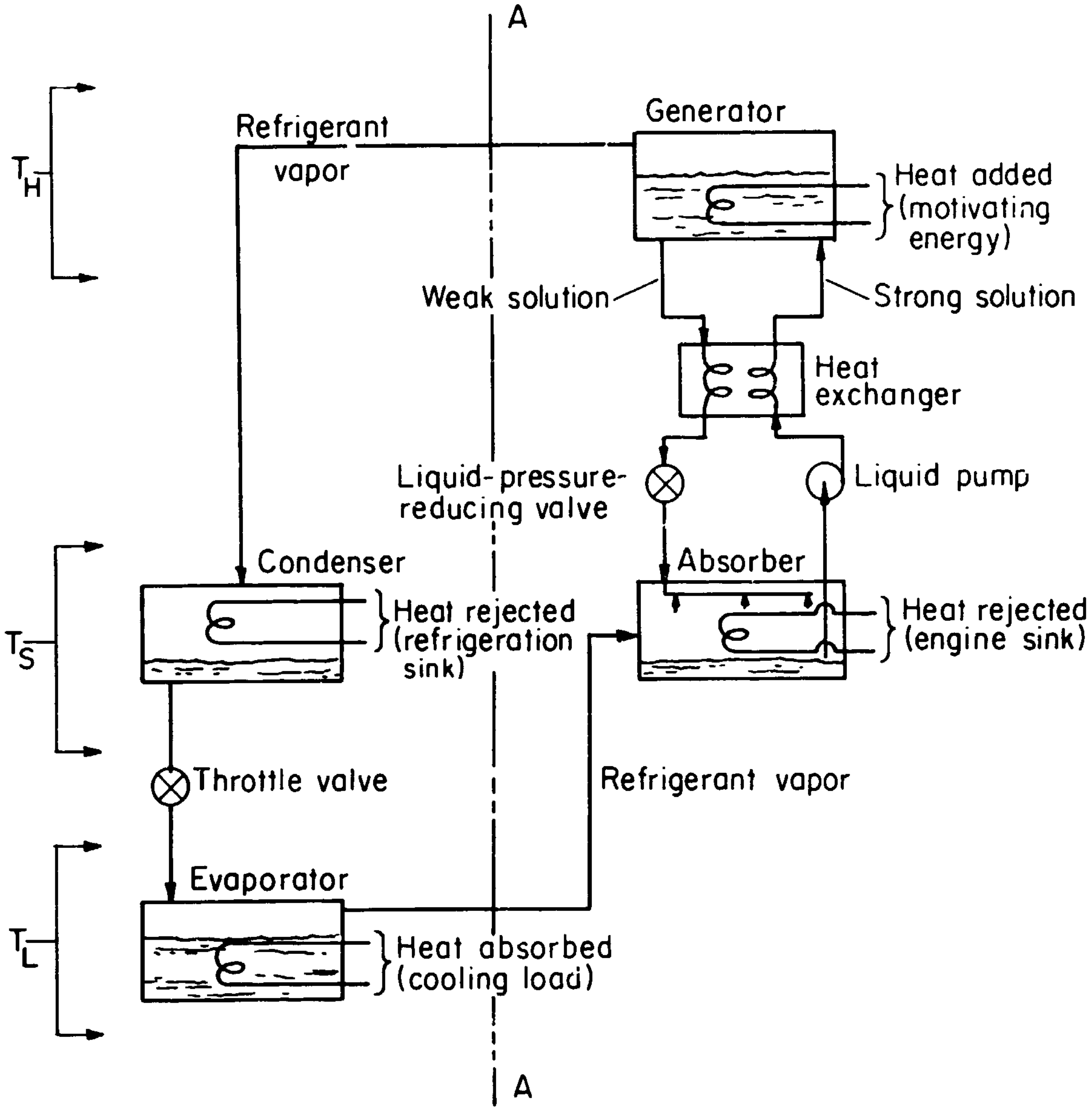

FIGURE 2. SCHEMATIC ARRANGEMENT OF COMPONENTS FOR THE BASIC ABSORPTION REFRIGERATION CYCLE

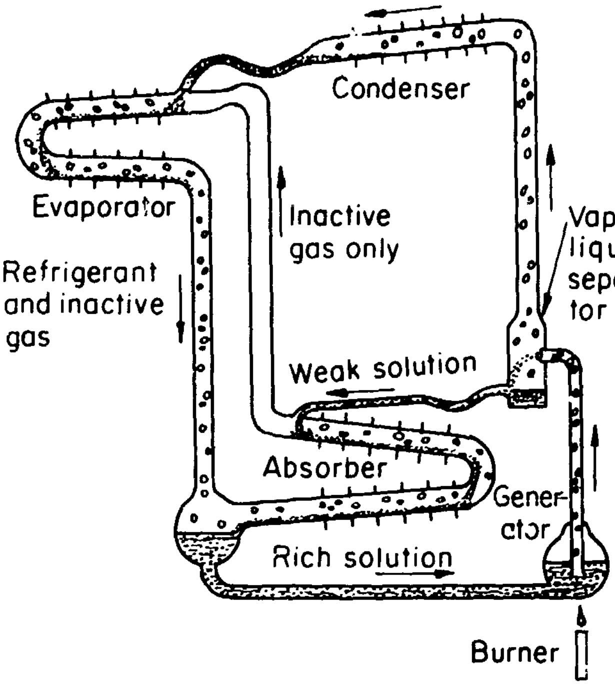

FIGURE 3. SIMPLIFIED DIAGRAM OF AN ABSORPTION CYCLE EMPLOYING AN INACTIVE GAS IN THE EVAPORATOR-ABSORBER SECTION

From the mechanical standpoint, it is extremely simple and, therefore, somewhat crude in the manner in which some of the processes, such as absorption, are effected. This normally results in a machine that is rather large for a given capacity.

Ejector Cycle. The ejector refrigeration cycle, shown schematically in Figure 4 is another implicitly coupled heat-motivated cycle based on the vapor-compression cycle. In this cycle, the compression of the refrigerant vapor from evaporator to condenser conditions is accomplished by an ejector. As shown, the motivational heat is added to a portion of the liquefied refrigerant in a vapor generator. The resulting vapor is conducted to the ejector where a portion of its enthalpy is transformed, by the expansion nozzle, into the mechanical kinetic energy of a high-speed stream. The high-speed stream leaving the nozzle entrains refrigerant vapor from the evaporator, and together they enter the diffuser section of the ejector. The diffuser decelerates the combined stream in an attempt to recover as much of the mechanical kinetic energy as possible and transform it into a pressure head sufficient to allow condensation of the vapor at condenser temperature. Once condensed, a portion of the refrigerant is conducted to the evaporator via a throttling valve to complete the refrigeration cycle. The remaining portion of the condensed refrigerant is pumped to the vapor generator to complete the thermal-engine cycle.

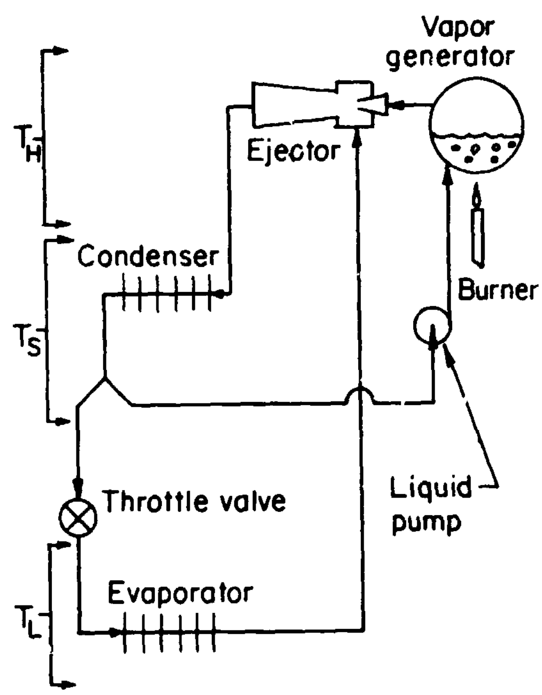

FIGURE 4. SCHEMATIC DIAGRAM OF THE EJECTOR REFRIGERATION CYCLE

The large amount of turbulence present in an ejector makes the device extremely irreversible and therefore provides a very inefficient way to compress the refrigerant vapor. Because of this extreme inefficiency, the ejector cycle has not yet found wide application.

Double-Loop Cycle. A double-loop refrigeration cycle is simply a mechanical vapor-compression refrigeration cycle driven by a Rankine-cycle thermal engine. A schematic sketch of a double-loop arrangement is shown in Figure 5.

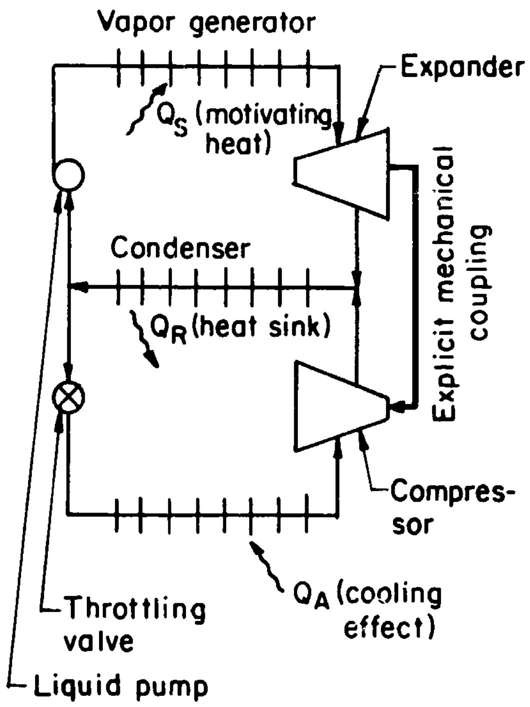

FIGURE 5. SCHEMATIC DIAGRAM OF A DOUBLE-LOOP REFRIGERATION CYCLE

While the double-loop cycle does not present quite as subtle a coupling of thermal engine and refrigerator as does the absorption or ejector cycles, it is included in the discussion of implicitly driven cycles because it does receive its motivating energy in the same manner, i.e., through a heat exchanger. The explicit separation of thermal engine and refrigerator permits the engine section of a double-loop cycle to absorb its motivating heat at as high a temperature as is practical. This feature provides the double-loop cycle with a definite thermodynamic advantage over the absorption cycle.

While the double-loop cycle offers an opportunity to improve on the efficiency of direct heat-actuated refrigeration cycles, it has not found wide application. The reason for this, when compared with an absorption-cycle system, is that its gain in efficiency normally does not justify the added complications introduced by the mechanical-expansion and- compression devices.

Less Conventional Cycles and Devices

The following discussion relates to possible new refrigeration techniques being considered for extravehicular space-suit thermal conditioning. These techniques can be classified either as cycles (both explicitly and implicitly driven) or as systems. The succeeding discussion is directed primarily to their competitive position at the present time. Further evaluation and anslysis will be necessary in order to determine which, if any, of these techniques hold the potential for future development.

Azeotropic Cycle. In the three-fluid-type absorption refrigeration cycle previously discussed, an inactive gas was employed in the evaporator-absorber section to depress the equilibrium vapor pressure and, therefore, the vaporization temperature of the refrigerant, while allowing the entire system to operate at essentially the same total pressure. The use of an azeotropic solution permits a similar arrangement.

The boiling temperature of the azeotropic composition of a lower boiling-point binary solution is below that of either pure component for a given total pressure. In other words, the presence of the one liquid depresses the boiling point of the other, and vice versa. If one liquid is now considered as the refrigerant working fluid, then the other assumes the same function as the inactive gas in a normal three-fluid-type absorption system. To elaborate somewhat, the azeotropic solution is placed in the evaporator, in which a portion of it vaporizes until it comes into equilibrium with the vapors of its two components. The evaporator is connected to an absorber that circulates a solvent having a preferential affinity for only one of the components, i.e., the one considered to be the refrigerant working fluid. Once absorbed into the liquid phase and removed from the presence of the inactive component, the refrigerant can be desorbed by the addition of heat and condensed in its pure state at a higher temperature than exists in the evaporator. The liquid refrigerant is then returned to the evaporator via a liquid trap to maintain the azeotropic composition and to complete the refrigeration portion of the cycle. The absorbing solution follows the same procedure as in a normal absorption cycle.

Liquid-Phase Working Fluids. The use of a liquid as the working fluid in a thermodynamic cycle may offer some interesting possibilities for rejecting heat in space suits. Although from a purely thermodynamic point of view, there appears to be no advantage to be gained. In actual practice the liquid may offer some practical advantages over a gaseous working fluid.

A regenerative thermodynamic cycle would appear to offer the most opportunity for the application of a liquid working fluid. The actual machine could quite possible be designed on the order of the displacer-piston type employed by previous developers during their work on Stirling-cycle engines and refrigerators. One property that might prove to be an advantage is the usually higher coefficient of thermal conductivity of liquids as compared with that of gases. This, of course, would be an aid to heat transfer.

It is emphasized that the comparative advantages and disadvantages of liquid and gaseous working fluids have not yet been evaluated, and the use of liquids for refrigeration cycles is mentioned here with the hope that someone may recognize an advantage and somehow put it to good use.

Fog Cycle. By operating a cycle totally within the quality region of a condensable working fluid, it is ideally possible to achieve Carnot-cycle efficiencies and therefore gain at least a theoretical advantage over the normal vapor-compression cycle. However, it remains to be seen whether this theoretical advantage continues to aid the cycle when irreversibilities are considered. The fog-cycle concept is an interesting one and certainly appears to warrant further consideration as a means for rejecting heat in extravehicular space suits.

Heat Pipe. Possibly no single other thermal-control concept has generated as much interest as has the heat pipe in the past 5 years. The passive nature of the heat pipe, in conjunction with its relatively high heat capacity, has placed this device high on the list of candidates for any space-oriented thermal control application. Considering the current interest in and the research activity centered around the heat pipe, it is only natural that this approach to thermal control be applied to space-suit thermal conditioning.

There appear to be possible configurational advantages associated with the use of the heat pipe for heat transport in a space suit. For example, it may be possible to utilize the external surface of the suit as a radiator for waste-heat rejection, with the space-suit wall acting as a variable-conductance heat-transfer device. This controlled thermal conductance could, of course, be effected by incorporating heat-pipe devices as an integral part of the suit shell. Theoretical and experimental studies have shown this approach to be feasible, and there is indication that the fabrication techniques that are needed to reduce this to practice are presently being pursued.

The heat pipe can also be considered a a possible link in a heat-rejection scheme that is based primarily on some form of powered mechanical-refrigeration system. For example, a system such as space suit-to-liquid loop-to-Stirling refrigerator-to heat pipe-to radiator appears to be potentially an efficient and versatile means for rejecting heat in a space suit.

Vortex-Tube Cycle. Of all the ways known for producing a decrease in temperature, few if any have received the range of interest that has been accorded the vortex tube (also known as the Hilsch tube, Ranque tube, Ranque-Hilsch tube, vortex refrigerator, T-tube, or separator tube). It was invented and patented in the early 1930's by M. G. Ranque, a French metallurgist, and brought to the attention of scientists in this country by Rudolph Hilsch in 1947.

Although several types of vortex tubes exist, the counterflow design shown in Figure 6 is the most prevalent. The basic idea behind all types, however, is to establish a tangential velocity in the gas. In the case of the counterflow type, this is accomplished by an inlet nozzle that enters the main tube tangentially at the outer perimeter. The remainder of the device consists of an orifice plate located directly to one side of the nozzle and a throttling valve located somewhat downstream and to the opposite side of the nozzle.

The observed operation of this device for various positions of the throttle valve is as follows:

With the valve wide open, all of the gas that enters by way of the nozzle exits at the hot end, with no change in temperature. The formation of a vortex within the tube reduces the static pressure along the axis of the tube, inducing gas to be drawn from the cold end through the orifice and out of the throttle valve at the hot end, with still no appreciable temperature change. As the throttle valve is closed, the gas flow from the cold end first ceases, then reverses, direction. When this reverse occurs, it has been observed that the stagnation temperature of the gas leaving the hot end rises above it. For a fixed

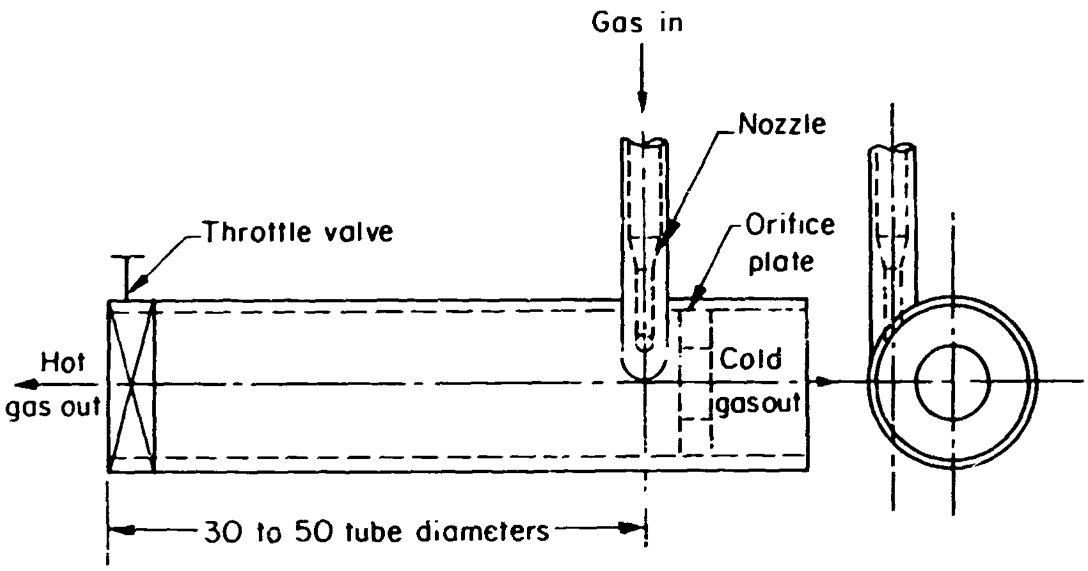

FIGURE 6. COUNTERFLOW-TYPE VORTEX TUBE

exhaust pressure and a given geometric configuration of the tube, the temperatures at the hot and cold exits vary with the percentage of the total gas flow leaving the cold exit and the stagnation pressure and temperature of the gas at the inlet nozzle. A comparison between the enthalpy increase at the hot end with the enthalpy decrease at the cold end shows that the two are numerically equal. The vortex tube is then a constant-enthalpy device that somehow exchanges energy between the resulting hot and cold gas streams.

The mechanism by which the vortex tube operates has never been explained to everyone's satisfaction. The most logical explanation, however, seems to center on the exchange of kinetic energy due to viscous shear. In brief, the gas initially entering the main tube establishes a vortex in which angular momentum is conserved, that is, the velocity times the radius of each particle is a constant. This is a velocity field where the tangential velocity increases with a decrease in radius. As the vortex moves axially along the tube, viscous forces tend to establish a constant angular velocity, or, in other words, the slower particles are accelerated at the expense of the faster ones. This kinetic-energy transfer from the faster particles along the inner portion of the tube to the slower particles along the outer perimeter of the tube tends to increase the temperature of the outer layers and lower that of the inner layers. The resulting temperature gradient then begins to reheat the inner layers, expanding the gas and forcing it out the central hole in the orifice plate. Naturally, it is held that the kinetic-energy transfer radially outward is greater than the heat flow radially inward.

A possible refrigeration system employing a vortex tube as the means for accomplishing a decrease in temperature is shown in Figure 7.

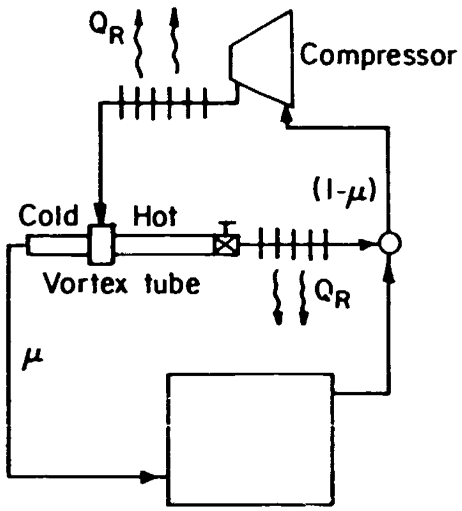

FIGURE 7. THE VORTEX TUBE AS APPLIED TO COOLING CYCLE

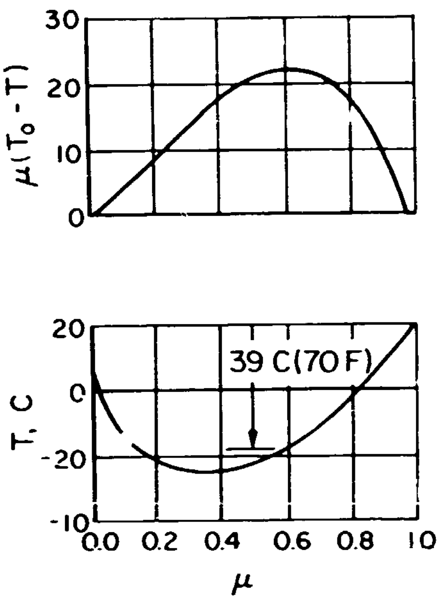

FIGURE 8. TEMPERATURE AND COOLING-CAPACITY FOR A VORTEX TUBE

Thus far, the vortex tube has not found application in air-conditioning practice, primarily because of its inefficiency. To demonstrate this, consider Hilsch's data for a tube operating at 6 atm delivery pressure, as shown in Figure 8. The lower curve is a plot of the temperature of the air leaving the cold end of the tube, as a function of the ratio of cold air flow to total air flow, μ. The upper curve is a plot of the product of μ and the temperature drop $T_o - T$. This curve shows that the maximum cooling capacity available does not occur at the lowest temperature attained, but at some slightly higher temperature at which the proportion of cold-air flow to total air flow is greater. In this case, maximum cooling occurs at $\mu = 0.6$. The temperature at $\mu = 0.6$ was -19 C, for a total temperature drop of 39 C (70 F).

As noted, the vortex-tube cycle, employing air as the working fluid, thus far has been found to be most inefficient, and the use of a gas other than air as the working fluid does not appear to offer any reason for altering this situation. For example, experiments have shown that carbon dioxide is only slightly more efficient than air, while hydrogen is somewhat less efficient. On the other hand, the static nature of the vortex tube should be appealing to the designer of space-suit thermal-conditioning systems and, should advances in the art bring about increased efficiencies, it is conceivable that in time this device may receive serious consideration for space-suit thermal conditioning.

Water-Vapor Electrolysis Cell. One of the primary goals to be pursued in developing space-suit thermal conditioning schemes is the possible utilization of the functional life-support-system components for rejecting heat. One device that appears to offer promise as a means for including the heat-rejection function as an integral part of the space-suit life-support loop is the water-vapor electrolysis cell. It is conceivable that this device can be used for the environmental control functions of dehumidification and heat rejection, in addition to supplying breathing oxygen.

The water-vapor electrolysis cell with phosphoric acid, shown schematically in Figure 9, is essentially a special type of oxygen generator designed to utilize directly the water vapor contained in air. As shown by Figure 9, water vapor in the air, flowing through the cell and across the anode, is absorbed in the electrolyte, and oxygen simultaneously generated at the anode is added to the air stream. Hydrogen collected at the cathode can either be used in a separate hydrogenation system for carbon dioxide reduction to water or vented overboard.

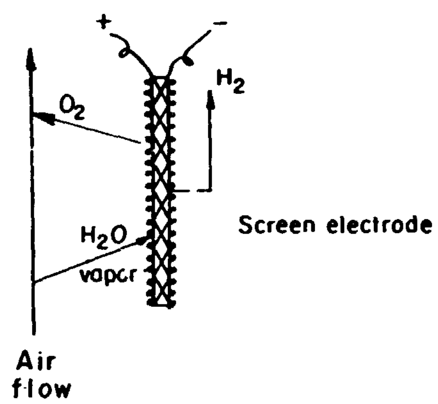

FIGURE 9. SCHEMATIC DIAGRAM OF "P_2O_5" WATER VAPOR ELECTROLYSIS CELL

A number of laboratory versions of the phosphoric acid water-vapor electrolysis cell have been built and tested at Battelle-Columbus for research on space-cabin life-support systems, sponsored by the National Aeronautics and Space Administration, Ames Research Center, under Contract NAS 2-2156. A typical laboratory module is capable of operating at any rate of oxygen generation up to a one-man rate (2 lb O_2/man-day) or higher. However, for an assumed power penalty of 300 lb/kw, the nominal design operation is at the 1/2-man rate (2/3 lb O_2/day) at the optimum current density of 20 amp/ft^2 for minimum system weight. The unit is designed for a nominal operating voltage of 28 volts for a battery of 12 cells in series (2.34 volts/cell average).

Although the primary purpose of the water-vapor electrolysis cell is currently oxygen generation, the unit is also a dehumidifier, since the water used for electrolysis is extracted from recirculated air. Therefore, it is easy to see how the water-vapor electrolysis-cell concept can be extended to space-suit thermal conditioning, considering that the metabolic-heat-generated water vapor can be continuously removed from the loop by electrolysis. In addition to providing a means for heat rejection, the water-vapor electrolysis cell would also generate supplemental oxygen for astronaut breathing. The water output from man by respiration and perspiration normally provides more than enough water vapor in the air for electrolysis to satisfy man's oxygen-consumption needs.

From a configurational standpoint, the water-vapor electrolysis cell offers some interesting possibilities. For example, it may be possible to construct the cell as an integral part of the space suit so that it is in close proximity with the astronaut's skin. This would permit direct diffusion of water vapor into the cell matrix, thus eliminating the need for a high rate of air circulation within the space-suit enclosure.

CONCLUDING REMARKS

Considering the intermediate status of the subject study, it would be premature to present conclusions at this time as to which thermal-conditioning technique or techniques will eventually emerge as having the greatest potential for use in future-generation extravehicular space suits. The concepts that have already been identified are subject to more extensive evaluation. Also, as with any conceptual study of this type, ideas are continually being generated, so it is virtually impossible to say at any given time that all potentially useful concepts have been identified. Many of the concepts that are found not to be workable, as is often times the case, serve to stimulate thought toward workable revisions or even totally new concepts.

It is hoped that this paper will generate sufficient interest in concepts that go beyond immediate hardware potential so that the normally slow process of advancing the state of the art by successive, incremental breakthrough can be appreciably speeded up.

ACKNOWLEDGMENT

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DEVELOPMENT OF THE PORTABLE ENVIRONMENTAL CONTROL SYSTEM

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SUMMARY: The Portable Environmental Control System (PECS) program began with development of two prototype units which were successfully subjected to systems testing. Test results, experience gained during the Gemini program, and analysis of early AAP mission requirements have justified the continuation of the program. This second phase of the development will resu in fabrication and verification testing of a prequalification prototype reflecting a substantial upgrading in performance capabilities and optimization for earth orbital or lu nissions.

INTRODUCTION

PECS is an example of an advanced technology program conducted for both immediate and long-term applications.

This program, which began in late 1964, was designed to advance the state of the art in portable life support equipment. Under the sponsorship of the NASA Office of Advanced Research and Technology, and the technical monitorship of the Crew Systems Division of the Manned Spacecraft Center, which was concurrently directing development of the Extravehicular Life Support System (ELSS) for the Gemini program, the PECS program, contracted to the AiResearch Manufacturing Company, A Division of The Garrett Corporation, soon achieved a good balance of advanced thinking, tempered or abetted by actual flight experience.

PECS program development, from its inception to its present position, is summarized in the PECS geneology shown in Figure 1. The new technology inputs developed within the framework of the program and supplied from contractor resources, as well as the experience gained from the Gemini EVA, are shown here in calendar relationship to the program.

New technology areas of special note were: (1) utilization of the brushless dc motor, (2) the development of a high-efficiency, magnetic gear, speed reduction concept, and (3) the study and development of solid oxygen supplies (sodium chlorate candles) for extravehicular applications. The results of the Gemini EVA experience became available late in the first phase of the PECS program and were a major influence in problem definition for

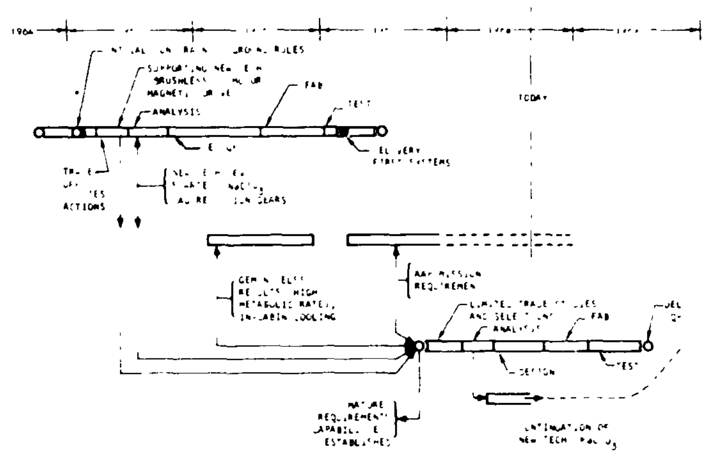


Figure 1. PECS Geneology

subsequent development underway today, primarily in the requirement for greater heat rejection (3500 Btu/hr maximum; 2000 Btu/hr average) and for maintenance of a comfortable suit environment for the EVA-ready astronaut in the pressurized spacecraft cabin.

Additionally, the experience gained by both NASA and AiResearch during the Gemini EVA program was used in the more work-a-day approach to the PECS design philosophy and packaging details.

PROTOTYPE PECS SYSTEM DESCRIPTION

The PECS consists of two principal fluid circulation systems; i.e., water for heat rejection and oxygen for respiration, suit pressurization, and carbon dioxide control. The PECS prototype (Figures 2 and 3) water loop consisted of a pump magnetically coupled to the fan motor, evaporator, evaporant reservoir, and an accumulator, all of which completed a closed loop with a liquid-cooled garment. In addition to transporting the metabolic heat from the liquid-cooled garment, the liquid loop cooled and condensed moisture from the oxygen loop and maintained a uniform temperature for gas produced by the sodium chlorate ($NaClO_3$) oxygen generator.

The oxygen loop included the fan driven by a brushless, photoelectrically commutated dc motor, emergency ejector pump, suit pressure control, CO_2 and odor control bed, condenser, and condensate reservoir. Primary and emergency oxygen were supplied from a $NaClO_3$ oxygen generator system which bled oxygen at a fixed rate directly into the oxygen circulation loop. Power and oxygen could be supplied alternatively by umbilical. A fully serviced prototype weighs 69.4 lb with a volume of 1450 cu in. Complete recharge requires five $NaClO_3$ candle/igniter assemblies, water, LiOH and canister, and battery recharge for a total of 20 lb. A hand-held control unit, shown in Figure 4, provided system control and status monitoring.

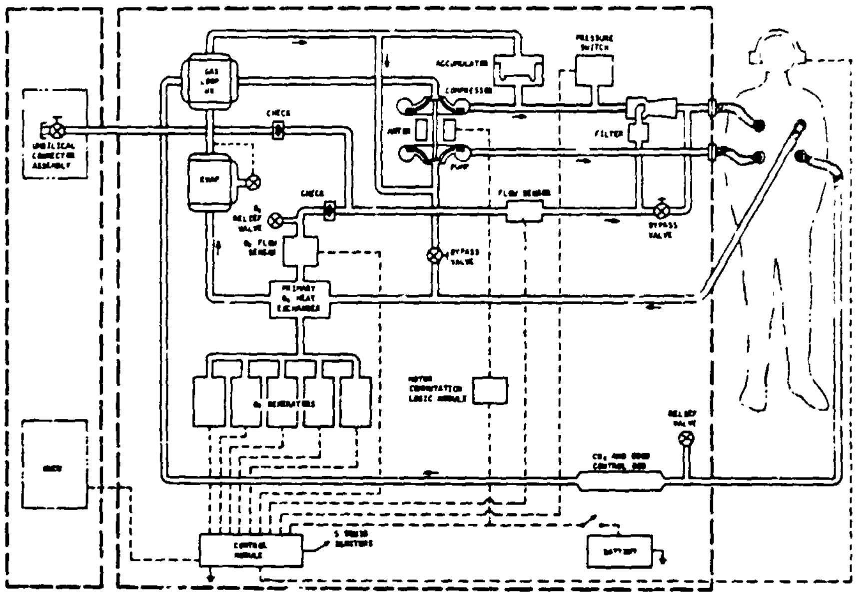


Figure 2. Portable Environmental Control System Schematic

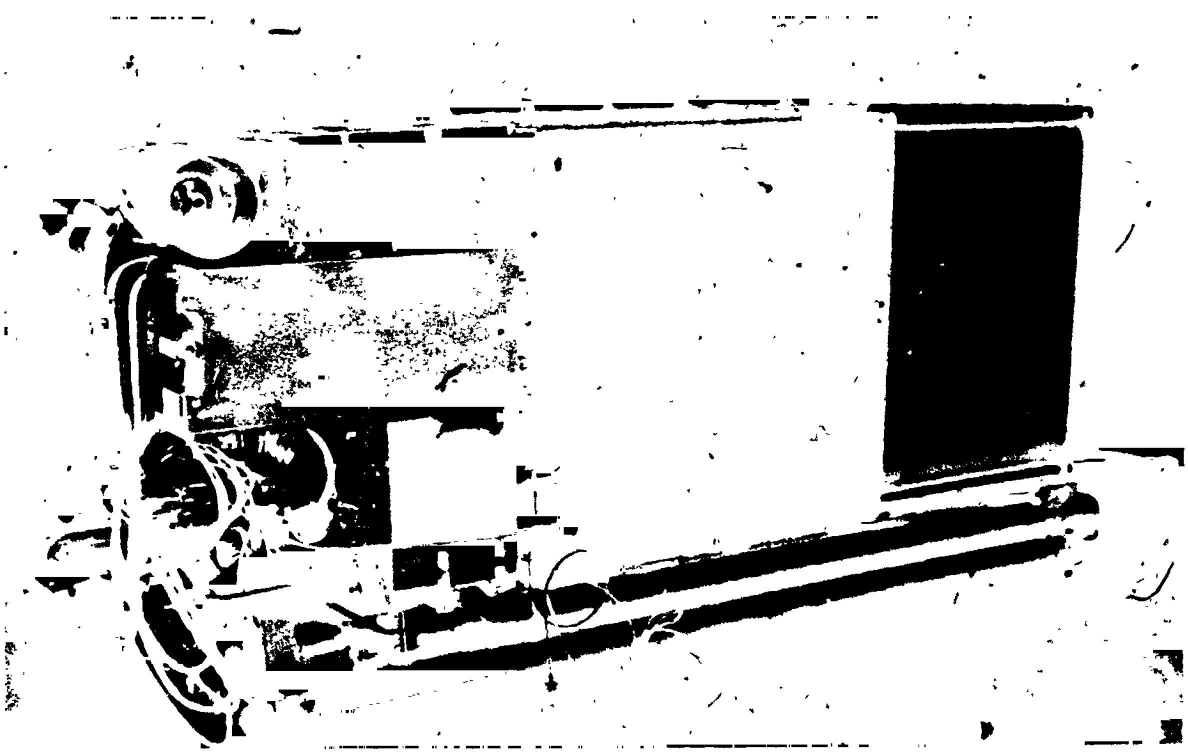


Figure 3. PECS Prototype Backpack

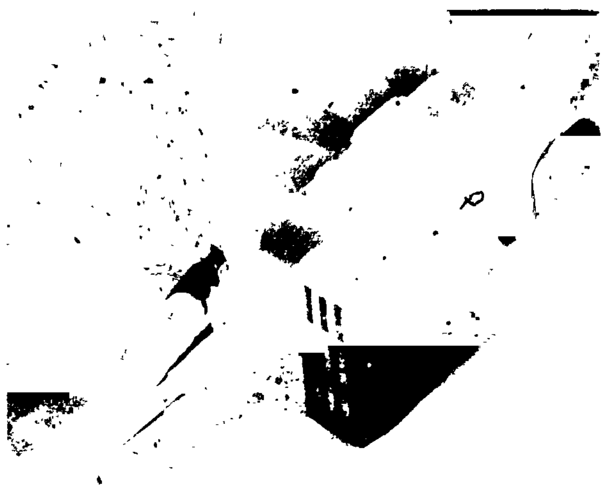


Figure 4. Hand-Held Control Unit

Initial Performance Requirements

With certain exceptions, the performance requirements for the initial PLCS development were based on criteria established for Apollo extravehicular equipment design. These requirements are as follows:

Mission time	4 hr
Heat load	
Metabolic average	1500 Btu/hr
Metabolic peak	2000 Btu/hr
Environmental	+250 Btu/hr
Ventilation flow rate	3.5 ACFM
Liquid loop (H_2O) flow rate	140 lb/hr
Coolant temperature at suit inlet	50 $\pm 5^{\circ}$F
PCO_2 (first 2.5 hr)	7.6 mm Hg
(last 1.5 hr)	10.5 mm Hg
Emergency oxygen (nominal)	30 min

PREQUALIFICATION PLCS

Gemini EVA Background Experience

Experience gained during Gemini extravehicular activities and supporting tests showed early estimates of metabolic energy expenditures of 1400 Btu/hr (nominal) and 2000 Btu/hr (peak) to be quite low. Heart rate measurements correlated to ground simulations data indicate average energy expenditures of approximately 2000 Btu/hr, with peaks up to 3500 Btu/hr.

Another problem resulted because the Gemini Extravehicular Life Support System (ELSS) rejected heat by means of an evaporator-condenser that required exposure to a vacuum for operation. During the final stages of extravehicular preparations prior to cabin decompression, the spacecraft ECS hoses were removed from the suit, and crew members were left with little or no cooling. This was further complicated by certain tasks, difficult in nature, that the crew performed working against the constraints of a fully pressurized suit. Under these conditions, metabolic heat was stored by the crew until the evaporator-condenser became operative, i.e., was exposed to a vacuum.

Mission Considerations

Proposed AAP missions were analyzed to determine design requirements for future EVA life support equipment. This analysis considered the basic mission operations associated with the various combinations of spacecraft, airlock module/docking adapter, orbital labs, and planned scientific experiments. Also considered were the contingency operations of transfer of crewmen between undocked vehicles; reservice and activation of a dormant environmental control system; failure of a cabin to pressurize, and requirement to egress the parent vehicle, ingress an unpressured compartment, and perform necessary corrective action; retrieval of damaged or fouled EVA equipment; and initial checkout of vehicle environmental control systems.

It became apparent from these considerations that even greater use of the umbilical, combined with the self-sustained capability, should be made. Typical instances are:

1. System operation from umbilical-supplied expendables during pre-egress preparations to conserve self-contained expendables for EVA. If an EVA is to be conducted from a vehicle with a two-gas oxygen/nitrogen atmosphere, the crewman must prebreathe pure oxygen for approximately 45 minutes prior to cabin decompression. The oxygen for this period must be supplied as a purge flow which exceeds closed-loop makeup flow. Oxygen for this purpose would exceed the capacity of expendable contained in a life support pack, whereas an umbilical supply to the pack permits utilization of existing valving and pressure regulation components. Furthermore, use of an umbilical supply to the life support pack will preclude the need for a separate, umbilical-fed prebreathe mask, the more complicated suit purge equipment and procedures, and the complex transfer from a mask to the suit loop while maintaining an O_2 purge.

2. Pre-egress checkout of the EVA equipment has become more complex. During this period the metabolic heat generated by increased activity may be removed by an umbilical-supplied coolant, thereby, precluding the type of problems encountered during Gemini that would require cabin decompression before the life support system heat exchanger (evaporator or sublimator) could handle the metabolic heat load.

3. Missions conducted on umbilicals effect a substantial reduction in post-EVA system reservicing.

4. During umbilical missions, if the umbilical becomes entangled or is damaged, the self-contained expendables allow the crewman to disconnect from the umbilical to complete his mission. When

a pure umbilical system is used the emergency system is sized quite conservatively to allow little more than a return to the parent vehicle. If the umbilical is damaged or fails, the astronaut could complete his scheduled mission or at least return the umbilical for inspection and repair while relying on the system's self-contained expendables.

5. With an umbilical, the system may be operated to provide full cooling, ventilation, etc., in a pressurized cabin, such as a workshop, where it is desirable to conduct zero g experiments that require simulation of actual EVA, i.e., pressurized suit, etc. As previously indicated, an evaporator or sublimator could not be utilized for this type of activity.

6. Emergencies occurring during EVA's conducted without an umbilical may be reduced in severity by permitting the crewman to return to the vicinity of the spacecraft where he could attach an umbilical, thereby, increasing the extent of his expendables until he could ingress and repressurize the spacecraft cabin.

7. Although a self-contained system is larger, it will permit umbilical-free transfer from a tumbling vehicle to a rendezvousing vehicle. This system will surpass the performance capabilities and the life limits of a bail-out bottle type system that is employed as a backup to a pure umbilical system.

8. The self-contained capability may also reduce spacecraft modifications and duplication of systems in space station modules necessary to support umbilical EVA or contingency transfers between vehicles during emergencies. A space station may have a particular module for EVA support which provides for an umbilical system. Weight, volume, cost, etc., preclude using this same umbilical and its subsystems in all spacecraft that rendezvous with the space station.

Increased Performance Requirements

This evidence for increased performance capacity and flexibility was coupled with data from design verification testing of the PECS prototypes, and the second phase of the program began, based on substantially upgraded performance requirements, including:

	EVA	IVA
Mission duration, hr	4	4
Heat load		
Metabolic average, Btu/hr	2000	2000
Metabolic sustained, Btu/hr	2500	2500
Metabolic peak (10-min duration), Btu/hr	3500	3500

Environmental, Btu/hr	+250	+250
Ventilation flow rate, ACFM	7	7
Liquid loop (H_2O flow rate), lb/hr	275	275
Coolant temperature of suit inlet, °F	41 +4/-1	41 +4/-1*
PCO_2 (normal and sustained), mm Hg	0.5	0.5
PCO_2 (peak and emergency), mm Hg	2.0	2.0
Emergency, min	30	30

*Based on umbilical water inlet temperature of 37°F

Prequalification PECS System Description

Using these requirements, the PECS was reanalyzed and the following changes were established for continuation of system development for a prequalification prototype (Figures 5 and 6):

a. A full umbilical operational capability was provided, i.e., H_2O, O_2, power, communications, and biomedical telemetry

b. Provision for a redundant H_2O pump and motor actuated by a loss of pressure head across the primary pump was incorporated.

c. The ejector pump, which previously had been in series with the fan, was relocated parallel to the fan to reduce pressure losses across an inoperative fan. The ejector is automatically actuated as a function of loss or reduced ventilation flow rate.

d. A dual-channel RF transceiver with multiplex TM capability was incorporated.

e. A separate emergency battery was provided to power the redundant pump, the secondary transceiver mode, and controls and displays.

f. Because of the extreme difference in oxygen requirements during nominal operation and peak activity levels, the $NaClO_3$ oxygen generator imposed a volume and weight penalty. Therefore, this system was replaced with a 7500-psig oxygen supply system composed of separate but interchangeable emergency oxygen storage and regulation modules. Oxygen resupply is done by module replacement or, where the facility exists, by direct recharge.

g. The biomedical and system performance parameters were increased to include the following:

1. Suit pressure

2. PECS outlet PCO_2

3. Primary oxygen pressure

4. Emergency oxygen pressure
5. Battery voltage
6. Electrocardiogram
7. Impedance pneumograph
8. Low ventilation gas flow
9. Auxiliary pump activation
10. Battery current
11. High primary O_2 flow
12. LCG inlet H_2O temperature
13. LCG Delta T
14. Condenser gas outlet temperature
15. Feedwater quantity
16. Oxygen partial pressure

PQP weight and volume are estimated at 102 lb and 3200 cu in., respectively.

Reservicing the PQP expendables includes replacement of the primary oxygen modules, water refill, replacement of the LiOH bed, and battery recharge for an estimated resupply weight of 36 lb. This does not include replacement of the emergency O_2 module.

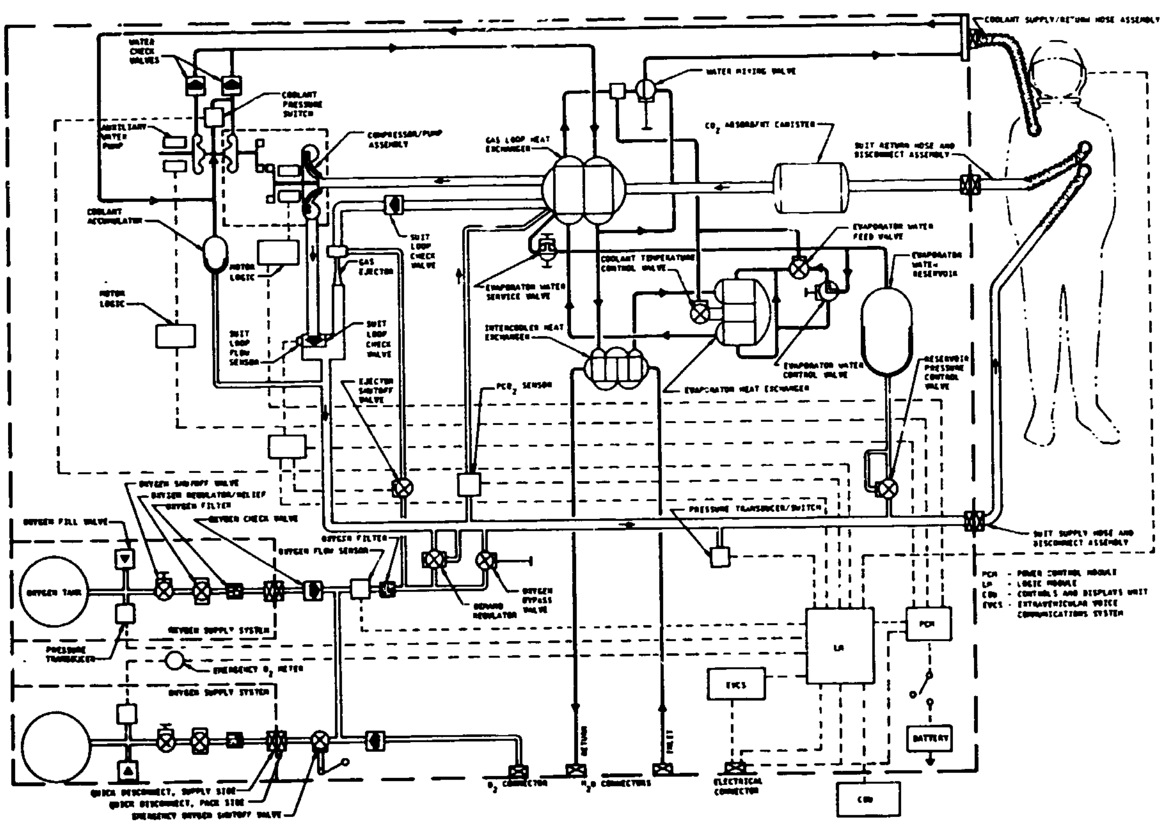


Figure 5. Prequalification PECS Schematic

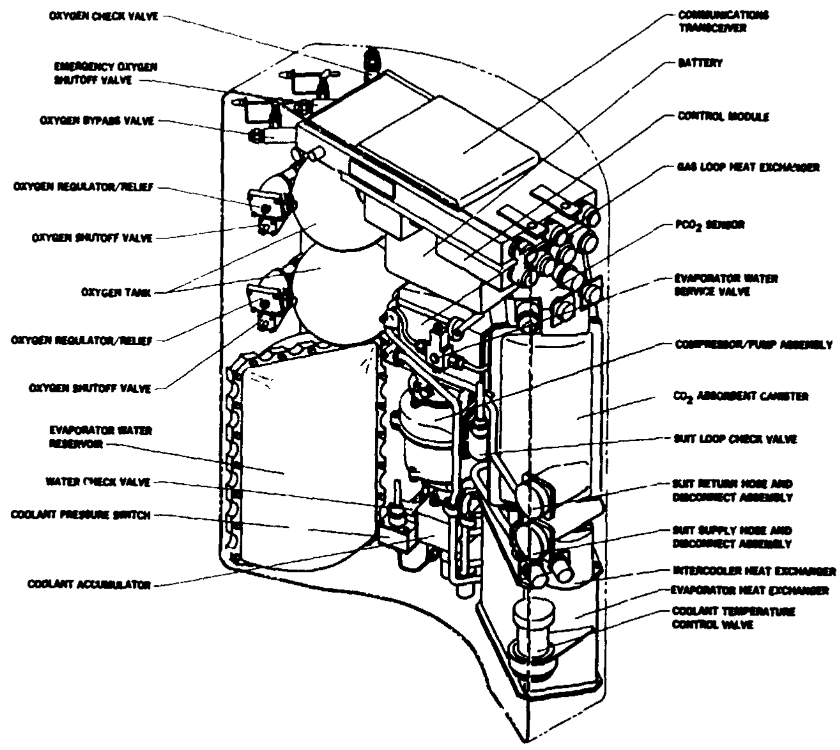


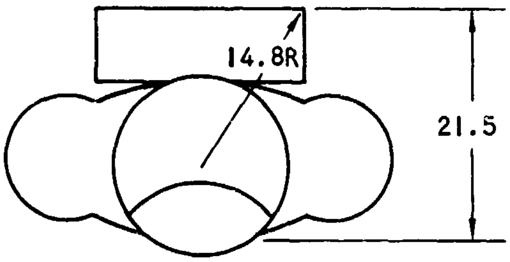
Figure 6. Portable Environmental Control System

Packaging

The prototype PECS system incorporated much of the same design philosophy that was used in the Gemini EVA program, i.e., wherever possible, components and subsystems were shaped to fit the constraints of a somewhat arbitrarily selected space. Iterations were made of the entire pack until the maximum packaging density was achieved. Weight, accessibility, and maintainability were secondary to the minimization of volume.

For the prequalification PECS system packaging, a modification to this approach was considered. The narrow concept of minimum pack volume per se was broadened to consideration of minimizing the critical dimensions of the man-pack system.

The critical man-pack dimensions considered were the distance from the front of the crewman's suit to the back face of the pack and man-center to pack corner. The comparative results of this approach are shown in Figure 7. Profile (a) shows these dimensions for the prototype PECS and profile (b) for the prequalification PECS; the front-to-back dimensional increase is 1.3 in. despite the fact that the total heat load is increased 33 percent and the maximum instantaneous heat rejection rate was increased 75 percent. Moreover, the man-center to pack corner shows a decrease of 0.8 in.



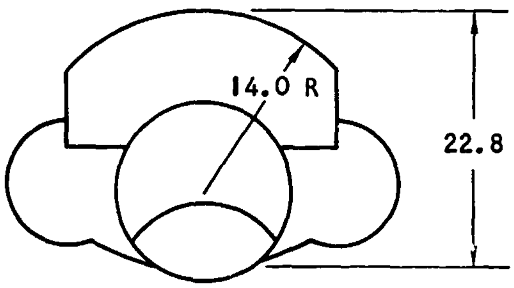


Figure 7. Combined PECS-PGA Envelope

Heat Transfer Subsystem

As discussed earlier, one of the design goals of the prequalification PECS was increased flexibility and conservation of spacecraft expendables. These goals are accommodated in the heat transfer subsystem, primarily by the addition of the intercooler, which provides (1) in-cabin cooling for the EVA ready crewman, before egress to space vacuum, where the heat is then rejected to an evaporative heat sink; and (2) the utility of the spacecraft heat sink (radiators) on umbilical missions, thus reducing the need for water (an expendable) as a heat sink.

Referencing Figure 5, the heat transfer subsystem consists of:

a. Intercooler

b. Evaporative heat sink

c. Gas loop heat exchanger

d. Water reservoir

a. Intercooler

As stated, the intercooler adds to the system flexibility in that the suited EVA crewman now has cooling capability during in-cabin check-out and EVA preactivities. The concept is valuable since it eliminates the time constraint for uncooled preparations prior to the actual EVA mission itself.

b. Evaporator

The evaporator is wick-fed, with a plate and fin construction similar to that used on the Gemini and Apollo ECS. The evaporator control is a wax-powered valve system which controls the coolant loop temperature between the limits of 40^0F and 45^0F by means of water feed and evaporant pressure control. These temperature limits are maintained over the heat rejection range of 50 to 3700 Btu per hour. Higher liquid-cooled garment temperatures are selected by the astronaut by means of a mixing valve.

c. Gas Loop Heat Exchanger

The gas loop heat exchanger removes sensible and latent heat produced by the man metabolic process and by the chemical process of the carbon dioxide reaction with the LiOH bed. The water condensed in the latent heat removal process is subsequently fed to and re-evaporated in the system heat sink. For umbilical operation the condensate is stored in the water reservoir.

The gas loop heat exchanger is designed for a maximum heat rejection of 1756 Btu/hr.

d. Water Reservoir

The water reservoir stores evaporant water used during self-contained EVA missions and stores water condensed from the gas loop during missions where the intercooler is used. The reservoir has a window for verifying the adequacy of the fill procedure.

Control and Display Subsystem

The hand-held control unit pictured in Figure 4, developed as part of the PECS prototype, was evaluated by NASA astronauts and human factors personnel. These displays indicate system status by way of backlighted windows with superimposed lettering identifying the particular parameter. The controls are either manual-action (linear or rotary motion) telefex cables or electrical toggles and momentary pushbuttons. Operation of the hand-held control unit relies on a fairly sophisticated electronic logic circuit which was developed only to breadboard status. The evaluation was conducted using subjects in pressurized EV suits; lighting intensities were varied to simulate day and night conditions.

These test results have been incorporated in the studies presently underway, to develop the prequalification PECS control and display unit. A chest-mounted unit is currently favored because of difficulties experienced in one-handed operation and the need for controls and displays visibility.

NEW TECHNOLOGY

A major consideration in the PECS program was the minimization of the use and/or volume of expendables. In support of this effort, two areas of technology were investigated and developed.

1. The development of the rotating component subsystem, which included a photoelectrically commutated, speed-controlled, brushless dc motor; gas compressor; magnetic speed reduction and coupling drive train and liquid pump. All elements, including supporting electronics, are in one package.

2. The development of a solid oxygen supply ($NaClO_3$) of low containment yield with an oxygen release rate matched to metabolic requirements. The estimates of metabolic rate variation, compiled from the Gemini program, coupled with data obtained from the PECS prototype program, indicate that sodium chlorate candles are not, at present, a competitive concept. However, the continuing development of high-mobility suits and increased understanding of efficient performance of EVA tasks should reduce the metabolic range variation to the extent that solid oxygen supplies are competitively superior.

Both of these developments are discussed below.

Rotary Component Subsystem

Minimizing the battery requirement (one of the four pack expendables) was the main motivation for adopting the photoelectrically commutated, brushless dc motor concept. In addition to the higher efficiencies (and smaller battery) obtained by the use of the brushless dc motor itself, additional reductions in power were achieved by incorporating a magnetic gear-reduction train. In this approach, one motor drives both the gas loop compressor and the liquid loop pump, each at their optimum efficiencies. The magnetic reduction gear system itself has an efficiency of 78 percent, in contrast to estimated efficiencies of 40 to 50 percent for conventional gear trains delivering the same power at similar speeds and reduction ratios.

Also, power is transmitted to the liquid loop pump by a magnetic coupling similar to that used on the Apollo EC/LSS water-glycol pumps. The magnetic coupling eliminates the requirement for dynamic shaft seals and, hence, results in a completely static sealed liquid system.

a. Trade Studies

Trade studies early in the PECS program determined quantitatively the weight and volume savings of the dc brushless motor, as compared to the conventional ac motor inverter system. These comparisons are summarized in Figures 8 and 9. For study purposes, separate motors were used for the compressor and pump; although the design point described is not that of the present PECS system, the trends are the same. On the basis of battery design specific energies (3.0 w-hr/cu in. and 35 w-hr/lb), the volume increase is 11.7 cu in./lb of weight increase. The penalty studies were based on the following flow rate and pressure drop conditions:

Water	250 lb/hr at 5 psi pressure rise
Air	5 cfm at 3-in. H_2O pressure rise

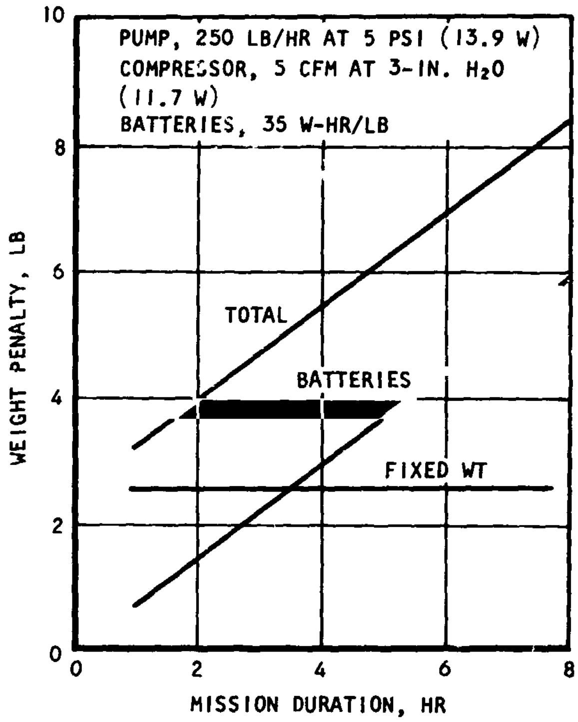


Figure 8. Electrical-Energy System Weight (AC Motors)

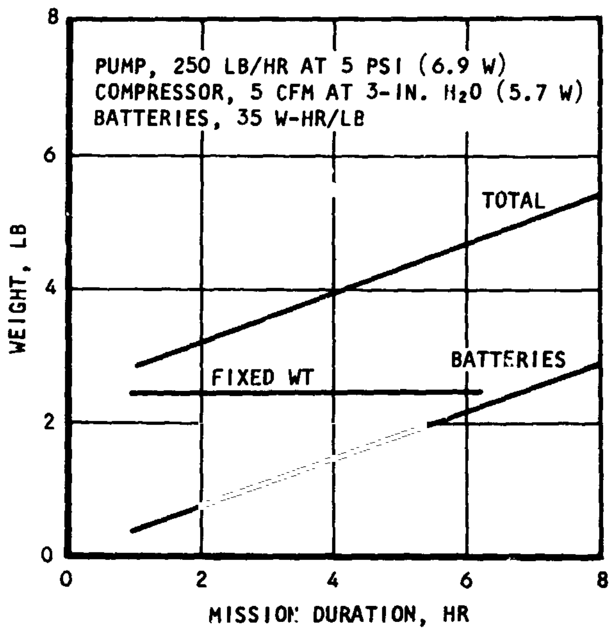


Figure 9. Electrical-Energy System Weight (Brushless DC Motors)

As shown in Figures 8 and 9, for an 8-hr mission, the brushless dc motor system is 3 lb lighter than the ac motor system. For repetitive missions the expendables difference is, of course, additive, and can amount to substantial weight savings.

b. The Brushless Dc Motor

The photoelectrically commutated brushless dc motor is a relatively new development that eliminates the usual dc motor brush problems. The function of the conventional brush commutator is duplicated by a solid-state electronic switching system that eliminates sliding contact brushes and the accompanying friction, arcing, and wear. As a result, the motor has a longer life, virtually no radio-frequency interference, and a high efficiency-to-weight ratio. A simplified schematic of the solid-state commutation circuits is presented in Figure 10.

For the PECS application, a brushless dc motor would weigh about the same as a comparable ac motor inverter and would occupy less space.

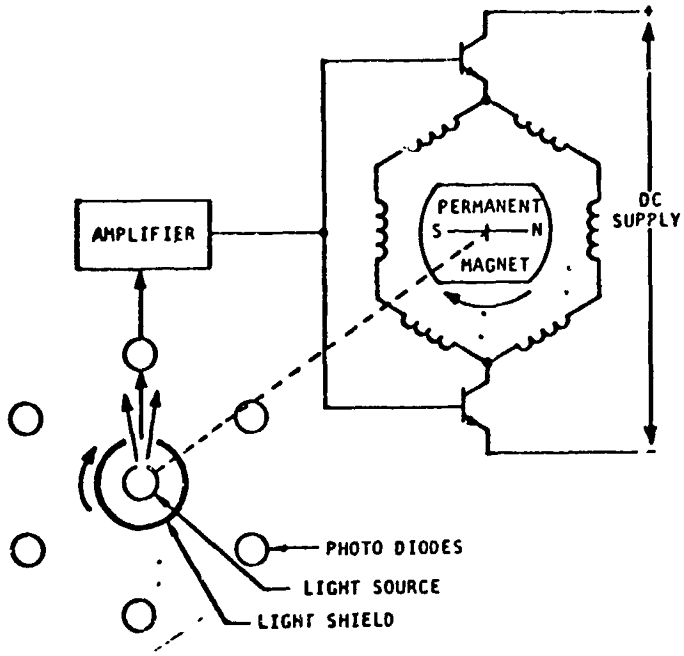


Figure 10. Simplified Solid-State Circuit Schematic, Brushless DC Motor with Photoelectric Commutation

In the brushless dc motor, the rotor is a permanent magnet field and the stator resembles a conventional armature winding without a mechanical commutator. A series of matched solid-state light emitters and detectors are mounted within the housing. A cup, which is rigidly attached to the rotor, rotates between the light emitter and detector. Holes in the cup allow specific detectors to be energized when the rotor field is aligned with a specific stator coil. The light detector signal is used to switch a power amplifier which excites the specific stator winding with rated direct-current voltage. The rotor magnet is air-stabilized and will not be demagnetized by transient or stalled current conditions. The ironless stator design allows more space for copper and accounts for the improvement in efficiency.

A cutaway view of the construction of a typical brushless dc motor is shown in Figure 11; the complete PECS machine, including the magnetic reduction gear and coupling, is shown in Figure 12. Figure 13 is a photograph of the compressor-motor-pump system used in the PECS prototype system.

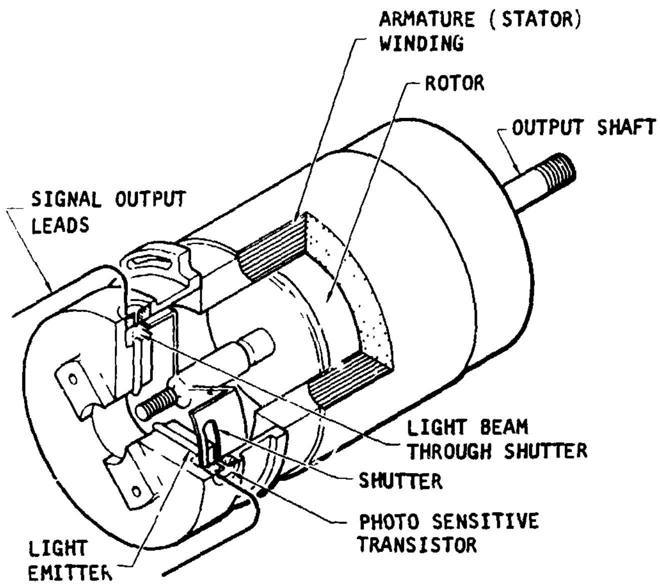


Figure 11. Brushless DC Motor

c. <u>Magnetic Reduction Gears and Coupling</u>

The magnetic reduction gears contribute to the overall machine efficiency. In addition to this increase in efficiency achieved by the gearing, the use of the magnetic clutch, which is similar in concept, provides a safety feature not available in direct, shaft-driven systems. Should the liquid pump seize, due to bearing failure or ingestion of foreign material into the pump impeller, the torque on the magnetic coupling will increase until it exceeds the design value (4.2 in.-oz). When this occurs, the magnetic drive will declutch and the resultant torque on the, motor-side, magnet

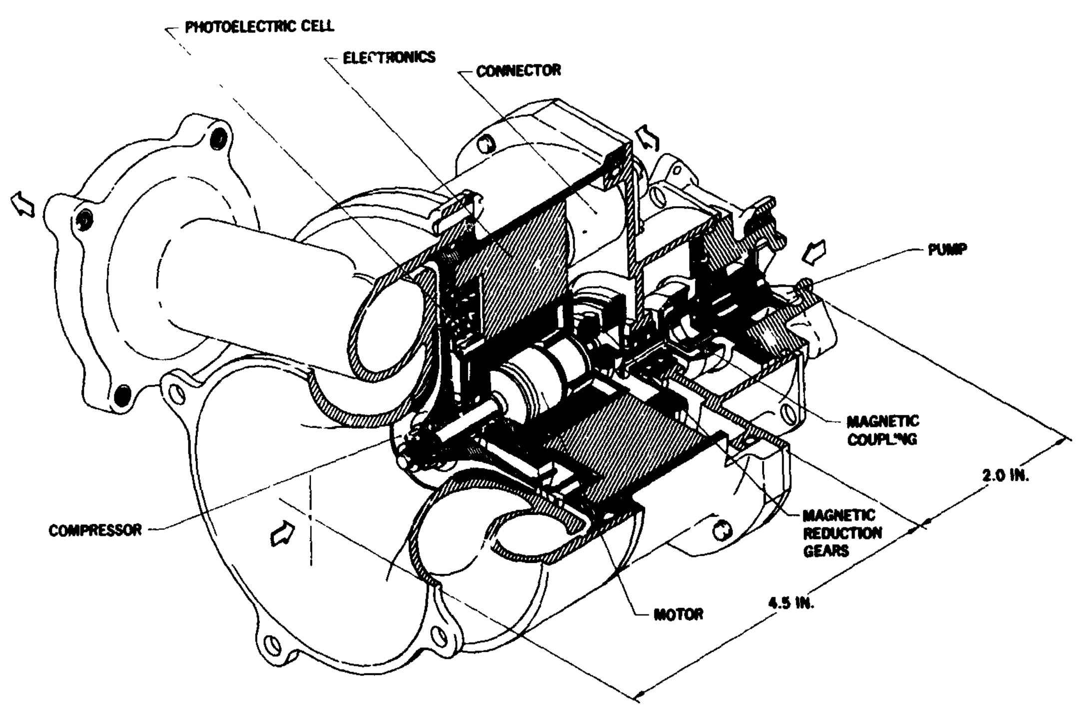


Figure 12. Complete PECS Machine Including Magnetic Reduction Gear and Coupling

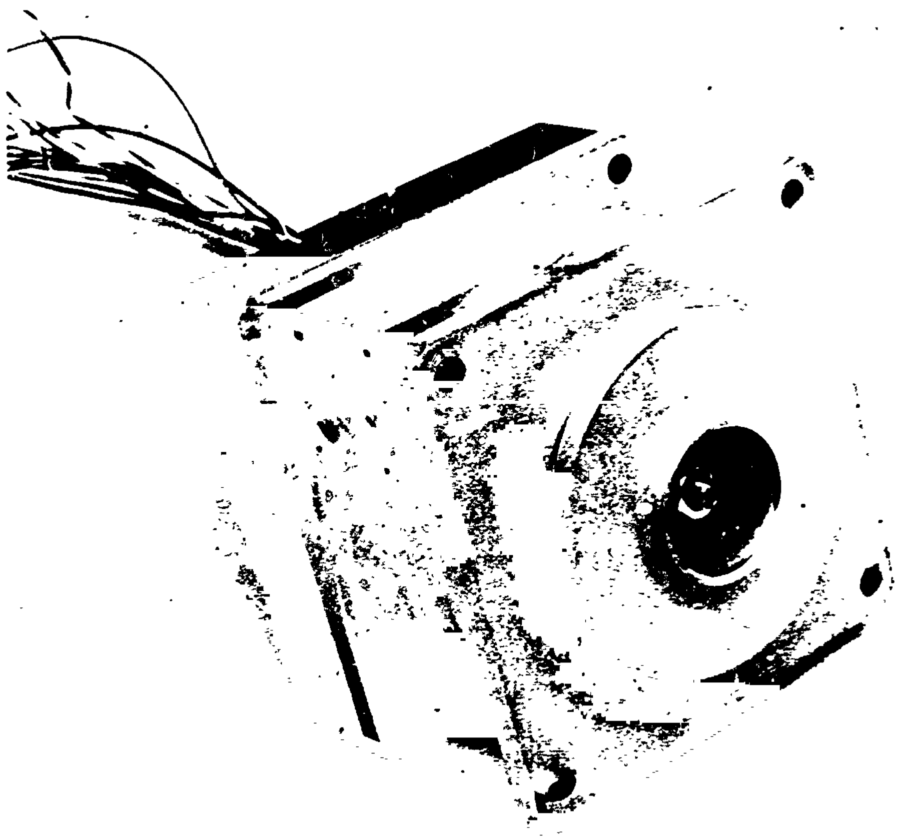


Figure 13. Compressor-Motor-Pump System Used in PECS Prototype System

will approach zero. Thus a failure in the liquid pump will not cause the motor and gas compressor to stop.

Magnetic gears are directly analogous to conventional gears with teeth; the difference is in load transmission. Where conventional gears transmit loads from tooth to tooth, by physical contact, the magnetic gear train transfers the load by magnetic field interaction of opposing magnet "gear" poles with no physical contact. The elimination of mechanical contact enhances the value of magnetic gears for this application. The difficult problem of lubricating gear teeth without contaminating the oxygen environment, is eliminated. The volume and weight advantages that accrue from minimizing power requirements by running both the pump and compressor near peak efficiency while using a single motor are realized without sacrifice of part of the advantage to cover the friction losses which results from driving a poorly lubricated gear train. The braking torque of the magnetic reduction gears has been designed for 2.1 in.-oz. The starting and running torques are 0.8 and 0.2 in.-oz, respectively. In case of a momentary overload that might cause breakaway, a pin cog mechanism has been provided in the reducing gears.

d. Performance of the PECS Compressor-Motor-Pump

The performance of the PECS Rotary Components Group is shown below. Although performance figures are given for a nominal 24 v, the performance of the machine can be expected to be substantially the same over the range of voltages shown. For example, the performance variation, in terms of efficiency vs rpm and shaft torque, is shown in Figure 14. The data were plotted from tests of the PECS prototype system.

PERFORMANCE
PREQUALIFICATION PECS
COMPRESSOR-MOTOR-PUMP-ASSEMBLY

Compressor, rpm	30,000
Pump, rpm	5000
Gas flow rate, cfm	7.0 at 3.7 psia
Gas pressure rise, in. H_2O	7.0
Water flow rate, lb/hr	275
Water pressure rise, psi	7.0
*Motor efficiency, percent	75
Compressor efficiency, percent	65
Magnetic gears efficiency, percent	78
Magnetic coupling efficiency, percent	90
Overall efficiency, percent	25.9
Voltage variation	17.6 to 30.0
Power, watts	30

*Includes bearing and windages losses

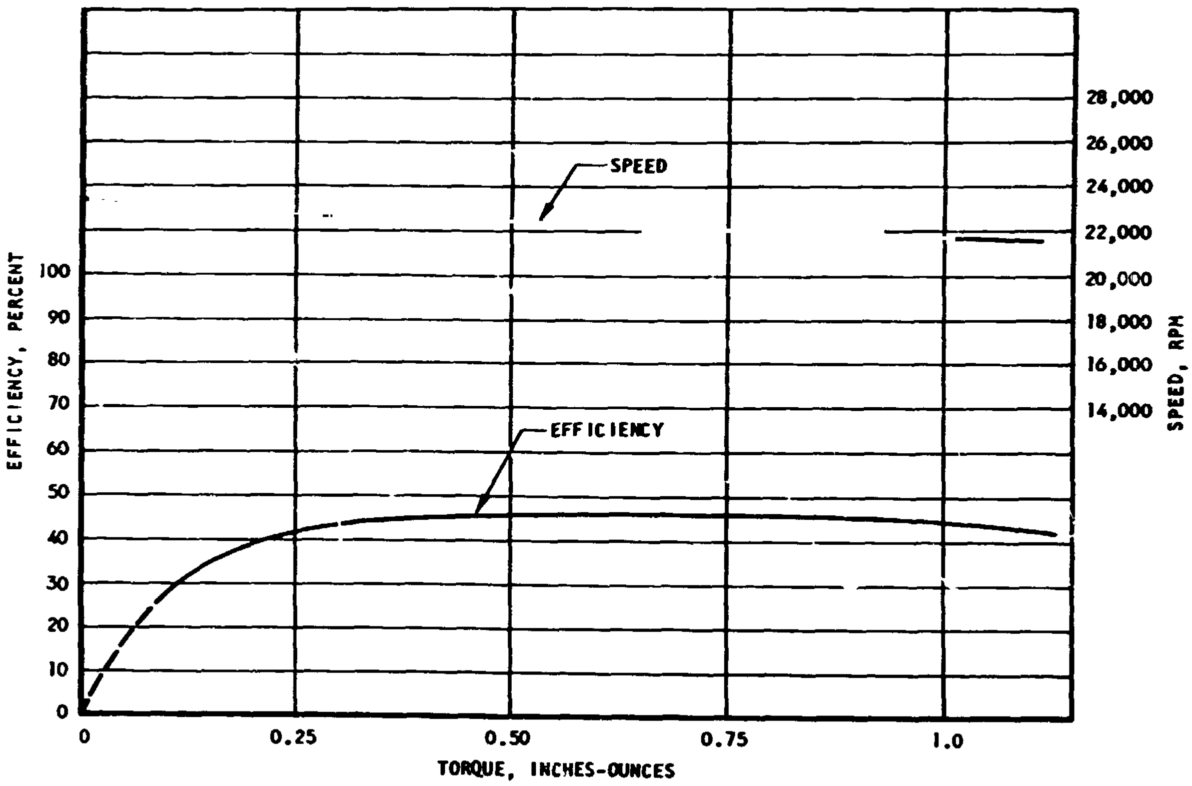


Figure 14. Brushless DC Motor Performance Curves

Sodium Chlorate Candles

The relatively high concentration of available oxygen contained by sodium chlorate has stimulated the development of the chlorate candle as a competitive source of breathing oxygen for use in restricted environments such as submarines, airplanes, and, most recently, space application (Table I).

TABLE I

OXYGEN SUPPLY CHARACTERISTICS

	$NaClO_3$	LOX	GO_2 7500 psi
Theoretical Performance			
Weight, lb O_2/lb source	0.451	1.0	1.0
Density, lb O_2/cu ft^3	70.1	71.3	36

Devices for generating oxygen by the pyrolysis of sodium chlorate have been described in the U. S. Patent literature as early as 1888. More recent studies on oxygen candles have been conducted under the sponsorship of the Naval Research Laboratory, Washington, D.C., the Aerospace Medical Research Laboratories, and presently by the NASA Manned Spacecraft Center, Houston, Texas.

Chemistry

Analysis by differential thermal analysis shows that conversion of chemical oxygen into molecular oxygen (Figure 15) occurs in the following sequence. Pure sodium chlorate melts at approximately 260°C (500°F) and appears as the endothermic reaction:

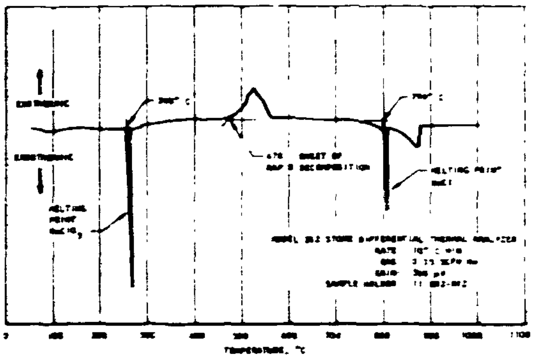


Figure 15. Differential Thermal Analyzer Date for Pure $NaClO_3$

$$NaClO_3 \text{ (Solid)} \rightarrow NaClO_3 \text{ (Liquid)}$$

The onset of rapid decomposition, which is exothermic, takes place at 478°C (892°F) and the overall reaction is described by the equation:

$$NaClO_3 \text{ (Liquid)} \rightarrow NaCl \text{ (Solid)} + \frac{3}{2} O_2 \text{ (gas)}$$

The endothermic reaction occurring at 799°C (1470°F) represents the fusion of sodium chloride:

$$NaCl \text{ (Solid)} \rightarrow NaCl \text{ (Liquid)}$$

The endothermic reaction following the fusion of sodium chloride has been shown to represent the volatilization of NaCl. This latter point is illustrated by thermal gravimetric tests (not shown) that show initiation of weight loss (i.e., the occurrence of submlimation) at temperatures slightly below the melting point of sodium chloride. Heat management, therefore, becomes a significant factor if aerosolization of the NaCl is to be avoided, i.e., the maximum temperature of the candle should be maintained below the melting point of the salt to avoid excessive production of NaCl vapors.

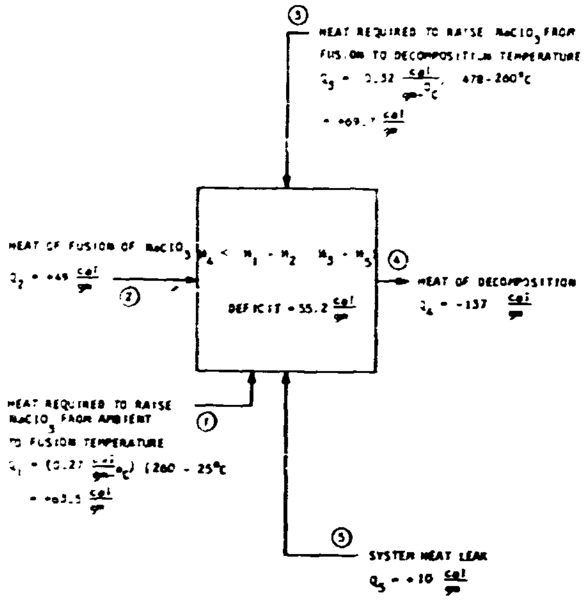


Figure 16. Thermal Balance on $NaClO_3$

Since the heat of decomposition of $NaClO_3$ is insufficient to sustain a thermal balance (Figure 16), additional heat (fuel) must be continally supplied to the system. For this reason, a powdered metal such as reduced iron, is usually mixed with the chlorate and, upon ignition, undergoes oxidation to produce the heat required to sustain a self-propagating reaction. Barium peroxide is also added to the mixture to serve as a catalyst and, in addition, can combine with any halogen evolved to further ensure the purity of the product oxygen. Fiber is usually incorporated into the mixture so that it may be compounded into a rod-like shape for (1) structural integrity and (2) physical retention of the burning front. A typical configuration for a chlorate candle oxygen generator is shown in Figure 17.

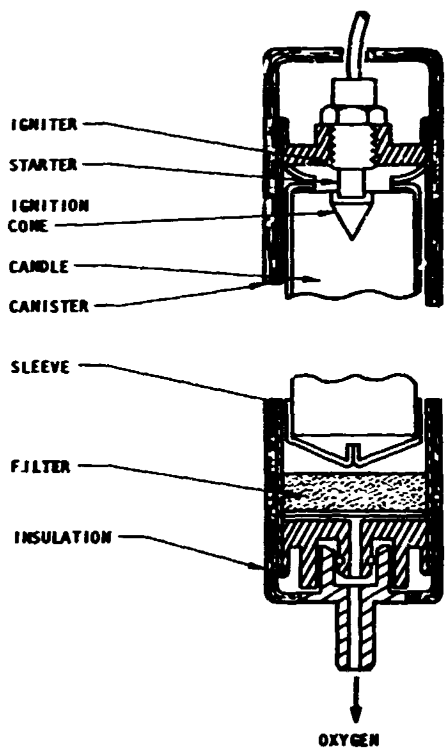


Figure 17. Chlorate Candle Assembly

a. Program Background

Investigations of the literature and an examination of existing hardware indicated that the oxygen generation requirements for the PECS would not be satisfied by existing state-of-the-art chlorate candle technology. For example, the sodium chlorate candle systems used onboard submarines have available surge and storage capacity to compensate for irregular candle burn rate, deficiencies in thermal management, and scrubbing equipment to remove contamination present in the evolved oxygen (Table 2). Further, the rate of oxygen evolution for the PECS candle should be designed to match the metabolic rate of one astronaut, which is an order of magnitude less than previous requirements.

TABLE 2

SODIUM CHLORATE CANDLE
PERFORMANCE SPECIFICATIONS

	U.S. Navy Submarine Specification	PECS Requirements
Burn duration, min	40 - 60	60 - 70
Maximum oxygen flow, lb/hr	5 x avg	0.631
Ignition match	No. 11 coated foundry nail	electrical
Oxygen purity, volume percent	*	99.98
Contaminants		
Water vapor, Mg/L	10	0.05
Chlorine, Ppm	10	0.1
Carbon monoxide, Ppm	25	5.0
Acetylene (C_2H_2), Ppm	*	0.02
Ethylene (C_2H_4), Ppm	*	0.2
Ethane (C_2H_6), Ppm	*	2.0
Nitrous Oxide (N_2O), Ppm	*	1.0
Halogenated compounds	*	0.1
Methane (CH_4), Ppm	*	25
C_3 and higher hydrocarbons (C_6H_{14} equivalent), Ppm	*	1.0
Carbon dioxide (CO_2), Ppm	*	5.0
Other, Ppm	*	0.1

*Not specified

b. Prototype Generator Studies

During the early stages of the PECS Program, formulations for the candle and ignition train and manufacturing methods were established. The effects of candle diameter, gas pressure, and thermal soak temperature on the linear burn rate were investigated to determine the optimum configuration for thermal control. A prototype five-barrel oxygen generator (Figure 18) with vacuum jacketing and radiation heat shields was designed, fabricated, and subjected to design verification testing. The logic system, shown schematically in Figure 19, automatically sequenced the candles in series fashion to provide a constant flow of metabolic oxygen for the mission. The control logic also provided protection against the contingencies of ignition or candle burn failure, by automatic progression and ignition of the next candle. One candle in the generator subsystem was designed to provide an emergency flow rate of 0.7 lb/hr and was manually ignited.

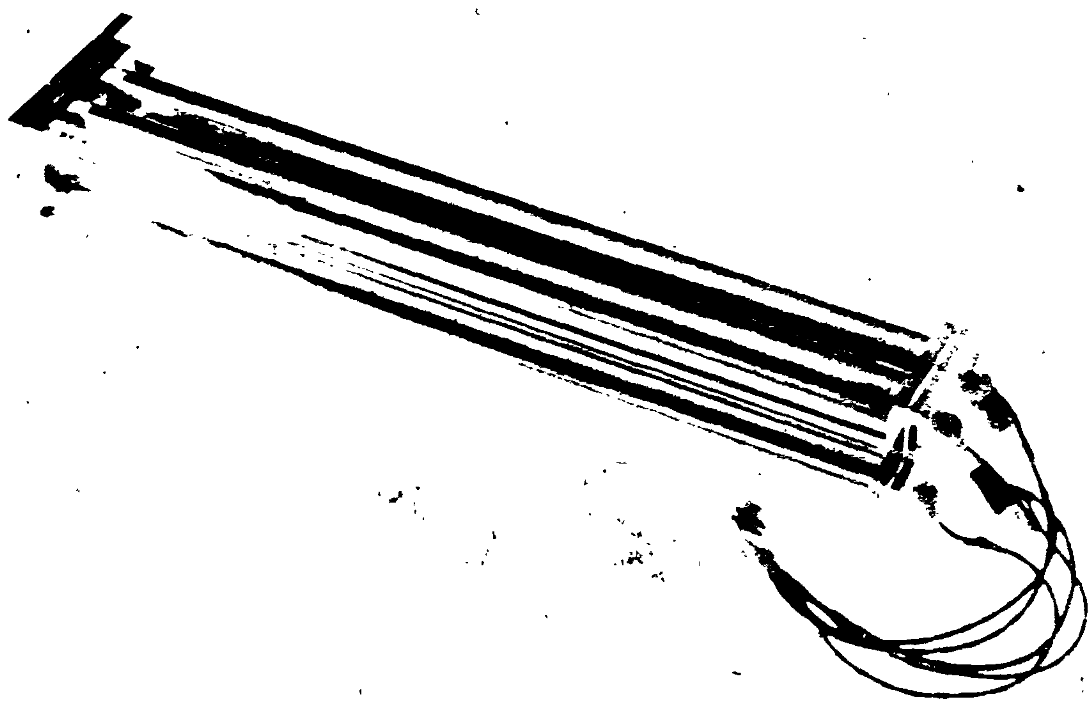


Figure 18. Prototype Five-Barrel Oxygen Generator

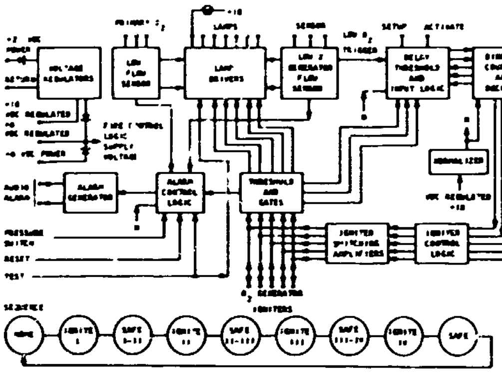


Figure 19. PECS Control Logic

One of the major goals of the prototype test was to achieve uniform oxygen production. Typical results, taken during the test series are shown in Figure 20. Oxygen outlet temperatures were in the range of 90 to 130°F. Clean ignition, however, proved a definite problem due to the use of commercially available electric matches, not designed for this application.

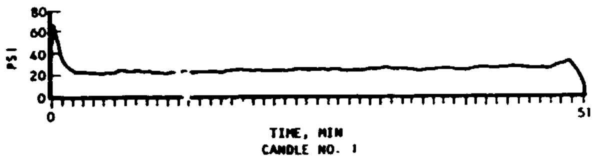


Figure 20. Pressure vs Time Curve for Oxygen Generator Verification Test

CURRENT STUDIES

Manufacturing

During the current prequalification PECS program, a review of the major candle fabrication processes was undertaken in an effort to (1) further increase the equivalent stored oxygen density and (2) increase the uniformity of the candles produced. (During the PECS prototype program the stored density was 52 lb per cu ft with a variation of ±5 percent.)

Techniques and equipment for compression molding, casting, and extrusion (Figure 21) were evaluated for their ability to reproduce a uniformly dense candle structure. Of the three processes examined, hot compression molding was selected as a first choice. This method was successfully employed in the laboratory to fabricate candles possessing a density in excess of 98.5 percent

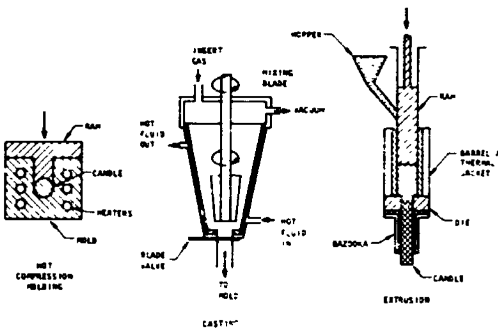


Figure 21. Oxygen Candle Processing Techniques

of the theoretical value for this formulation and compression strengths exceeding 10,000 psi. The theoretical oxygen density of the formulation used ($NaClO_3$ 86.5 percent; BaO_2 4 percent; Fe 3.5 percent; glass fiber 6 percent) is 59 lb/cu ft. This process was selected as superior to the casting process which had inherent problems of porosity and settling of the constituents. Preliminary tests utilizing the extrusion method show promise that this technique could supersede hot compression molding, because of its higher production rate capability and lower tooling costs. However, further development is required.

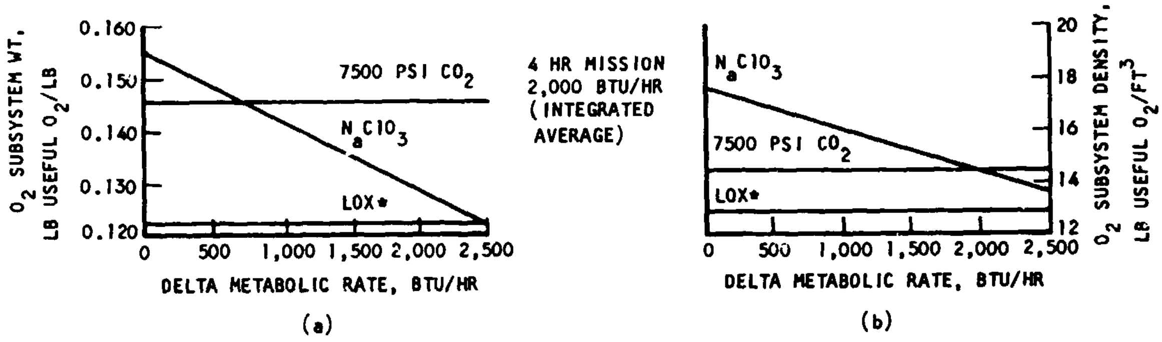
Ignition Train

The poor reliability and contamination problems experienced with the electric match ignition prompted a reexamination of the entire ignition system, i.e., the igniter, start pellet, and igniter cone. An examination of various electrical igniter products, showed that a charge composition, consisting of zirconium iron oxide and diatomaceous earth, produced the least amount of gaseous contaminants. Present efforts have also been directed to determine the feasibility of utilizing a canned igniter (pyrotechnic heater) to effect a single stage ignition system.

MISSION IMPACT ON OXYGEN SUPPLY CONCEPTS

The uniform oxygen production rate of the sodium chlorate candle is ideally suited to uses where a constant, or near constant, mass rate of supply of oxygen is required.

The volume advantage of the solid oxygen supply concept diminishes, however, in proportion to variation in oxygen flow rate required during a specific use cycle. The comparative weight and, more importantly, volume advantage of the chlorate candle subsystem is shown for the PECS in Figure 22, as a function of the change in the metabolic rate of the EVA crewman. The abscissa of the two curves (a) and (b) is given in terms of delta (or the variation in) metabolic rate. The absolute metabolic rate at the delta base point of zero, was taken as 500 Btu per hr.



*DOES NOT INCLUDE 9-POUND BATTERY

Figure 22. Parametric Comparison of Oxygen Requirements for PECS

Referencing curve (b) of Figure 22, the maximum metabolic rate variation considered in the prototype PECS program, was (2000-500) or 1500 Btu per hr. As shown, the candle concept still exhibited a volume advantage, even though the metabolic rate varied by a factor of four. The results of the Gemini EVA program indicated, however, that a metabolic rate variation by a factor of seven (3500-500) is possible; at least under then existing EVA technology. The prequalification PECS was therefore designed conservatively to meet this metabolic rate variation and, hence, the smaller (high pressure gas) oxygen supply concept was adopted.

In the PECS program the use of chemical oxygen suppiies has been confined to use on a back pack system. Decreases in the variation of the oxygen use rate are required to optimize the sodium chlorate generator for this type application. This is logical by improved space suit design (to reduce work expended in overcoming opposing suit forces) and by EVA task design, such as was successfully done on Gemini XII.

Solid oxygen supplies also have utility elsewhere in manned space exploration provided that, as in all concepts, the use is tailored to the capabilities of the concept in question. Typically, the chlorate candle concept is suited for applications having some or all of the following requirements:

a. High density (59 lb of oxygen per cu ft)

b. Indefinite, zero loss, standby (hard vacuum preferred)

c. Constant mass flow delivery at variable pressure (15 to 500 psi)

d. Programmed delivery variation (by varying candle cross section)

e. Oxygen storage space is irregularly shaped (if variable mass flow delivery is acceptable)

f. Uncontrolled temperature environment (-200 to +300°F)

CONCLUSIONS

1. The versatility of the self-contained Portable Environmental Control System has been enhanced through incorporation of umbilically supplied utilities for prebreathing, on-board cooling and near-vicinity EVA.

2. The two factors having the the greatest impact on system design are increasing work loads, based on previous EVA experience, and the incorporation of provisions for mission contingencies. Improvements such as the increased efficiency of a fan and pump driven by a single photoelectrically commutated motor and high packaging densities have worked to offset the above factors.

3. At present, a sodium chlorate oxygen generator is not the optimum oxygen supply concept from the standpoint of weight and volume for self-contained extravehicular astronaut life support systems, where there is a large differential between nominal and maximum oxygen consumption rates. Reductions in these differentials, through suit technology advances and/or the development of a means of varying oxygen production rates will, however, re-establish the $NaClO_3$ generator as a competitive concept for extravehicular life support systems.

4. The sodium chlorate candle, as presently developed, is applicable as an oxygen source where high-storage density storage, coupled with long standby in a hostile environment is required.

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FIRE RESISTANT SPACESUITS

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SUMMARY: Metal fabric has been efficiently joined by spot welding and suitably coated to provide impervious, fle ble composites for general application to inflatable high effic... pressure vessels. A very significant advantage of the coated metal fabrics for space applications requiring flexible structures is the retention of the structural integrity of th system in extremely high temperature environments.

The preliminary spacesuit study included the evaluation of several elastomeric sealant/metal fabric material combinations. A silicone rubber was selected as the elastomeric sealant displaying the best combination of required properties. Several flexible, cylindrical pressure vessels were fabricated from Karma fabric and were internally coated with the silicone rubber to simulate a spacesuit limb. The cylinders were pressure cycled to proof pressures over 200 times with no measureable change in leak-tight integrity. Also it was demonstrated that the cylinder could withstand a "burst" test at a pressure of 34.2 psig for over 15 minutes with no observable structural degradation or permanent deformation.

New materials systems and technologies have been investigated for solutions to the problems associated with high temperature and fire resistant flexible pressure vessels such as space suits. The material system utilizes a high temperature metal fabric, joined by a series of spot welds, and an impregnated or coated elastomeric sealant. The spot weld seams are very flexible and have consistently demonstrated joint efficiencies of 85-95 percent, even at temperatures over 900 degrees Farenheit.

These materials systems and technologies have general application to inflatable high efficiency, structural pressure vessels such as re-entry vehicles for the recovery of both manned and unmanned payloads and as blast and radiation resistant enclosures. Other applications include high speed. high "q" decelerators and re-entry drag devices, etc. A principal advantage of this technology for flexible and inflatable structures is the retention of the structural integrity of the system at temperatures well above even the melting point of "S" glass (1535 degrees Farenheit). Figure 1, a graph of normalized fabric tensile strengths vs. temperature, illustrates the significant advantage of metal fabrics for flexible structures which must operate in high temperature environments.

Metal fabrics have many distinctive advantages for these applications. In addition to their high temperature resistance, the thermal conductivity of the metal fabric assures dissipation of the heat from local hot spots, thereby reducing the temperature of surface and substrate materials. Other desireable characteristics include a high modulus, the electrical and mechanical properties associated with metallic materials, as well as excellent tear and abrasion resistance. Fold endurance, according to the MIT Tester, is about equivalent to Fiberglas. "S" glass fabrics, a primary competitor for many high temperature applications, are specifically not recommended for applications in which severe folding and packaging are followed by high temperature flexing; the metal fabrics, however, do appear to perform satisfactorily. In the annealed condition metal filaments have an elongation more than twice as great as Fiberglas, which permits the redistribution of high localized stresses. Disadvantages of metal fabrics include a lower strength to weight ratio at room temperature and some reduction in flexibility.

The selection of the particular impregnant/sealant used with the metal fabric to obtain a composite is obviously dependent upon the use for which the composite is intended. Several different elastomers were evaluated as sealants for an astronauts space suit and several simulated space suit limb segments were fabricated and tested during a recent program for NASA/MSC (NAS 9-7253).

The sealant evaluation was directed primarily toward the evaluation of flexibility, abrasion resistance and flame retardant properties of selected elastomeric materials. As flame temperatures typically are a minimum of about

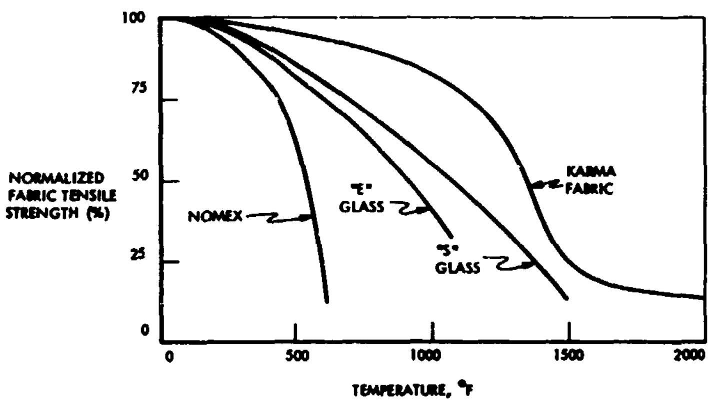

Figure 1. Normalized Fabric Tensile Strength vs. Temperature

Table 1

SEALANT MATERIAL TRADE-OFFS

Sealant	Trade-Off Parameters				
	Decomposition Temperature °F	Toxicity of Decomposition Products	Flammability in Oxygen	Relative Flexibility	Abrasion Resistance
1. Silicone (RTV-655)	750	Non-Toxic	Inflammable ([illegible]°F)	Good	Poor
2. Silicone (modified)(1)	900	Non-Toxic	Inflammable (750°F)	Good	Good
3. Fluorocarbon elastomers	600	Noxious-Toxic	Inflammable (600°F)	Fair	Good
4. Teflon (FEP)	750	Noxious-Toxic	Inflammable	Good	Excellent
5. Dexsil	500	Toxic	Probably inflammable	Fair	Good
6. Polyimides	1400	Noxious	Inflammable	Poor	Good
7. CNR Rubber	550	Noxious-Toxic	Not flammable	Good	Fair

(1) Provided by the R.E. Darling Co., Inc., Gaithersburg, Maryland.

1500 degrees Farenheit, there are no known elastomers (or non-metallic fabrics) that will not decompose or burn, if exposed to these temperatures for a sufficiently long period of time. Therefore, the fundamental problem was related to a trade-off between critical material characteristics, as illustrated by Table 1. In addition to the principal trade-off parameters listed in Table 1, other factors considered in the sealant material selection were oxygen permeability, rate of decomposition or flame propagation, heat transfer characteristics, impregnation or coating processing requirements, and availability. Composite systems of several sealant materials were also considered, such as an external coating of carboxy nitroso rubber (CNR) to provide flame resistance with an internal silicone pressure bladder. It was also considered probable that the flammability characteristics of the elastomer system would not be too significant as the fabric would serve as a very large heat sink with excellent heat transfer characteristics, thereby reducing the flame temperature below the propagation point.

The primary candidate sealants were silicones, CNR rubber, fluorocarbon rubbers, and polyimides. The CNR rubber and polyimides were not experimentally evaluated during the contract because of availability and/or processing limitations. Test coupons were fabricated from the modified silicone and from Viton B fluorocarbon rubber, using a 1.0 mil diameter filament nickel chromium or Karma alloy fabric. An external coating of FEP Teflon was laminated onto some of the test coupons. Stoll and Taber abrasion tests, a 180 degree bend flexure test, and flammability tests were performed.

Only the Teflon coated specimens were significantly superior in the abrasion tests. For example, in the Taber test (1000 gram weight, 70 rpm) a Viton/Teflon sample lost only 0.14 percent by weight after 5100 cycles while a Viton sample had lost 1.7 percent by weight after 2500 cycles. The flexure test showed that the parent metal fabric performed about the same as all of the coated samples.

In the standard flammability tests, where the specimens are held vertically above a specifically sized ignition source, all of the elastomer coating materials burned in a 16.5 psia oxygen atmosphere. None of the coating materials burned under the standard test conditions when a sufficiently large specimen was mounted horizontally above the ignition source, thereby reducing the influence of edge effects; however, the coating on the horizontally oriented samples was induced to burn when four pieces of facial tissue, instead of the standard two pieces, were used as the flame source. The metal fabric did not burn in any of the flammability tests. The results of some of the tests on horizontally mounted samples are presented in Table 2.

No significant difference between the silicone or the Viton sealants was evident as a result of the three types of tests discussed above. The silicone was therefore selected for further studies, primarily because it was more flexible than the Viton. A composite of Teflon on the external surface for abrasion resistance and silicone for sealing the inner surface of the metal fabric was selected as the most promising sealant material system.

Three test cylinders, representative of a flame resistant astronaut suit section, were fabricated in order to demonstrate the applicability of the materials system to a space suit. The demonstration and test articles were 7 inches in diameter and 24 inches long with 18 inches of unsupported metal fabric. The longitudinal seam in the 1.0 mil Karma fabric was made with a

Table 2

FLAMMABILITY TEST RESULTS(1)

Materials System(2)	Specimen Size (inches)	Burning Time (seconds)	Burning Characteristics and Comments
Viton	1.0x2.5x0.027	13	Edges ignited first. Burned with a yellow white flame.
Silicone	1.0x2.5x0.027	20	All of sample ignited rapidly. Burned with a white flame.
Viton	3.0x4.0x0.046	-	Did not burn.
Viton/Teflon	3.0x4.0x0.028	35	Used four pieces of facial tissue instead of two. Center of specimen burned through in about 18 seconds. Specimen then burned with a yellow flame from the center out.

(1) All tests were at 16.5 psia oxygen. Test procedure and equipment similar to MSC-A-D-66-3, Revision A, NASA/MSC, June, 1967, Test No. 1, except that the specimens were held horizontally, directly above the ignition source.

(2) All specimens were coated on one mil diameter filament Karma fabric.

Figure 2. Simulated Space Suit Limb

special spot welder, as discussed later. The fabric was circumferentially heliarc seam welded at each end between two 0.060 inch thick 300 series stainless steel sheets to produce a joint efficiency in excess of 95 percent of the parent fabric strength. Calendered silicone sheets 0.015 inch thick were applied to the inside of the cylinder and the elastomer was cured. Bondable FEP Teflon was also applied to a portion of the surface for comparative evaluation. Aluminum rings with "O" ring seals were bolted and sealed in each end to provide mounting rings for end closure plates. A photograph of one of the cylinders is shown in Figure 2.

The simulated space suit limbs had excellent structural integrity and flexibility. The circumference of one of the cylinders was measured during proof pressure tests and the diameter increased only 0.050 inches (0.07 percent) between 5.0 and 20.0 psig, and no hysteresis was observed after 200 cycles. Figure 3 is a photograph of this cylinder, at an internal pressure of 20 psig, during the cycle testing. No degradation was measured or observed, and the leak rate remained constant after the pressure cycling. Assuming zero leakage through the end seals and fittings the leakage rate through the coated fabric was calculated to be 817 cm^3/

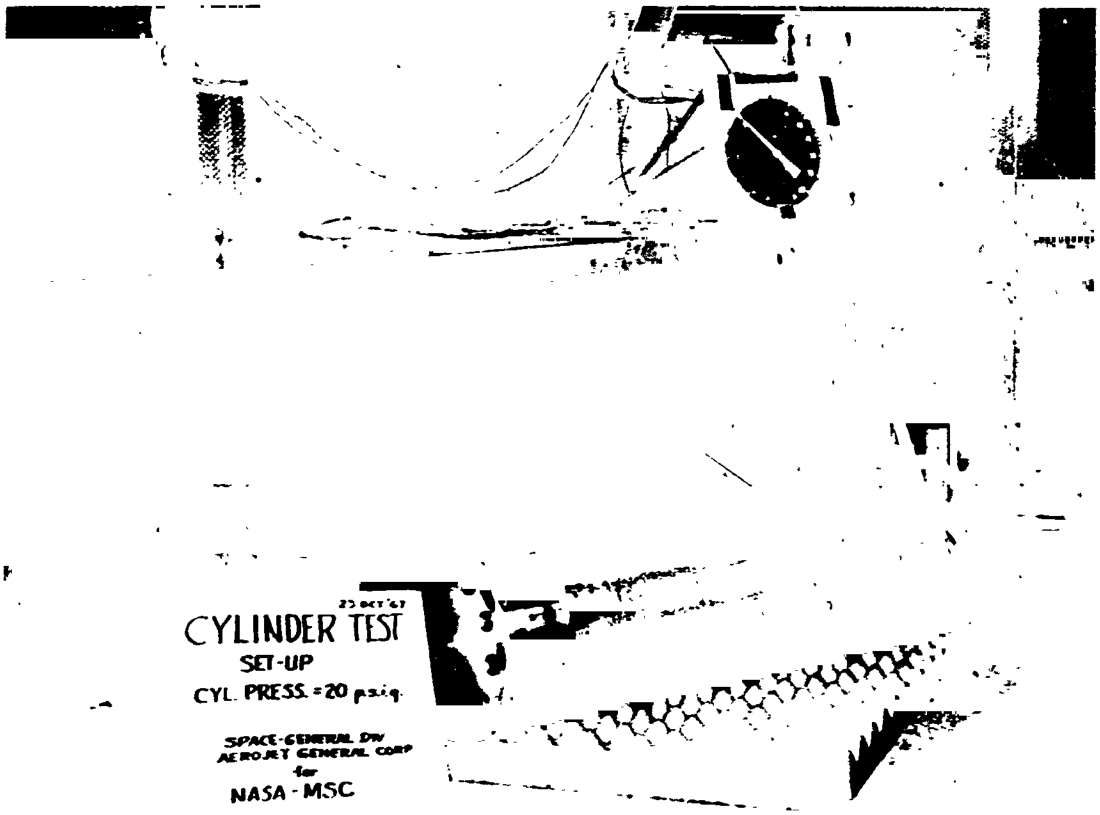

Figure 3. Test Cylinder at 20 psig

sq.ft./24 hours at 10.0 psig. A "burst" test was conducted for 15 minutes at 34.7 psig. This pressure allowed for a factor-of-safety of 2.0 in the calculated hoop tension yield pressure. As before, no degradation of the structure or sealant was observed during or after the test.

The 1.0 mil diameter filament metal fabric used during the majority of this program was 14 mils thick. The ultimate tensile strength of the cleaned fabric in the warp direction was 318 pounds/inch with a yield strength of 286 pounds/inch. This tensile strength is about twice that required for the lines of a space suit. Accordingly, several small coupons were fabricated from a nickel chromium alloy fabric whose tensile strength was about 170 pounds/inch. The filament diameter of this fabric was 1/2 mil, and the fabric was 7 mils thick. No problem was encountered during the processing of these samples, and the flexibility was greatly increased because of the smaller diameter filaments and reduced thickness of the metal fabric. The coupons represented a very attractive composite from which a space suit could be fabricated.

The spot weld technique for joining metal fabrics was developed during a contract with the Manufacturing Technology Division, AF Materials Laboratory (AF33(657)-10252). Numerous joining methods for making high efficiency seams in metal fabrics were studied during this program, including sewing, exothermic brazing, resistance brazing, interrupted and continuous seam welding, ultrasonic welding, and spot welding. The latter method was ultimately selected as the most satisfactory.

Joint efficiencies of greater than 90 percent have been reported for sewing techniques; however, this requires a relatively stiff double-fold or French seam which results in four layers of fabric to join two pieces. Also, this joining technique does not permit the fabrication of high efficiency crossing or compound seam intersections. The brazing techniques in general resulted in seams that were too rigid. Seam and ultrasonic welding caused severe weakening of the fibers adjacent to the weld and therefore resulted in low joint efficiencies.

The spot weld seams were the most flexible and did not severely damage the fabric even without an absolute inert atmosphere. The joining technique developed consists of two rows of closely spaced, small diameter spot welds with the rows about 1/4 inch apart and using 1/2 to 3/4 inch overlap of the fabric layers. Extensive qualification testing on this welding system showed that the weld strength exceeded 85 percent (even at temperatures above 900 degrees Farenheit) of the parent fabric strength with a 99.95 percent confidence level. As many as five layers of fabric can be joined such that intersecting seams and multiple plies can be accommodated, permitting the construction of surfaces with compound curvatures. The previously referenced metal fabric made from 1.0 mil diameter filaments of Karma alloy was used during this study. A photograph of the spot welder is shown in Figure 4.

The Air Force contract was directed at the development of an inflatable re-entry glider. To provide a suitable pressure bladder, a technique to completely impregnate the metal fabric with a silicone elastomer was developed. The impregnation system avoids entrapped air spaces which would blister upon ejection into the space vacuum. It would also minimize heat conduction and internal abrasion during flight. An external coating of an ablative silicone was applied to provide thermal protection to the system.

During this program a number of components were tested using simulated re-entry loads and temperatures to prove out the design technology as well as the fabrication methods. All of the test components failed at internal pressures in excess of that theoretically calculated regardless of the effects of previous packaging, creasing and folding, or high bending, shear and torsion loads even at environmental test temperatures as high as 2000 degrees Farenheit. The high burst pressure indicated that these factors had little, if any, detrimental effect on the ultimate

Figure 4. Metal Fabric Spot Welder

strength of the component. These observations indicate the superiority of elastomer coated metal fabric composites for flexible and/or expandable space systems which must function in high temperature environments.

SESSION VII

ASSOCIATED SPACE EXPERIMENTS AND SIMULATION

Session Chairman: Colonel J. Green
SAMSO Det. 2, AFSC Field Office
NASA Manned Spacecraft Center

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EXPERIMENT M-509
ASTRONAUT MANEUVERING EQUIPMENT

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SUMMARY: The proposed experiment would be an inflight investigation of man's maneuvering capability. Several different types of astronaut maneuvering unit concepts would be evaluated using a single test vehicle. The data gained would be used to develop a maneuvering technology base and to calibrate ground-based zero-gravity simulators.

INTRODUCTION

Two maneuvering units were developed for the Gemini Program: the Hand-Held Maneuvering Unit (HHMU) and the Astronaut Maneuvering Unit (AMU). Neither unit received sufficient evaluation during orbital flights. The HHMU was used during the Gemini IV mission, but limited propellant capacity allowed only 20 seconds of thruster operation. The Gemini VIII mission was aborted before the scheduled extravehicular activity (EVA) and HHMU evaluation. During the Gemini X mission, the HHMU was used for two translations between the Gemini spacecraft and the Agena spacecraft and for one attitude correction. The HHMU was not used to stop a translation, and the EVA was curtailed before the scheduled extensive HHMU evaluation because of low spacecraft propellant. Body-positioning problems during the Gemini XI mission EVA resulted in terminating the EVA prior to any use of the HHMU. Body-positioning problems during the Gemini IX-A mission EVA lead to fogging of the pilot's visor as he attempted to check out and don the AMU. The AMU was not flown on the Gemini IX-A mission and was dropped from the Gemini XII mission so that time could be devoted to more basic problems of working while performing EVA. As a result of the Gemini Program, much experience was gained in the design, fabrication, and qualification of maneuvering unit hardware for space, but very little knowledge was gained about man's maneuvering capabilities and limitations. In each case where the maneuvering unit evaluation was aborted, the problem was not with the maneuvering unit, but with the spacecraft or the unexpected problems of working in the EVA environment.

Manned spaceflight programs planned for after the Apollo lunar landing will make it possible to gather data in orbit on man/machine maneuvering performance, without the problems of supporting man in the space environment. Specifically, the Saturn IVB (S-IVB) Orbital Workshop (OWS) planned for the Apollo Applications Program (AAP) would provide a large-volume (greater than 10000 cubic feet), zero-gravity laboratory with a habitable environment inside. A maneuvering unit could be used inside the OWS to perform EVA-type tasks without subjecting the astronaut to the hazards and risks of EVA. Experiment M-509, Astronaut Maneuvering Equipment, is a proposed investigation to do just that.

The purpose of this paper is to discuss the relationship of experiment M-509 to the development of an astronaut maneuvering technology base and to describe the proposed experiment.

MANEUVERING TECHNOLOGY DEVELOPMENT

The proposed M-509 experiment is to be a part of an overall program to develop a capability to support and enhance future manned spaceflight missions, that is, an astronaut-maneuvering technology base. The purpose of the experiment would be to obtain engineering and technological data in flight on selected maneuvering techniques and man performance capability for specific maneuvering tasks. The primary objective of experiment M-509 would be to gain the necessary experience and technology to establish maneuvering techniques which will enhance man's EVA capability. The focal point of the experiment would be, of course, the inflight evaluation of the experimental hardware. The corollary effort would include an assessment of current technology and capabilities; development and test of experiment hardware; flight simulation for optimizing the inflight time line, gathering base-line data, and astronaut training; and data reduction and analysis of the inflight test results.

Another aspect of this maneuvering technology base in that it would provide good analytical tools for maneuvering systems design and operational analysis. Several ground-based simulation techniques (computer-driven simulators, air-bearing platforms, and neutral buoyancy) appear to be useful in this respect. However, these techniques would be far more valuable if their fidelity were known. Base-line data gathered before and after the inflight test, when compared with the same parameters gathered in flight, would yield an assessment of the accuracy of the simulators. Also, particular simulators may be good for only certain portions of an EVA maneuvering task. Although the cost of ground-based evaluations would be far less than space evaluations, some simulators would be less expensive to operate than others. Knowledge of the advantages, limitations, applications, and accuracy of each simulator would allow the best technique to be used for a given simulation task.

Significance

The development of a maneuvering technology base would make an important contribution to the overall manned spaceflight program by enabling man to perform a wider range of EVA tasks and to perform many tasks more effectively. Experiment M-509 would be a major part of developing this technology base. In addition, the experiment would be designed to take maximum advantage of planned missions, existing technology, and available time to: (1) insure astronaut safety; (2) develop versatile equipment; and (3) complete a comprehensive inflight evaluation as discussed in the following paragraphs:

1. Astronaut Safety: The S-IVB OWS provides an enclosed, pressurized, zero g environment in which an experiment subject and his observer/assistant could operate under simulated EVA conditions without the hazards and constraints normally associated with EVA. The OWS is the only planned spacecraft which provides sufficient volume to conduct experiment M-509 without EVA. Although long translations (beyond 2[illegible] feet) could not be conducted, the very important problems associated with the astronaut's fine motor ability and close-in maneuvering capability (stationkeeping, altitude control, docking, and donning/doffing) could be adequately demonstrated and investigated. With its habitable environment, the OWS would allow both man and machine performance of EVA-type tasks to be observed and documented under laboratory-like conditions. This data under long-term zero gravity cannot be obtained from ground simulation. As an example, the OWS would provide the first valid method to evaluate an astronaut's ability to recover from a tumbling mode when two or more rotational axes are involved.

2. Versatile Equipment: The major item of flight hardware planned for experiment M-509 is the Astronaut Maneuvering Research Vehicle (AMRV). The AMRV would be a sophisticated experimental test bed which could be used to evaluate several different maneuvering techniques with various EVA-type tasks, to gather extensive performance data in orbit, and to be refurbished for multimission applications. It would be designed to insure maximum versatility and would be, as the name implies, a research vehicle. Other control modules could be added on later missions (such as a voice controller, a foot controller, or even a remote controller). The AMRV would be compatible with operation on an umbilical to the spacecraft, thus allowing use of the AMRV anytime the OWS is manned. Capability for reuse on later missions would allow certain tasks to be repeated, new tasks to be added, and additional astronauts to gain flight experience, as well as to test new control concepts.

3. Comprehensive Evaluation: To develop the desired astronaut-maneuvering technology base, an extensive and systematic evaluation of man's capability to maneuver free of his spacecraft would be conducted inside the OWS. The corollary effort of the flight hardware development program

would in itself develop a minimum technological foundation. However, development of a broad base maneuvering technology and the attendant confidence in its validity is dependent upon flight tests. The inflight evaluation would provide realistic experience about the handling qualities and performance of each maneuvering technique flown and about the capabilities and limitations of both man and machine. Specific maneuvering tasks would be representative of expected EVA tasks and, as new tasks arise, these could be added. Data reduction and analysis of flight results would provide a valid basis for establishing performance requirements of future maneuvering systems and an index of what man could be expected to accomplish when assisted by the appropriate maneuvering unit. The data would provide procedures for planning future manned spaceflight missions.

Application

At this point, one may ask "How would the results of the experiment be used or applied?" The inflight experiment would demonstrate certain applications when a maneuvering capability would enhance man's ability to perform those tasks which can best be accomplished outside the spacecraft. (As used here, a maneuvering capability provides the astronaut the ability to control his body attitude and position without the necessity to apply forces and/or torques to the spacecraft). Likewise, the utility of a maneuvering capability would be proven marginal or not required in some applications. In addition, some maneuvering techniques and equipment would be shown to be unsatisfactory for certain maneuvering tasks. Successful completion of experiment M-509 would provide an inventory of proven techniques which could be applied to a wide range of potential mission requirements.

Specific potential applications are described in two categories in the following paragraphs: Generic and Mission.

1. Generic Applications: Generic applications denote those applications when a maneuvering capability provides distinct advantages but is not necessarily a requirement.

a. Translational Capability: Free-space transfers between separate vehicles are possible with a maneuvering capability. A translation capability includes a means for the astronaut to control body attitude and position, that is, properly orient his body prior to initiating the transfer, control body orientation to keep the target or reference in sight during the transfer, and arrive at the target with the desired rate and body orientation.

b. Mobility Aid: A limited maneuvering capability - combined with handrails, umbilicals, and lifelines - would enhance the astronaut's effectiveness for EVA in the near vicinity (<100 feet) of the spacecraft. Numerous handrails are employed on the Apollo and AAP spacecraft. Life support umbilicals will likely be used. A maneuvering unit such as the HHMU would provide powered body-positioning capability to eliminate the astronaut from having to use his wrist to torque his body around to this desired position. A rapid means of returning to the spacecraft would also be available. In addition, by playing out the life support umbilical and by using the HHMU to control the umbilical direction, areas away from the preinstalled handrails could be reached.

c. Automatic Stabilization: A sophisticated maneuvering unit with automatic stabilization could be applied to a wider range of applications. The major advantage of this type unit is that it would hold a given attitude automatically (hands free). A fixed-attitude reference would allow the astronaut to visually detect relative motion between his target and himself. This feature would allow translations to take place without rotations caused by thrust misalinement and would allow pure rotations (pitch, roll, and yaw) to be executed without cross-coupling into other axes. The automatic stabilization subsystem would also produce rotations at the desired rate (either proportional or preselected discrete rates) and reduce these rates to zero when the command input is removed.

2. Mission Applications: Mission applications are discussed in terms of classes of applications. For the applications covered, a maneuvering capability is not necessarily a requirement but would enhance man's EVA capability. A maneuvering unit optimized for the specific mission would provide a means to accomplish the task which would be competitive with other EVA techniques.

a. Assembly: A number of proposed missions involve the assembly of large structures in orbit (such as erection of a large (>100-foot diameter) radio telescope, deployment of a large (>35-inch diameter) optical telescope, erection of a large area (>100 square meters) X-ray telescope, and deployment of large solar panel arrays). A maneuvering unit would enable the astronaut to string cables between distant points on the structure, move rapidly over the surface, make adjustments on linkages, monitor automatic system deployment, and back up automatic deployment mechanisms.

b. Inspection: A maneuvering capability would enable the astronaut to inspect visually large spacecraft surfaces. The large number of preinstalled handrails otherwise required would be impractical. Precise attitude and position control (stationkeeping) would enable inspection and/or photographing of small satellites without touching the target (for example, to keep from upsetting the attitude control system of the satellite). Operational systems inspections might include observing an overboard dump, checking external plumbing for apparent leaking (malfunction analysis, periodic inspections on long missions, examining spacecraft surfaces for reaction control system (RCS) plume impingement degradations, and monitoring many more external conditions which must be accomplished in orbit.

c. Maintenance: Repair, replacement, and replenishment are considered a part of maintenance. Numerous examples of possible tasks can be envisioned on both the parent spacecraft and other satellites. Some examples include replacement of solar array panels, cleaning sensor surfaces (lens), deployment of emergency oxygen reserves outside for support of subsequent EVA, replacement of a failed directional antenna, and so forth. In addition, a proven maneuvering capability would significantly affect the redundancy versus maintainability concepts.

d. Operational Support: Certain tasks, which are peculiar to a specific mission and do not fall into the other classes of applications, probably will be required. Those applications when a maneuvering capability could be used to advantage include activation or reactivation of dormant vehicles and resupply of active vehicles (cargo and crew transfer, installation of external umbilicals, solar panel deployment); EVA technology development (tests of new concepts, demonstration of a capability prior to its application); and support of in-orbit manufacturing facilities.

e. Rescue: A maneuvering capability would be very valuable in time-critical rescue operations, either for carrying additional consumables to, or retrieval of, the stranded astronaut. Also, rapid assistance could be provided to an astronaut caught in the structure in such a manner that he could not reach the point of entanglement.

f. Scientific: Many scientific experiments have requirements for extravehicular operations which could be enhanced by a maneuvering capability. These experiments include tasks such as mapping plasma wakes, radiation fields, and magnetic fields; calibration and alinement of large antennas and telescopes; retrieval and replenishment of data packages (such as film cassettes); visual readout of engineering data and functioning condition; experiment operations (changing filters, changing modes, calibrating, adjusting, etc.); and experiment monitoring (assuring proper functioning and contingency support).

EXPERIMENT IMPLEMENTATION

Having established the utility of experiment M-509, possible experiment implementation will be discussed. In this section, the proposed experiment preflight, inflight, and postflight activities are covered.

Preflight Activities

Approximately 85 percent of the time needed for the experiment program would be concerned with the preflight activities. These activities include selecting the maneuvering unit concepts to be evaluated inflight, developing and testing experiment flight hardware, and conducting various kinds of simulations.

1. Selection of Maneuvering Techniques: Over the past 10 years, numerous concepts have been proposed for astronaut maneuvering, and several concepts have

reached the hardware stage. Three criteria have been established to select the particular maneuvering techniques to be evaluated. First, the technique must provide the astronaut with powered maneuvering capability in six degrees of freedom, that is, fore/aft, left/right, up/down, pitch, roll, and yaw. Second, the technique must have already received extensive design analysis and ground testing. Finally, the technique must have unique design features (which cannot be adequately verified by ground test) and potential space applications.

Four maneuvering techniques (control concepts), embodying two maneuvering unit configurations, are planned for the AMRV design. Two techniques would employ manual control; that is, the astronaut must visually detect attitude and attitude rates and manually activate the thruster system to control attitude and position. The other two techniques would employ automatic stabilization. The two automatic stabilization subsystems planned are quite different in mechanization and handling qualities, but each provides a fixed-attitude position, provides rotation about the desired axis at a given rate, and stops rotations automatically. The method of use of each control mode and the planned AMRV configuration are covered in the next section.

2. Flight Hardware: The proposed AMRV consists of two parts as shown in figure 1: a backpack and a handgun. The backpack would be a back-mounted module with multiple fixed-position thrusters. This is a traditional maneuvering unit configuration and quite similar to the AMU. The handgun would be an updated version of the Gemini HHMU and would represent the hand-held, manually directed thruster configuration. For the handgun evaluation, the HHMU would be plugged into the backpack with a short umbilical. The backpack then serves as a support module for the HHMU and provides propellant and instrumentation. Specific control modes will be discussed by unit in the following sections.

a. Backpack: The backpack would contain five major subsystems. These subsystems are the propulsion, the electrical, the data, the attitude control, and the displays/controls.

(1) Propulsion: Numerous solenoid-operated thrusters would be mounted around the backpack in fixed positions. The thrusters would be arranged to provide translation in the fore/aft, left/right, and up/down directions with respect to the pilot. Maximum acceleration would be about 0.5 ft/sec^2. Also, the thrusters would produce pure couples for attitude control-pitch, roll, and yaw. A propellant isolation valve would be readily accessible to the pilot if a thruster sticks open. The propellant would be gaseous oxygen. Propellant would be supplied from the spacecraft through an umbilical or from a self-contained propellant module. Several propellant modules would be carried and as each was expended, a replacement would be installed.

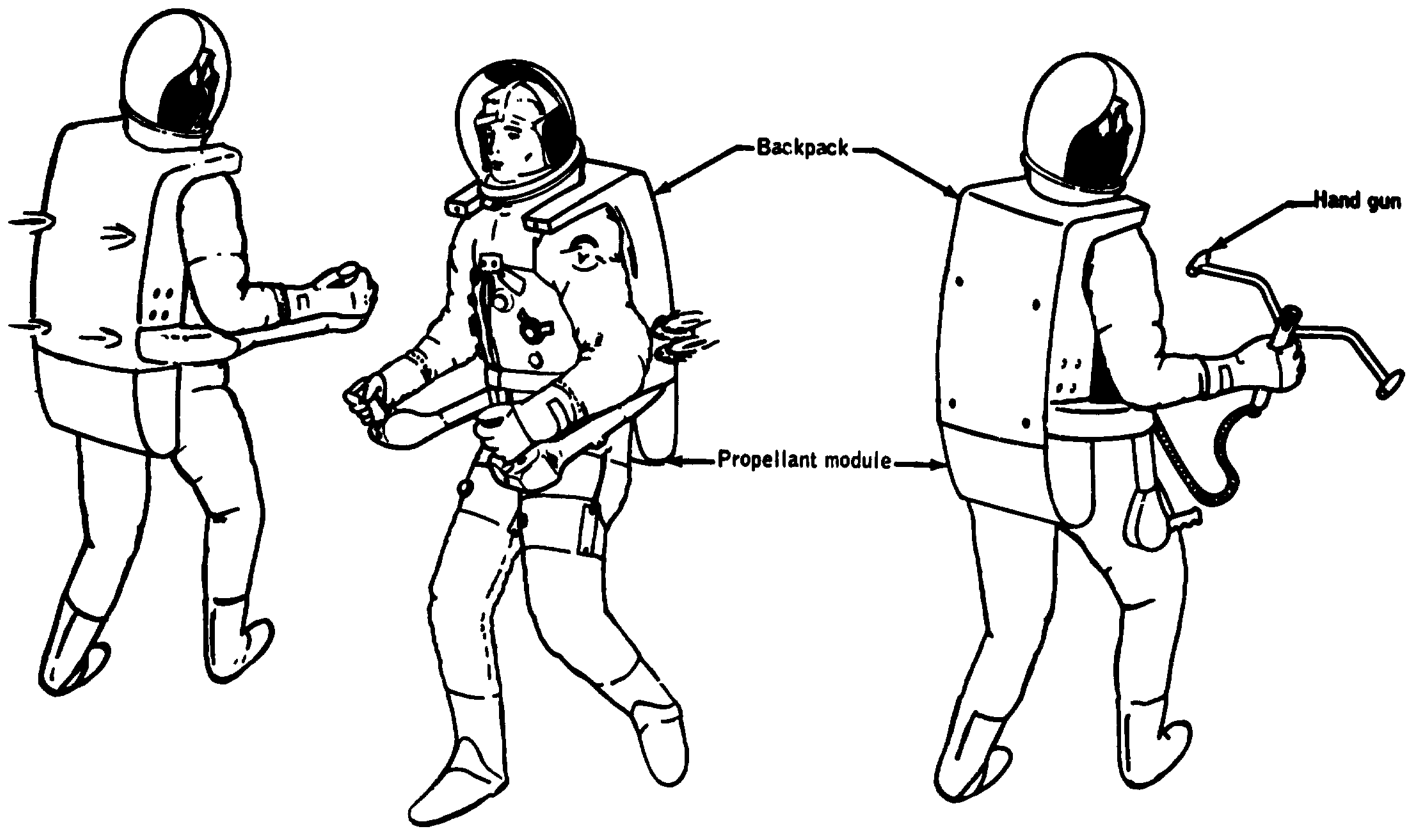

Figure 1. - Artist concept of AMRV.

(2) Electrical: Various electronic components would be employed in addition to a self-contained battery, battery charger, power distribution and protection components, switches, relays, and so forth. Electrical power from the spacecraft would be used for starting gyros, for checkout, and for most of the inflight tests.

(3) Data: The data subsystem would include sensors, signal conditioning equipment, an encoder/multiplexer, a transmitter, and a receiver. The receiver would be mounted on the spacecraft and hardlined to the spacecraft data system for subsequent air-to-ground dump. Parameters (such as propellant pressure and temperature, solenoid valve operations, angular rates, command inputs, and various housekeeping data) would be recorded.

(4) Attitude Control Subsystem (ACS): The ACS would be the heart of the AMRV. It would have four control modes, each selectable in flight. The first three modes would be used in conjunction with the backpack configuration evaluation, while the fourth mode would concern the handgun configuration evaluation. The purpose and operation of each mode follows:

(a) Direct: The purpose of operation in this mode would be to evaluate an astronaut's capability to control attitude and attitude rates of the backpack by manually inputting commands based upon his visual cues. Movement of the rotational hand controller from its neutral position would result in continuous angular acceleration as long as the command is held. Release of the controller would stop the acceleration, but the resulting rates would continue.

(b) Automatic No. 1: Two modes of automatic stabilization would be investigated. Each would provide automatic attitude hold, rotation about the desired axis at the desired rate, and reduction of attitude rates to zero when the input command is removed. The first mode-Automatic No. 1-would be used to evaluate handling characteristics of a rate rensing system, which uses rate gyros and the associated control electronics.

(c) Automatic No. 2: The second automatic stabilization mode would be used to evaluate handling characteristics of an inertial system, which uses control moment gyros (CMG) to resist external torques and internal torques to produce pitch, roll, or yaw as desired.

(d) HHMU: The HHMU would be evaluated in this mode. The backpack automatic control systems and hand controllers would be out of the control loop. The backpack would serve as a support module to provide instrumentation and propellant for the HHMU. Provision may be made for activating one of the automatic stabilization subsystems, if undesired rates develop while the HHMU is being evaluated.

(5) Displays and Controls: This subsystem would consist of separate rotation and translational hand controllers, a mode selection switch, a propellant isolation switch, and a self-contained propellant pressure gauge. Flight of the backpack would be

quite similar to maneuvering the Apollo spacecraft. Hand controllers would be similar to those of Apollo spacecraft with the left-hand controller used for translational control and the right-hand controller used for rotational control.

b. Handgun: The handgun or HHMU will be discussed in terms of the same five subsystems as the backpack even though some subsystems would not normally be classified as subsystems because of their simplicity.

(1) Propulsion: Two 1-pound tractor thrusters and one 2-pound pusher thruster would be employed in the same manner as the Gemini HHMU. A single-throttle valve would regulate the total thrust between 0 and 2 pounds. A shuttle valve downstream of the throttle valve would route the propellant to either the tractor or the pusher thrusters. The shuttle valve normally would be in the tractor position since the tractor mode is used most often. Propellant would come from the backpack through a short hose to the HHMU. The same propellant isolation valve used for the backpack will likely be used for the HHMU.

(2) Electrical: No electrical power would be required for the HHMU operation. Power would, of course, be required for operation of the backpack as a support module.

(3) Data: Two pressure transducers would be use: one in the tractor manifold and one in the pusher manifold. Output from the transducers would be fed into the backpack instrumentation subsystem. Pressure levels would indicate which thrusters are being operated and the amount of thrust being commanded. An item of primary interest here would be the comparison of commands for full thrust and commands for partial thrust.

(4) Attitude Control Subsystem: The astronaut would serve as the attitude control subsystem. Through visual cues, he must detect attitude rates and then properly orient and fire the HHMU to change and control attitude.

(5) Displays and Controls: Displays would consist of streamers attached to the nozzles so that they flutter when the thrusters operate and of a color band on the shuttle valve stem that would be visible when the pusher mode is selected. The only controls would be the manually operated shuttle and throttle valves described earlier.

3. Simulations: Various kinds of simulations are planned throughout the experiment program; however, most simulations would be conducted during the preflight phase. The purposes of the simulations are to screen the maneuvering tasks to be evaluated in flight, to optimize the inflight time line, to gather base-line data before and after the flight, and to train the astronauts. Three kinds of simulators would likely be used: computer-driven, air-bearing table, and neutral buoyancy.

The computer-driven simulators (either moving base or visual reference projection) would be used to eliminate the effects of the one g field and to gather data on the performance of the AMRV. The same parameters to be measured in flight would be gathered with the computer-driven simulator. The air-bearing table simulation would be primarily for astronaut familiarization. A special training unit AMRV, which would look and feel like the real unit, is planned. By the use of support scooters which would float on air cushion pads, three degrees of freedom (DOF) would be achieved. By alternate positioning of the AMRV on the scooters, all six DOF could be evaluated.

Neutral buoyancy facilities would be used to develop and validate the unstowage, deployment, don/doff, checkout, and stowage procedures and time lines. A special unit, which would be neutrally buoyant in water, is planned for these simulations.

In addition to the AMRV, the experiment flight hardware would include a propulsion gas umbilical (PGU), a data receiver, and miscellaneous mounting hardware. The PGU would be a special minimum-weight, maximum-flexibility hose with electrical power and communication lines. The PGU would allow spacecraft oxygen to be used as propellant modules which must be carried. Also, by using pure oxygen as the propellent gas, the gas expended would serve as makeup gas to maintain the spacecraft cabin pressure. The data receiver would be hardlined to the spacecraft data system and would receive the data transmitted from the AMRV. Various pieces of support hardware would be needed to allow rapid, inflight unstowing and donning/doffing of the AMRV.

Inflight Activities

Inflight activities would begin with unstowage and setup of the experiment hardware in orbit. It is planned that the hardware be stowed and operated inside the spacecraft. Hence, most of the setup operations could be completed by one crewman in shirtsleeve clothing. During the maneuvering operations with the AMRV, two crewmen would be required; one would serve as the subject and the other would serve as a safety observer/camera operator. The planned inflight procedures for evaluating the AMRV are described in the following paragraphs.

1. Operating Modes: Ideally, the complete AMRV evaluation would be conducted while the subject was wearing a pressurized space suit to represent operation in the vacuum environment. However, in order to reduce the life support requirements, the subject would wear shirtsleeve clothing for most of the experiment. Even though much of the propellant required would come from the spacecraft through the PGU, a part of the maneuvers must be conducted without any umbilicals or tethers to the spacecraft. This procedure is necessary to investigate the effect of the umbilical. For example, operation with the HHMU may be aided by an umbilical. However, a backpack

equipped with CMG may be hampered by an umbilical; that is, the umbilical may provide sufficient external torques to saturate the CMG. Therefore, the AMRV would be operated in the three following modes:

Operating Mode I - Subject in shirtsleeve clothing, PGU attached to the AMRV.

Operating Mode II - Subject in shirtsleeve clothing, no umbilicals or tethers.

Operating Mode III - Subject in a pressurized space suit, PGU and life support umbilical attached.

For Operating Mode II, the AMRV would have to be capable of operation from self-contained propellant and electrical power. The operating modes would be sequential for each subject so that the progression would be from the simplest to the most difficult operations.

2. Tasks: The subject would have to perform four tasks in each of the operating modes described in the previous paragraph. Each task would be related to a control mode of the attitude control subsystem and is summarized as follows:

Task A - Evaluate flying the backpack without automatic stabilization (that is, manual)

Task B - Evaluate flying the backpack with an attitude rate sensing system (Automatic No. 1)

Task C - Evaluate flying the backpack with an inertial attitude system (Automatic No. 2)

Task D - Evaluate flying the HHMU with the backpack as a support module.

3. Maneuvers: For each of the tasks required, several maneuvers would be flown. These may be classed as basic and mission, as follows.

a. Basic Maneuvers: Each subject would receive his initial familiarization with flying the AMRV in the zero g environment during Operating Mode I. Certain basic maneuvers would be flown to give the subject a "feel" for the unit and to gather data on the performance of the AMRV. These maneuvers would include pitch, roll and yaw attitude maneuvers, attitude hold, and short translations.

b. Mission Maneuvers: As the subject develops skill and confidence in the AMRV, more difficult maneuvers would be undertaken. One planned mission maneuver is a traverse across the spacecraft. This maneuver would require orientation to the proper body attitude, acceleration, midcourse corrections, deceleration, stopping near a target without contacting it, and flying around the target to simulate an inspection. Then the subject would reorient body attitude for a return across the spacecraft, accelerate, make midcourse corrections, decelerate, dock at a work site, and attach body restraints. Another mission maneuver would entail the observer to impart a spin to the subject-first about a single axis, then about multiple axes. The subject would then arrest the spin with the AMRV for each of the four tasks. Operating Mode II would be necessary for this tumble recovery maneuver. Other mission maneuvers will be defined during the simulation activities.

4. Flying Time: In order to minimize the crew time in orbit, all three crewmen would not get equal flying time with the AMRV. Those maneuvers which could be demonstrated satisfactorily in one run would be flown by only one crewman. Other maneuvers, when statistical data are needed or when the data are largely subjective, would require two or three crewmembers to participate. Also, in order to limit the amount of propellant (oxygen) expended per day, the experiment would likely be scheduled on several different days. Too much maneuvering would result in a buildup of cabin pressure so that the cabin relief valve would operate and dump oxygen overboard. A proposed flying time schedule by man, day, and operating mode is shown in table I.

5. Data Recording: In addition to the data gathered as part of the AMRV instrumentation subsystem, photographic and subjective data would be collected in flight.

a. Photographic Data: Motion picture photograph is planned using onboard movie cameras. This film would be returned to earth for correlation with AMRV data. The film would also provide data about position, linear and angular velocities, and accelerations.

b. Subjective Data: The subject would be connected into the spacecraft communications system during Operating Modes I and III. He would then provide a running commentary of his actions and reactions. The observer could also supply comments on the AMRV/man performance. Immediately after each run, the subject would provide a short critique of the test.

Postflight Activities

After the crew is recovered, debriefings are planned. All data would be reduced and analyzed to assess the performance of man and machine. A maneuvering unit handbook would be prepared establishing design criteria for future maneuvering systems and base-line data for planning future missions. A part of this handbook would be devoted to simulators-their applicable areas, fidelity, and limitations. This assessment would come from a correlation of flight data with preflight and postflight base-line data from ground-based simulators. Besides calibrating the simulators, recommendations for improved techniques would be developed.

TABLE I. EXPERIMENT RUN SCHEDULE

	EXPERIMENT RUN #1	EXPERIMENT RUN #2	EXPERIMENT RUN #3	EXPERIMENT RUN #4	TOTALS
SUBJECT #1	I - 1:00* II - 0:25	I - 1:00 II - 0:25	III - 0:40		3:30
SUBJECT #2	I - 1:00		I - 0:45 II - 0:25	III - 0:20	2:30
SUBJECT #3		I - 1:00		I - 0:30	1:30
TOTAL	2:25	2:25	1:50	0:50	7:30

*Operating Mode and Flight Time (Hours:Minutes)

EVA SUPPORT OF SCIENTIFIC/TECHNICAL EXPERIMENTS

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SUMMARY

A series of astronomy and related scientific and technical experiments proposed for earth orbital missions in the immediate post-Apollo period and defined as requiring astronaut extravehicular activity (EVA) for experiment success is examined for validity of the EVA requirement. Criteria for establishing validity as well as the analytic procedure are outlined. It was found that the EVA support requirement for certain experiments could be deleted or significantly reduced through modification of experiment procedure or equipment; whereas for other experiments a valid requirement for EVA support was found to exist, and in some instances the experiment could be enhanced through modification or expansion of the experiment-related EVA function. A set of suggested experiment procedure and experiment equipment design guidelines for EVA support is given.

INTRODUCTION

This paper is based on a recently completed study[1] of the extravehicular engineering activity (EVEA) required to support future manned earth orbital scientific and technical experiments. In that study, proposed experiments in a number of scientific and technical disciplines were investigated. This paper describes the method by which the proposed scientific/technical experiments were analyzed to establish the validity of the extra-vehicular activity (EVA) support requirement, describes some of the findings, and arrives at some tentative recommendations for scientific/technical experiment design to achieve more optimum utilization of astronaut EVA support.

The purpose of the EVEA Program Requirements Study was to define the extravehicular activity capability in terms of techniques and equipment required to support earth orbital scientific and technical experiments in the 1968-1980 period with emphasis on the 1971-1974 period. In this study, approximately half of the proposed experiments projected for this period logically require some degree of EVA support to satisfy the overall mission.

The candidate scientific and technical experiments investigated for potential EVA requirements were obtained from several experiment data sources including NASA in-house and contractor studies, experiment proposals, and AAP

[1]Extravehicular Engineering Activities (EVEA) Program Requirements with Emphasis on Early Requirements Contract NAS8-18128, May 1968

experiment data sheets. The approximately 1200 experiments that were identified as a baseline for analysis in the study were categorized into nine disciplines; examined (within the discipline) for redundancy of experiment objective, equipment or mode of operation; and examined for validity as a space experiment and for potential EVA requirement. The experiments were then grouped into three periods (1968-70, 1971-74, 1975-80) during which the experiment was likely to occur.

A total of 280 experiments for which manned EVA roles had been identified were projected for 1971-1974, the time period of principal interest. These experiments were reviewed to eliminate operational redundancy across disciplines by regrouping the experiments according to the types of astronaut EVA functions required to support the experiment activities, and by eliminating certain experiments that had EVA functional support requirements that were the same as those of another discipline. This selection process yeilded 102 scientific and technical experiments projected for the 1971-1974 period and requiring EVA support.

Because the next step in the study required a more thorough analysis of the scientific and technical experiments to define their detailed EVA support requirements, it was necessary to reduce the number of experiments projected for the period of interest to a manageable number. Accordingly, 16 of the 102 remaining experiments were selected as being representative of those requiring EVA support. Criteria for selection of the representative experiments included validity of experiment concept, state of experiment development, realism in terms of the time period, and extent and type of EVA involved.

Of the 16 representative scientific/technical experiments, approximately 1/3 (6 out of 16) fell within the scientific discipline of astronomy. This relatively high proportion of astronomy experiments does not necessarily indicate a predeliction toward astronomy on the part of those making the experiment selections. Rather, it is indicative of the importance of astronomy experiments at least in the early to mid-1970 decade, and of the importance of man in the conduct of this type of space experiment. Three of the six astronomy experiments involved optical telescopes of conventional design, one X-ray astronomy experiment, one radio astronomy experiment, and one general purpose electromagnetic radiation package experiment. In addition, the representative experiment list included four bioscience experiments, and two each in the areas of Communications/Navigation, Physical Sciences, and Advanced Technology - Orbital Operations.

The EVEA study was directed toward defining the relatively near-term EVA requirements and to some extent was constrained by the equipments and techniques likely to be available by that time period. However, the study revealed the necessity for equipment and technique development for the early time periods, and equally important, indicated the need for research and technology support to EVA for time periods beyond the early 1970's. Spacesuit mobility, EVA translation capability, and life support system capacity were all found to be limiting factors in EVA work task performance in early time periods. As space technology proceeds and manned space missions become more complex, astronaut EVA capability and work capacity will have to increase accordingly.

ANALYSIS OF SELECTED EXPERIMENTS

Each of the 16 representative experiments was subjected to a further, in-depth analysis for the purpose of establishing the validity of the EVA requirement. The analysis consisted of an investigation and review of the basic scientific equipment in terms of its capability to achieve the stated scientific objective, and in terms of its capability to operate in the space environment. The experiment equipment operational configuration was then reviewed, and physical interfaces with the spacecraft and spacecraft systems were defined. Subsequently, the in-orbit experiment procedures were examined and experiment functions analyzed. From this analysis, man's role in relation to the experiment was determined, and the EVA requirements were stated. Once identified, the EVA tasks were subjected to a functional flow analysis for the purpose of establishing the validity of the requirement in terms of experiment need and EVA astronaut capability.

Although the experiment definition analysis often resulted in a modification of experiment procedures to provide for more efficient use of astronaut EVA, it was found that the requirements for EVA were valid in virtually all the representative experiments. Subsequently, the EVA requirements were summarized for the purpose of identifying discrete EVA task functions and EVA equipments necessary to perform those functions. As a result of this EVA Operations Analysis, it was determined that over 90% of the EVA requirements could be met by only 22 discrete EVA task functions and subfunctions used either singly or in combination. Associated with these functions were various EVA equipments required to perform the function.

By way of illustration, consider a typical optical astronomy experiment. The experiment concept, Figure 1, utilizes reflecting telescope for stellar astronomy. The objective of the experiment is to obtain direct image photography and high resolution spectral photography of selected stars and galaxies in the ultra-violet (UV) portion of the spectrum (1050 A to 4,000 A). The experiment is planned for a relatively early time period (1971-74), and is intended to operate with a manned earth-orbiting space station (EOSS). The telescope system is mounted in the Apollo Telescope Mount (ATM) which in turn is attached to the Lunar Module (LM) docked to the Multiple Docking Adapter (MDA) of the orbital workshop configuration (OWS).

The basic experiment equipment consists of a compound reflecting telescope of 1-meter aperture and a focal ratio of approximately f/10. Light from the target source enters the telescope, is reflected from a primary mirror to a secondary mirror, then back ghrough a hole in the primary and comes to focus at a focal plane located behind the primary. A third, diagonal mirror, is used to reflect the light rays so as to place the focal plane at either a direct image camera, spectrometer, or similar ancillary or recording equipment.

This telescope configuration is typical of the optical astronomy experiments proposed for 1970-80 decade. The principal differences between early and later time periods is in size of relative aperature, quality and sophistication of the instrument and its related equipment. In the configuration illustrated, the telescope system is enclosed in a large thermal shield mounted to the ATM/LM configuration which operates attached to the OWS. The astronaut-observer commands telescope lunar module. Most of the experiment procedures, including checkout and observation, are according to a preprogrammed sequence. The observer's function is one of monitoring equipment operation, maintaining contact with ground principal investigators, and overriding or modifying the preprogrammed sequence, and performing EVA at stated times in the experiment sequence.

The analytical procedure called for an examination of the basic experiment technique and equipment to assure that the experiment concept was valid in terms of time period, of ability to meet the experiment scientific objective, and of ability to be carried out in the space environment while operating in conjunction with manned spacecraft. In the case of the example astronomy experiment, the telescope system was found to have a resolution capability sufficient to meet the measurement requirements of the experiment objective. Film capacity also was found to be sufficient to meet the objective, but not sufficient to allow for experiment updating or extensions of the observation program. The stability and angular-rate design requirements of the ATM are compatible with telescope requirements, and the basic equipment appears capable of meeting the experiment objective.

The experiment physical interfaces with the spacecraft also were considered and were found to be compatible although experiment operation probably would restrict other activities aboard the spacecraft, at least during critical observation periods. Alternate methods of launch also were considered in this phase of the analysis. It was found that for the astronomy experiment (as well as most of the others), launch of the experiment equipment as part of the basic manned space station complex was preferable to a later unmanned logistics launch and subsequent rendezvous with and transfer to the manned space station. The physical size of most of the experiments makes the latter alternative undesirable. Techniques for the in-orbit transfer of large experiment modules in a safe manner still require some development.

Experiment Functional Flow

A major portion of the analytical effort was devoted to analysis of the in-orbit experiment procedures for the purpose of identifying the EVA requirement. In order to proceed in a logical and systematic manner, the analytic approach used required description of the situation in terms of a functional flow diagram to identify the functions making up the set of experiments activities. Such a flow diagram describing the major functions required to conduct the in-orbit astronomy experiment is illustrated in Figure 2. Each block in the illustration represents a major function which must be completed to perform the experiment. The notations above the function blocks indicate inputs to that block; the notations below indicate discrete outputs which may remain outputs at this point or become inputs to the next block. This chart is a first level functional flow, and it indicates only gross functions. While it indicates the presence of man, the astronaut/observer for the in-orbit experiment and the principal investigator on the ground, this level of functional flow is insufficient to indicate the detailed functions of either man, particularly the astronaut/observer in orbit. With this type of analytical technique it is

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OPERATIONAL CONCEPT

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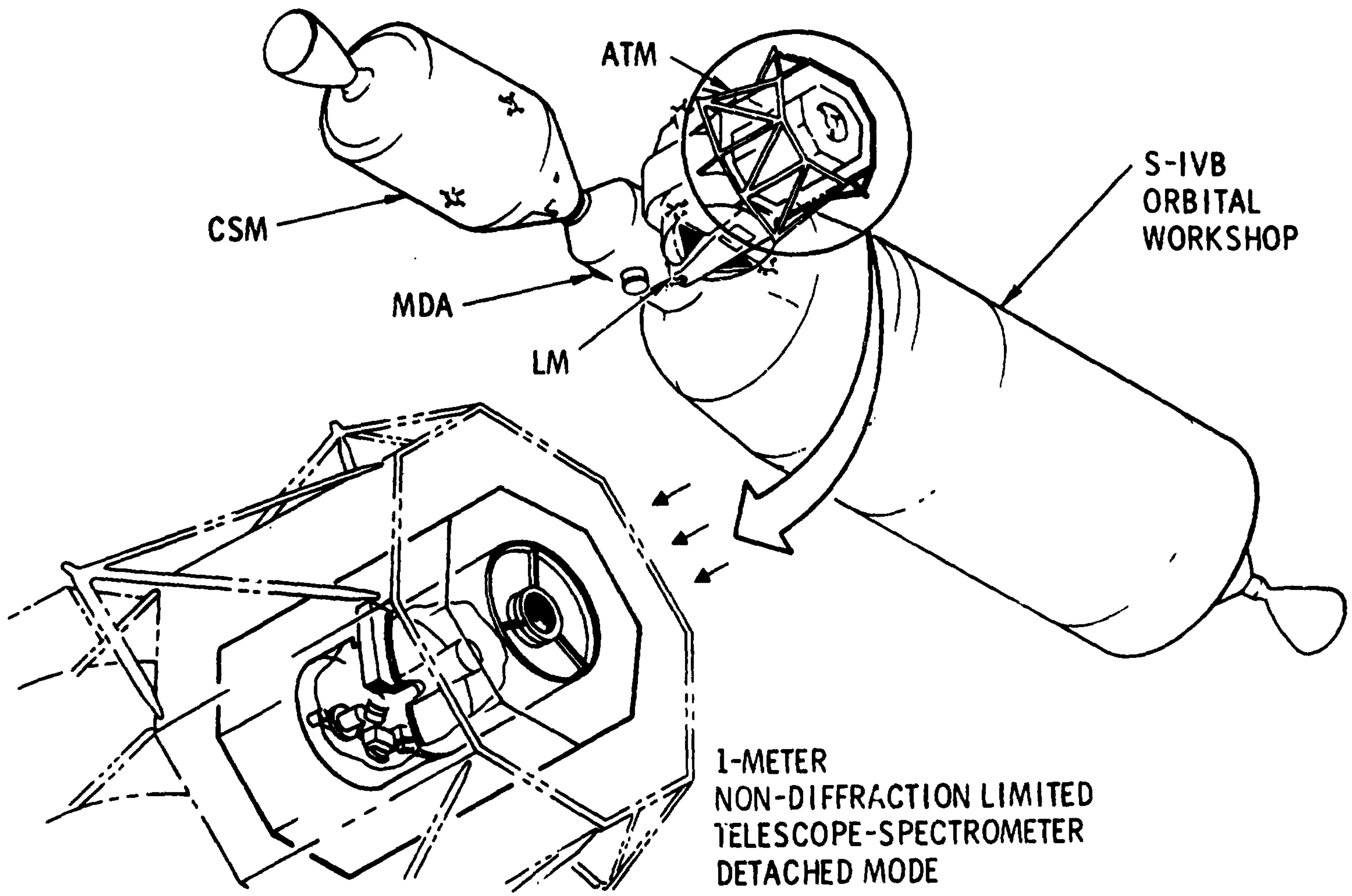


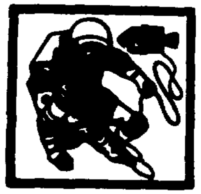
possible to take a discrete function block or sets of blocks and reduce them to sublevel functional flows. For example, the major function entitled Conduct Observation Program may be broken down to a next lower level as indicated in Figure 3. This illustration indicates that a discrete set of functions at the second level must be accomplished to conduct the observation program. It consists of such functions as aligning the spacecraft, alerting the crew, activating the telescope experiment, acquiring and verifying the target for observation, conducting an observation program, monitoring equipment observation, and retrieving data. In this example diagram, the EVA requirements first appeared at the Retrieve Data operation or at the third level in the functional analysis of the experiment. Actually, in some experiments it was necessary to go to third and lower-level functional flows before the potential EVA requirements were identified.

Once the EVA had been identified, the sequence of experiment procedures, including the tasks involved in the EVA, were delineated sequentially in a form similar to a timeline. Estimates of task time, however, were not made at this point.

Verification of EVA Requirements

Up to the point of identifying the potential EVA requirements, the analyses were conducted by technical personnel who were familiar with the scientific objectives of the experiment as well as with the experiment equipment. With the identification of a potential EVA requirement, an additional set of analysts was added to the investigative team. These analysts were systems-oriented personnel familiar not only with experiment procedure, but with the space station system, other experiment systems, and mission-operational requirements. The systems analysts, in conjunction with the technical personnel re-examined the experiment procedures, particularly the EVA requirement, to assure that the requirement was valid and rational. The functional procedures were judged from the standpoing of using an automatic mode rather than EVA, retaining the EVA mode in either its original or modified form, or using a combined mode. In all cases, the analysts were seeking effective use of space-station equipment as well as experiment equipment and EVA astronaut time. At the completion of this analysis, a preferred mode selection was made.

The overriding criteria in the EVA rationale portion of the analysis was maintenance of the experiment scientific objective and astronaut safety in EVA. During this analysis, the analysts were free to modify experiment procedure, to modify experiment equipment, and even to modify mission operations if deemed necessary. In addition to maintenance of scientific objective, other criteria for selection or rejection of the EVA mode were used. For the auto mode (IVA), the criteria were to minimize EVA interruption, EVA design provisions, and astronaut workload. For EVA, the criteria were to simplify experiment hardware, to minimize the experiment-spacecraft interface, to recognize experiment update potential, and to consider experiment maintenance. An important criterion favoring the automatic mode was that of avoiding the disruption of other mission activities caused by engaging in EVA. It was recognized in the early period that EVA would involve at least two men: a prime EVA astronaut and, for reasons of safety, a back-up astronaut. Further, considering that all space station personnel probably would be involved in both pre- and post-EVA operations, approximately 8 hours of



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EXPERIMENT PROCEDURES
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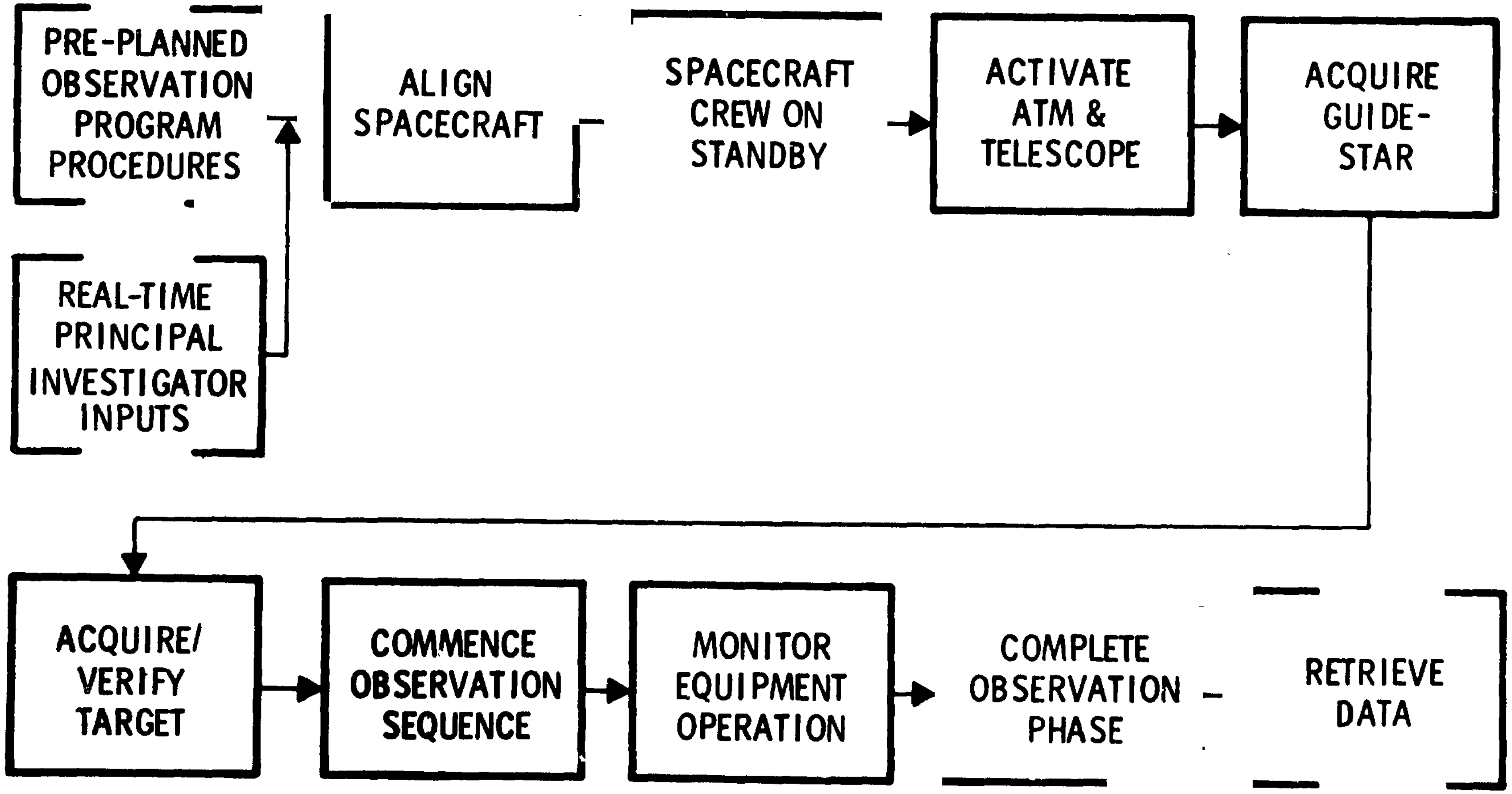
CONDUCT OBSERVATION PROGRAM



a mission day could be taken up by a single EVA. Minimizing EVA, therefore, becomes an important factor in considering the cost and efficiency of experiment operation. For example, one of the bioscience experiments involves the deployment and subsequent retrieval of a series of spore containers, and as initially conceived, required eight EVA's to complete the experiment. As a result of the EVA rationale analysis, it was found that by judicious use of an experiment airlock in the space station, the spore containers could be deployed in intravehicular in all but two instances, and still meet the experiment objectives. For this experiment, it was possible to reduce the number of EVA's from eight to two (one at midpoint and one at conclusion of the experiment to retrieve the final spore container), thereby achieving more optimum use of space station equipment and astronaut time. Another example is that contained in a navigational interferometer experiment concept. In this initial form, six separate EVA's were required to deploy the experiment equipment prior to the gathering of experimental data. By modifying the experiment equipment and procedures, it was found that deployment of the interferometer could be accomplished automatically, thereby eliminating EVA as a prime experiment procedure. In this instance, however, EVA was retained as a contingency operation for use in the event of failure of the automatic deployment equipment.

Automatic Data Retrieval Considerations

In the case of the 1-meter telescope astronomy experiment, several automatic means of data retrieval were considered. One involved the use of electronic detectors and subsequent transmission of data to the spacecraft via cable or a telemetry link; however, this technique was found to be incapable of meeting data quality requirements of the scientific objective. Another automatic means involved the translation of photographic film over a rather long film path from the telescope instrument section to an airlock adjacent to the Lunar Module. While this technique would have resulted in meeting scientific objectives, anticipated difficulties in transporting film over large distances resulted in a reduction in experiment equipment reliability to an unacceptable level. Consequently EVA was retained as a method of film retrieval primarily for reasons of equipment reliability and overall system simplicity. Further, with EVA as a prime experiment procedure, the possibility for improving the experiment equipment potential through later EVA modification and updating of experiment equipment was added justification for retaining the EVA requirements.

Analysis of EVA Tasks

As a result of the EVA experiment definition analysis, 84 separate EVA's were found necessary to perform the 16 representative scientific and technical experiments. The EVA functional tasks for many of these were found to be similar; whereas for others they were found to be unique but essential to a single discipline or even a single experiment. From the 84 separate EVA's, approximately 30 typical EVA's were selected for further analysis of the discrete astronaut functions required. This EVA operations analysis of the EVA tasks to identify the EVA functional performance requirements and EVA equipment requirements. The analysis was performed at a level of detail which included all steps required by the astronaut to accomplish a given task and concerned basic functions, such as egress/ingress, translation, and work performance that must be performed. For each such function, the necessary technique and EVA equipments were specified. Similarly, because there are various ways to perform an individual function, the level of astronaut performance for each option was specified also.

Because of the large volume of descriptive detail involved in this analysis, it was necessary to develop a special technique for handling and analyzing the data. The analytical method consisted of identifying basic sets of EVA functions, subfunctions, techniques required to accomplish the functions, equipment necessary to perform the EVA/functions/technique and, finally, a set of gross performance measures. These functions, techniques, etc., were grouped into logical combinations that could be used to describe a discrete EVA activity. Each of the descriptive groups, called building blocks (BB's), contained one descriptor for each basic function, technique, equipment set, etc. Each such group or building block was assigned a decimally coded number, the digits of which identified the descriptors contained in the building block. For example, there were found to be three basic functions involved in EVA; therefore, the first digit in the numerical code was used to identify the basic function; i.e., 1.0, egress/ingress; 2.0 translation; 3.0, work performance. The second digit identified the subfunction, and so on. The decimally coded number 3.3.2.3, for example, defines an EVA work performance function requiring a two-hand application of a heavy force (25 to 100 pounds) by means of a hand tool at a worksite where the work reach distance is no greater than four feet and indicates that the astronaut has the use of dual foot restraints and a flexible waist restraint. The code number also indicates that the astronaut must be supplied with a spacesuit and umbilically supplied life support system.

After the EVA operations analysis was completed, the building blocks were utilized to generate the overall EVA requirements, which were categorized according to a requirement that the astronaut have a certain capability to perform functions and to the requirements for EVA equipment that permit the astronaut to perform the functions. These requirements were then statistically analyzed to determine the frequency of occurrence for each requirement according to the number of times a capability was required in the performance of the representative EVA tasks. This statistical analysis was used as a basis for establishing future EVA requirements in terms of techniques and equipments.

The building blocks were structured in such a manner that all logical combinations of function, technique and other EVA required descriptors were included. Eighty-eight building blocks were assembled to describe all possible EVA's; however, the EVA operations analysis revealed that only 50 building blocks were actually required to satisfy all the EVA support needs of the 16 representative scientific and technical experiments. Furthermore, by substitution of building blocks to eliminate those less frequently used, it was found that 22 building blocks accounted for more than 90 percent of EVA usage. Of these 22, 4 involved the egreee/ingress function, 8 the translation function, and 10 the work performance function.

EXPERIMENT EQUIPMENT DESIGN

The experiment definition and EVA operations analyses lead to some tentative conclusions regarding the design of earth orbital scientific and technical experiments requiring EVA support.

For example, a review of the eight most frequently used translation building blocks indicated that the greatest usage involved translation of the EVA astronaut only; and the second most frequently used involved translation of an EVA astronaut plus cargo of moderate size (50 lbs mass). In both cases the translation distances involved were up to a maximum of 60 feet.

Similarly, an examination of the building blocks involving the work performance function, reveals that the most frequent worksite actions are position and re-positioning activities of the astronaut and of the equipment the astronaut is either using or handling at the worksite. It is no surprise, then, that the most common EVA equipments used at the worksite are restraints, both astronaut restraints and equipment restraints. The type of astronaut restraint condition most frequently required at the worksite involves a flexible/variable waist restraint plus foot restraints.

On the basis of this type of information derived from analysis of astronaut EVA functions as represented by the most frequently used building blocks, some tentative guidelines for experiment design can be formulated. For convenience the guidelines are keyed to the two basic EVA functions involved, i.e., translation and work performance. In addition, the guidelines are influenced by EVA safety precautions which are based on another phase of the study concerning astronaut safety and rescue in EVA. In general, the basic groundrules are safety and minimization of required EVA astronaut work effort.

Translation

Manual translation paths, those from the point of egress to the experiment equipment as well as those around the equipment, should by the shortest practical distance and must be free of interfering structure and obstructions which would cause excessive effort in the translation or which could cause entanglement or difficulty in tether and umbilical management. Similarly, the path must be free of sharp edges which could abrade, tear or otherwise damage the astronaut's space suit, life support system or other equipment. Also, from the standpoint of safety, the translation path of the prime EVA astronaut should be completely in view of the backup EVA astronaut stationed at or near the point of egress.

The path should contain pre-installed handholds and handrails. The requirement that the astronaut use portable handholds or handrails results in a time consuming translation, leaving too little useable time at the worksite.

Cargo carrying requirements, especially over long distances should be kept to a minimum. If such transfer is necessary, the cargo should be kept small in size and volume. An EVA astronaut is heavily encumbered with spacesuit, life support system and related essential EVA equipment. There is little room available on the astronaut's person for harnesses, slings, or other devices for carrying cargo. In the design of experiment equipment, serious consideration should be given to stowing spare modules, special tools and other items for use in EVA at the equipment worksite rather than elsewhere in the space station for subsequent transfer to the worksite during EVA.

Worksite

Like the translation path, the worksite area must be free of rough edges and surfaces which could abrade or damage the astronauts essential EVA equipment. The site must be accessible, be adequately illuminated, and must provide the necessary astronaut and equipment restraints to enable the astronaut to position himself in the most advantageous work position. The restraints should be pre-located at the worksite with sufficient flexibility to enable the astronaut to reposition himself. Four-foot lateral repositionings were found to be the maximum required in the study with two-foot repositionings being the most common. However, virtually all the astronaut repositionings also involved equipment repositionings.

The accessibility of the worksite should be such that the astronaut is not required to make more than a full arm reach during work performance. Similarly, waist bending motions and head and shoulder reach-in should be avoided largely because of the restricted mobility of the space suit and the danger of striking the life support system back pack against the opening.

Obviously, the worksite work performance activities will have to be limited to performance levels which are within the capability of the space suited EVA astronaut's limited manipulative dexterity. For example, maintenance and sub-assembly replacement functions should be at the modular interchange level with minimum fastening/unfastening activity required for removal and replacement. Modules or parts requiring accurate alignment or placement should be provided with guide rails or locating pins.

In designing equipments for scheduled maintenance (instrument or modular interchange), consideration should be given to designing the equipment to reduce the number of discrete work performance actions required to perform the maintenance. Rather than design a replacement module such that electrical connections have to be made and broken separately, a preferable approach would be to design the connection as an integral part of the module such that the electrical connection is made when the module is seated in position and fastened down.

Not all of the experiments examined in the analysis required the maximum time available for EVA. The EVA requirements for the various experiments ranged from slightly less than one-hour up to the full projected EVA capability of the EVA astronaut (3 to 4 hours). However, each separate EVA, even the shorter ones, require two men (primary EVA astronaut and a back-up EVA astronaut), and both men are involved in approximately 2 hours of pre-and two hours of post-EVA procedures (equipment donning, doffing, check-out, etc.). A nominal two-hour EVA, then, actually involves a minimum of two men for a period of 10 to 11 hours or approximately half a mission day. In the interests of utilizing mission time most efficiently, it is desirable that the EVA's be of maximum time duration and of fewer number than the other way around. This indicates that the scientific/technical experiments requiring EVA support, especially those requiring relatively short EVA's should be sufficiently flexible in design to allow the experiment integrators and mission planners some latitude in the scheduling of the EVA. This type of experiment flexibility will enable several separate EVA support requirements to be accomplished in one single EVA of maximum duration.

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A SIX-DEGREE-OF-FREEDOM SIMULATION TECHNIQUE FOR EVALUATING SPACE MOBILITY AIDS AND TASKS

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SUMMARY: A simulation technique has been developed to evaluate space mobility aids and tasks. The six-degree-of-freedom, computer-controlled, moving-base simulation is used to analyze a variety of mobility aids as applied to specific tasks and to make detailed analyses of an individual mobility aid as applied to elementary tasks that, in combination, make up more complex tasks. Task-related hardware (cameras, tools, mockups) are integrated into the simulation.

INTRODUCTION

Ground-based simulation has and will continue to have a significant role in evaluations of space mobility aids and in analyses of experiment tasks requiring astronaut mobility and stabilization. For the purposes of this paper, a mobility aid is defined as any device that aids an astronaut's mobility while performing extravehicular activities (EVA). Thrusted maneuvering units (backpacks, handguns, jet shoes), vehicle-powered aids such as serpentine actuators (Serpentuators), and passive devices (handrails, tethers) are, therefore, covered by the definition.

Problems that require analysis and evaluation of mobility aid applications present themselves, typically, in two ways. In the first category, an EVA mobility task is defined in terms of an experiment goal. For example, an astronaut must leave the Orbital Workshop (OWS), travel to the LM/ATM, remove and replace film cassettes, and return to the OWS. The problem here is to determine the best mobility aid to use by comparing their performance, as established in a proven simulation, as well as their subjective desirability and mission integration characteristics.

The second category of problems arises when a maneuvering unit has been defined, and possibly built, and its overall performance needs to be established. The problem here is to perform a ground-based simulation with sufficient detail and accuracy to support believable predictions of performance in space.

This paper discusses the Martin Marietta Corporation's six-degree-of-freedom, moving-base simulation and its application to these problems. The simulation concept incorporates realistic dynamics, structures, tools, and supporting hardware. A test subject, riding on the simulator carriage, undergoes translational and attitude dynamics duplicating those he would encounter in zero-g while using a particular mobility aid. The position servos on the carriage and gimbals are commanded by a hybrid computer that determines, through inputs to its program, the appropriate inertial responses to test subject maneuvering unit commands, limb motions, and vehicle-contact forces and torques. Limb motions and vehicle contacts are measured, respectively, by a Limb Motion Sensor (LIMS) and load cells.

Virtually all dynamic parameters in the simulation are accessible at the computer. For each study, the appropriate parameters are selected, processed, and recorded in formats that minimize data reduction.

Following the discussion of the mobility aid problem areas given above and how Martin Marietta's approach can aid in achieving their solution, the paper presents a description of the first program, a "Jet Shoes" evaluation, that utilized this simulation technique. This material is followed by a description of the methods used to integrate structural environments, such as portions of the OWS, and experiment hardware, such as powered maneuvering unit mockups, worksites, and motion picture cameras, into the constraints of this dynamic simulation.

MOBILITY AID APPLICATION PROBLEMS

As stated in the Introduction, analyses of the application of mobility aids to EVA tasks generally are spawned when (1) a mobility aid must be chosen for an assigned EVA task or (2) the overall potential performance of a mobility aid must be established. The following two sections describe examples of these problems and the simulation data that is required to aid in their solution.

LM/ATM Film Cassette Removal/Replacement

In current plans for AAP missions 3 and 4 (required to orbit and man the cluster configuration with the LM/ATM), four EVA's are scheduled for LM/ATM film cassette handling. In each EVA, seven cassettes must be removed from the ATM canister and, in the first three EVA's, the cassettes must be replaced by cassettes stored in the LM interior or the Crew Provisions Module. As shown in Figure 1, the astronaut will travel between the airlock module, A, and work stations B and C on the LM/ATM.

Keeping in mind that this task is an example and that a similar task in the future will in all likelihood have different vehicle and mobility constraints, it is possible to envision using any of the previously-mentioned mobility aids in transporting the astronaut along the paths shown. In order to create a more general example, assume that any of the mobility aids can be incorporated in the mission. Looking at these aids in a general sense, a thrusted maneuvering unit would give the most complete freedom of motion. Handrails or a Serpentuator would be more restrictive, in terms of available maneuvering volume, but would provide positive physical attachment to the vehicle and a controlled path relative to it at all times. In the future, EVA tasks may also have to take into account the possible availability of a material handling

FIG. 1 - LM/ATM CLUSTER CONFIGURATION

device mounted on a maneuverable vehicle or Serpentuator. Such a device would require no EVA and might afford greater overall physical protection for the astronauts.

In this generalized, hypothetical example, assume that typical constraints exist such as (1) a limited amount of EVA time can be used, (2) a potentially marginal amount of maneuvering unit fuel will be available, (3) a stabilized, long-term experiment on the cluster requires that the EVA astronaut keep his forces and torques on the cluster below some known level, and (4) the astronaut must maneuver around and between structures such as solar panels without contacting them. In this case, either a dress rehearsal in space or, and more preferably, a proven ground-based simulation must provide the time, fuel, vehicle perturbation, and position controllability data needed to aid in selecting the proper mobility aid.

Overall Performance of a Mobility Aid

The second category of mobility aid application studies consists of detailed analysis of the overall mobility performance of a particular device. Solving this type of problem with a simulation requires a generalized program that examines each facet of a mobility aid's performance characteristics in detail.

To accomplish this analysis for thrusted maneuvering units, EVA mobility tasks can be broken down into their basic elements and all of the elements grouped in a list of basic maneuvers such as the one given below. Test subjects would then be given the appropriate maneuvering aid and required to perform the maneuvers repeatedly. The detailed, reduced data from this simulation program would allow analytical construction of complex EVA mobility tasks through sequencing of basic maneuvers. The following is a typical list of basic maneuvers:

Single Axis Attitude Control
From a stationary position, the test subject makes an attitude change of a specified size about one of his principal axes and stabilizes in his new orientation. He should minimize his translations and non-specified rotations. This is repeated for each of his three principal axes separately.

Three Axis Attitude Control
This is the same as single axis except the specified rotation does not coincide with one of the subject's principal axes. Therefore, two or three axes of control must be exercised.

Transfer Initiation
Starting from a stationary position, the test subject is given a fixed distance in which to establish a specified velocity vector and start coasting toward a target. Prior to entering the coast phase, the subject should act to reduce his tumbling rates to a minimum while keeping the target in view.

Braking
Starting with a velocity vector toward a target, the subject arrests his approach and comes to a stabilized holding position at, but not contacting, the target.

Tumble Recovery
The subject is given a tumbling rate about an axis not aligned with his principal axes. He then brings his attitude rates to zero as quickly as possible.

Moving Inspection
Starting from a stationary position, the subject moves along an S-curved surface without contacting the surface. A visual task is provided so that he must stay within some predetermined distance from the surface to complete the maneuver successfully.

Translate Around Corner
Starting with a velocity vector parallel to a surface, the subject continues along the surface without contacting it, turns a right angle corner, and proceeds along the new surface, still without contact.

During thest tests, recorded data should be obtained on time, total impulse, and range of astronaut dynamics (position, rate, and acceleration histories in translation and attitude).

Repeating this process with a variety of maneuvering units and, with minor variations, other mobility aids, would provide a data base for predicting complex task performance profiles. The data should include, where applicable, not only the astronaut dynamics, but also the self-induced torques and vehicle contact forces and torques. The latter items are particularly important if the astronaut's motions interact with and place a working load on a vehicle or mobility aid stabilization system.

SIMULATION OF MOBILITY AID APPLICATIONS

The Martin Marietta Space Operations Simulator (SOS) facility provides a basis for studying a wide range of space problems: EVA, IVA, space to ground tracking, space to space tracking, rendezvous, docking and Serpentuators.[1] Space hardware such as optical radars can be evaluated. This section includes a description of the SOS facility and the technique used to evaluate mobility aids in the facility. Also, a discussion of a data generation and analysis capability is included.

Facility

Simulation experience, gained from studies of space problems over the past several years, has provided the base for determining the present configuration of the Space Operations Simulator facility as shown in Figure 2. Simulator

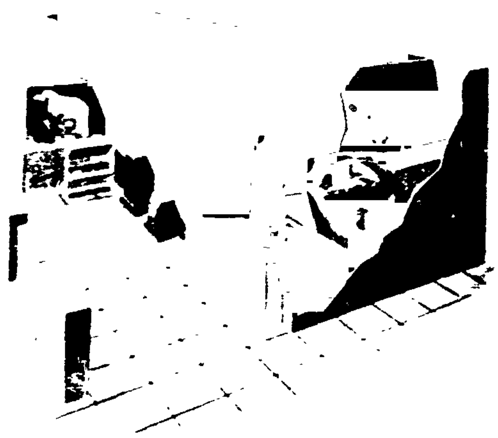

FIG. 2 - SIMULATOR FACILITY

design and performance requirements have been considered and presented previously.[2] The Space Operations Simulator facility consists of five interdependent parts -- the moving base carriage, the control capsule, simulation instrumentation, the test monitor station, and the hybrid computational equipment -- as shown in Figure 2.

Moving Base Carriage

The Martin Marietta moving base carriage (Figure 3) utilizes

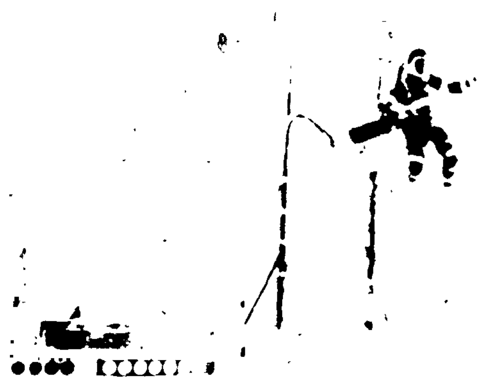

FIG. 3 - MOVING BASE

the "powered" simulation approach rather than the "free-motion" approach. A 90 by 32 by 24 foot room, providing an 8640 cubic foot maneuvering volume, houses the moving base which is servo-driven in three translational axes and three rotational axes. The base of the carriage translates the length of the room on three rails and is driven by four one horsepower AC motors which engage two gear racks mounted on the floor. The vertical pedestal translates on rollers and rails laterally on the base structure and is driven by two, one horsepower AC servomotors. The gimbaled head located on the front of the pedestal is supported by a set of negator springs and a counterbalance weight. This system effectively counterbalances the weight of the gimbaled head and its payload. Two one-quarter horsepower DC motors, which engage two vertical gear racks on the front of the pedestal, provide the servopower for the vertical translation. The gimbaled head (Figure 4) has been designed to provide maximum safety and freedom of motion for the test subject. Motors and gear drives are enclosed in the structure of the gimbals. The gimbal sequence of roll, pitch and yaw was selected because it allows all three drive

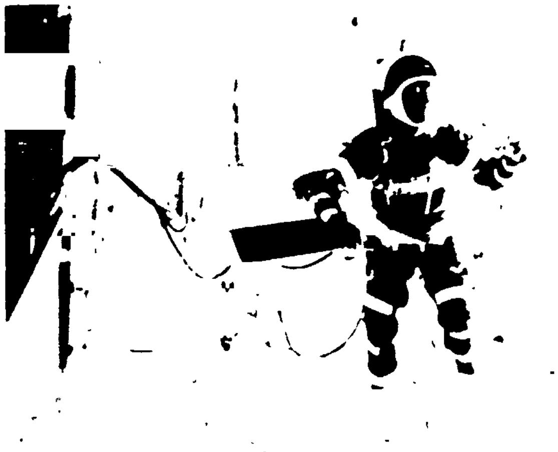

FIG. 4 - GIMBALED HEAD

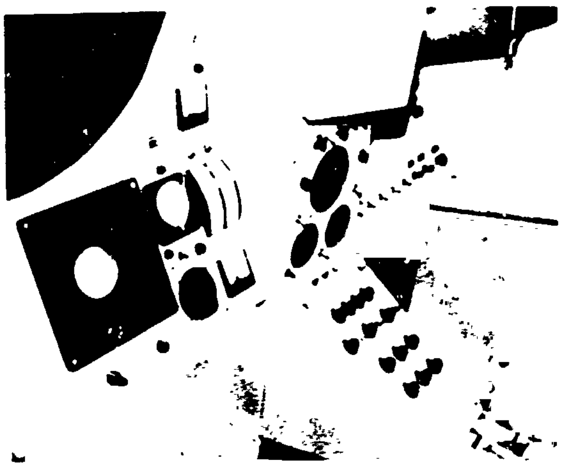

FIG. 5 - CONTROL CAPSULE

axes to nominally pass through the c.g. of the test subject. Thus, counterbalance and overhanging moment problems on the gimbal axes are minimized. Each gimbal is driven by a one-quarter horsepower DC motor. The overall weight of the moving base carriage is 5000 lbs. Simulator performance is shown in Table 1.

Control Capsule

Several control capsules have been in operation in the SOS laboratory; however, the one that seems to provide desired flexibility is shown in Figure 5. This mockup has a two-man capacity and contains sufficient display and control instruments to allow several types of space mission tests to be run. These tests include both manual and automatic piloting and ground tracking as well as aerial tracking. Normally, a flight requires the use of both the out-the-window scene and several instruments.

Simulation Instrumentation

Limb Motion Sensor (LIMS) - For many of the manned simulations (EVA, IVA, Maneuvering Units, etc.) it is necessary to compute c.g. shifts and inertia changes as a function of limb position. Also,

TABLE 1 - MOVING BASE PERFORMANCE

	Longitudinal	Lateral	Vertical
Travel (ft)	60.0	$\pm$ 6.0	$\pm$ 6.0
Velocity (fps)	3.0	3.0	3.0
Acceleration (fps^2)	6.0	3.0	3.0
	Roll	Pitch	Yaw
Travel (rad)	$\pm$ 1.0	$\pm$ 3.8	$\pm$ 3.1
Velocity (rad/sec)	2.0	2.0	2.0
Acceleration (rad/sec^2)	8.0	8.0	8.0

in the case of some maneuvering units, the orientation of a thrust vector relative to the test subject's torso must be known. The LIMS (Figure 6) was designed to

FIG. 6 - LIMB MOTION SENSOR

fill this need. By monitoring the orientation of each of the body pivot points and knowing the length of the connecting links, the dynamic effects of c.g. shifts, inertia changes and self-induced rotations can be computed and introduced properly into the problem. Also, the orientation and location of thrust vectors for simulated handguns can be computed using LIMS extensions.

Load Cell Array - For EVA/IVA simulations where the test subject is in contact with the simulated space station, the contact forces and moments must be measured in order to be able to simulate the problem. In the SOS facility, an array of load cells is used to measure the contact conditions.

The worksite or vehicle mock-up is mounted on a load-sensing platform which is an equilateral triangle with two load cells at each apex. The load cells are mechanically attached to the platform with pairs of flexure joints. The total configuration is symmetrical about any axis that contains the centroid and any apex of the triangle.

Test Monitor Station

Simulator operation is controlled by an engineer at a test monitor station. From the monitor panel he has control of the servo-power to the moving base and the computer control signals. Also, a wide range of test points can be monitored. The monitor station provides another safety check on the proper functioning of the simulator. The engineer can override the computer and remove power from the moving base at any time.

Computational Equipment

The simulator is supported by two computer systems. One system contains four EAI 231R computers located as an integral part of the facility. The other consists of a hybrid computer (Figure 7) in-

FIG. 7 - HYBRID COMPUTER

stallation consisting of three EAI 8800 analog computers, one EAI digital computer, and one EAI 8930 linkage subsystem.

Technique

The information flow of the mobility aid simulation technique is shown in Figure 8. A test subject is suspended from the inner gimbal ring of the moving base simulator (Figure 4). The test subject can maneuver in the room using the controls of a simulated maneuvering unit. A math model of the maneuvering unit including switching logic is contained in the computer program. A Limb Motion Sensor (LIMS) attached to the test subject's body is used to monitor the position of the various body segments. Information from the LIMS, along with thrust initiate signals from the maneuvering unit controls, is used by the hybrid computer to determine continuously the positional servo-commands for the moving base and gimbals. The computer program contains the equations of motion and approximates the test subject by a model man consisting of nine rigid body segments. When the subject comes in contact with the worksite, the contact forces and moments, as measured by a set of load cells, are sent to the computer. The effects of the contact forces and moments on the test subject's motion are computed by the hybrid program and the simulator positional commands are modified accordingly. Thus, the test subject moves dynamically about the room or worksite as if he was working in zero "g".

Data Generation and Analysis

Since all the dynamic parameters are contained at some stage of computation in the hybrid computer program, they are available for data generation and analysis. Typical parameters which can be monitored during a simulation are: body rates, body attitude, body velocity, body position, contact forces and moments, thruster signals, and fuel consumption. The objectives of a particular

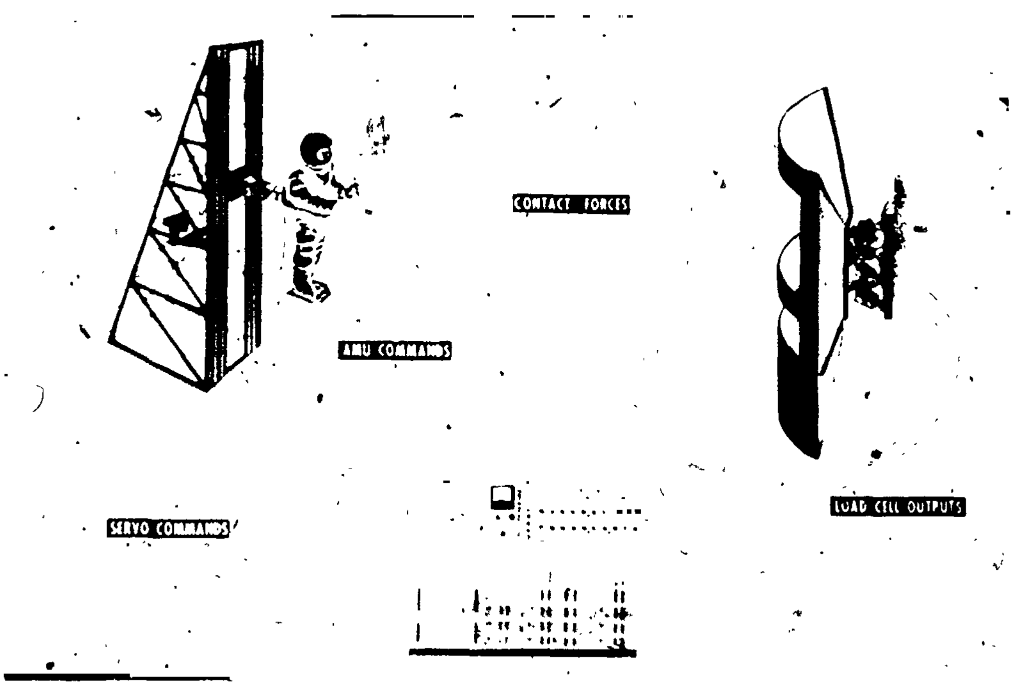

FIG. 8 - INFORMATION FLOW

program determine which parameters should be monitored to allow the desired analysis to be performed. Time histories of the dynamic parameters can be recorded in various ways: digital tape, X-Y plots and strip charts. The digital tape allows for analysis of the data using a separate digital program. When large quantities of data must be analyzed, a digital program is the most efficient method. End conditions can be printed out automatically. These, along with the X-Y plots and strip charts, allow the test conductors to survey the subject's performance during the test runs.

"JET SHOES" SIMULATION

At the time the decision was made to modify the simulation to accommodate maneuvering aid analyses, the simulator was configured for studies of crew motion effects on vehicle stability. Appropriate blocks of equations were added to the hybrid program so that it included the effects of a maneuvering aid. "Jet Shoes" were selected as the first maneuvering unit to be analyzed because of existing interest and because they offered the chance to use the LIMS directly in generating thrust vector information. This initial program was, in effect, a trial run to explore the problems of using this simulation technique for analyzing maneuvering aids.

Subject's Learning Phase

Before the test subject was allowed to fly a test maneuver in six-degrees-of-freedom, he went through a sequence of runs with limited degrees of freedom. This allowed the test subject to gain a realistic feel for the available controllability about each of the attitude axes. Also, he learned the relationship between ankle positioning and the resultant motion when the thrusters were fired.

First the test subject was given only pitch freedom (face up or down motion). Pitch control is obtained by rotating the ankles in a parallel action up or down. An ankles-down motion results in a face-up body rotation. The subject could travel 300 degrees in pitch having a maximum acceleration of three-tenths of a radian per second squared. Next the subject was allowed to practice roll attitude control (front-to-back axis through his mid-section). Roll control is obtained by tucking the ankles in or out. One ankle must be tucked in and the other out. The maximum roll acceleration is one-tenth of a radian per second squared. Next the subject practiced yaw control (left or right rotation). By pitching one ankle up and one ankle down, yaw control is obtained. The left ankle down and the right ankle up produces a yaw motion to the right. The subject could travel 90 degrees in yaw having a maximum acceleration of two-tenths of a radian per second squared.

Following the single degree of freedom attitude control practice, the subject was allowed to practice controlling all three degrees of attitude control simultaneously. This permitted the subject to gain a feel for the degree of coupling between the rotational modes. The attitude modes are essentially decoupled from each other when the feet are properly positioned. This and the single axis practice aided the subject considerably in learning to control his attitude.

To introduce the subject to his translational control capability, the subject was allowed to fly a planar maneuver having one degree of attitude control (pitch) and two translational degrees (vertical and longitudinal). The subject started out in a vertical orientation facing a red vertical target forty feet away. He performed a face down pitch maneuver to point the thrust vector at the target through his center-of-gravity (c.g.). Then the subject fired the thrusters to close on the target. The maximum available acceleration was three-tenths of a foot per second squared. He nominally maintained an attitude with his head pointing at the target during the coast phase. The maximum closing velocity reached was two feet per second with a flying time of twenty-five seconds.

The test subject was also allowed to practice two six-degree-of-freedom test maneuvers. These maneuvers, discussed in the following section, consisted of flying to a target in six-degrees-of-freedom. The subject accumulated approximately five hours of flying time during practice and while making the data runs.

Data Runs

Translating to a target having an initial relative velocity is one of the more difficult tasks that an astronaut will have to perform. Therefore, it was considered important to start the subject out with non-zero initial velocity conditions. The data runs consist of two mission type maneuvers, each repeated five times. The two mission type maneuvers are similar except for the relative position of the test subject to the target. For the one maneuver the target was positioned to the subject's left and for the other to his right.

For the target left runs, the subject was given initial velocity conditions of one-half foot per second downward velocity and one-half foot per second longitudinal velocity. He was required to translate eight feet laterally and forty feet longitudinally to reach the target. He was given an initial yaw condition such that he did not face the target. When the run started, the subject had to alter his velocity so as to close on the target. In six-degrees-of-freedom, he had to yaw to face the target while at the same time arresting his vertical velocity. Then a pitch maneuver was performed and the subject accelerated at the target. The test subject was instructed to fly so that the target was within his reach at the end of the run. Also he was required to have his rotational rates reasonably arrested upon contacting the target with his hand(s). He was allowed to contact the target with the velocity he had accumulated while maneuvering toward the target. Thus, he did not have to perform a braking maneuver at the end of the run. The target right runs had the same initial conditions as the target left runs except the target was on the subject's right. During the forty foot translation to the target, the subject also had to maneuver laterally two feet to his right.

Data

Summaries of the data taken during the test maneuvers are shown in Table 2. Since the target left and target right maneuvers are very comparable, the data from each of the maneuvers should be comparable. The

TABLE 2 - "JET SHOES" SIMULATION DATA SUMMARY

Maneuver	Thrust Time (Secs)	Thrust Pulses Left	Thrust Pulses Right	Translational Velocities X Max Vel (Ft/Sec)	Translational Velocities Y Max Vel (Ft/Sec)	Translational Velocities Z Max Vel (Ft/Sec)
Target Left						
Run 1	15.3	11	9	.8	.2	.3
Run 2	19.9	12	11	1.4	.5	.4
Run 3	13.5	11	10	1.1	.4	.2
Run 4	13.7	10	10	1.2	.2	.4
Run 5	10.3	9	6	1.2	.4	.5
Average	14.5	11	9	1.1	.3	.4
Target Right						
Run 1	14.3	14	10	1.7	.2	.5
Run 2	6.9	9	7	.9	.1	.1
Run 3	20.6	18	16	1.1	.1	.0
Run 4	20.1	18	18	1.4	.1	.4
Run 5	13.3	10	10	.9	.1	.2
Average	15.0	14	12	1.2	.1	.2

Maneuver	Flight Time (Secs)	Rotational Rates Pitch Max Rate (Rad/Sec)	Rotational Rates Yaw Max Rate (Rad/Sec)	Rotational Rates Roll Max Rate (Rad/Sec)
Target Left				
Run 1	78.3	.220	.105	.002
Run 2	63.0	.280	.100	.001
Run 3	68.0	.188	.090	.001
Run 4	61.6	.300	.105	.001
Run 5	69.3	.125	.024	.002
Average	68.0	.223	.085	.001
Target Right				
Run 1	63.0	.250	.080	.003
Run 2	80.3	.150	.045	.003
Run 3	74.5	.280	.070	.004
Run 4	69.0	.220	.150	.005
Run 5	70.5	.300	.033	.004
Average	71.5	.240	.076	.004

data in the table verifies that this is true. The average flight time for a typical run was 70 seconds, reaching maximum velocities of: 1.2 ft/sec in X, 0.3 ft/sec in Y, and 0.4 ft/sec in Z. The average thruster on time during a run was 15 seconds with each thruster being fired an average of 12 times. Maximum angular rates of 0.30 rad/sec were reached. This rate was about the subject's pitch axis as would be expected because a large pitching motion is required to perform the maneuver.

For the target left runs, continuous recordings showing the subject's position in the simulator room were taken. Figures 9 and 10 are plots of these data. Figure 9 shows a history of the subject's vertical position versus longitudinal position. Since the target was vertical and extended from the top to the bottom of the room, the subject could contact the target vertically anywhere over a distance of 10 ft. From Figure 9 it can be seen that the subject contacted the target with a vertical position spread of 8 ft. Figure 10 shows a history of the subject's lateral position versus longitudinal position.

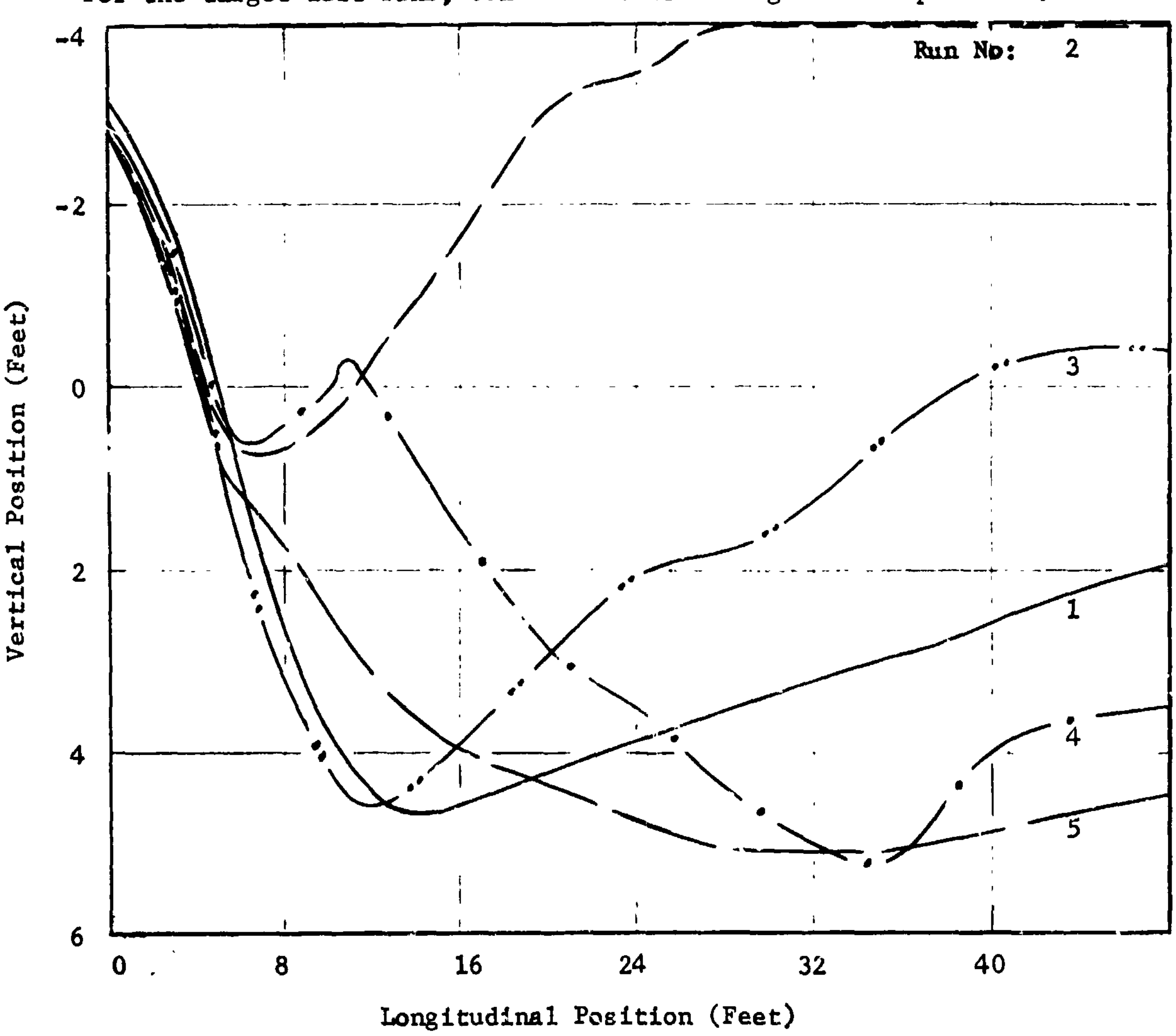

FIG. 9 - SUBJECT'S VERTICAL POSITION HISTORY

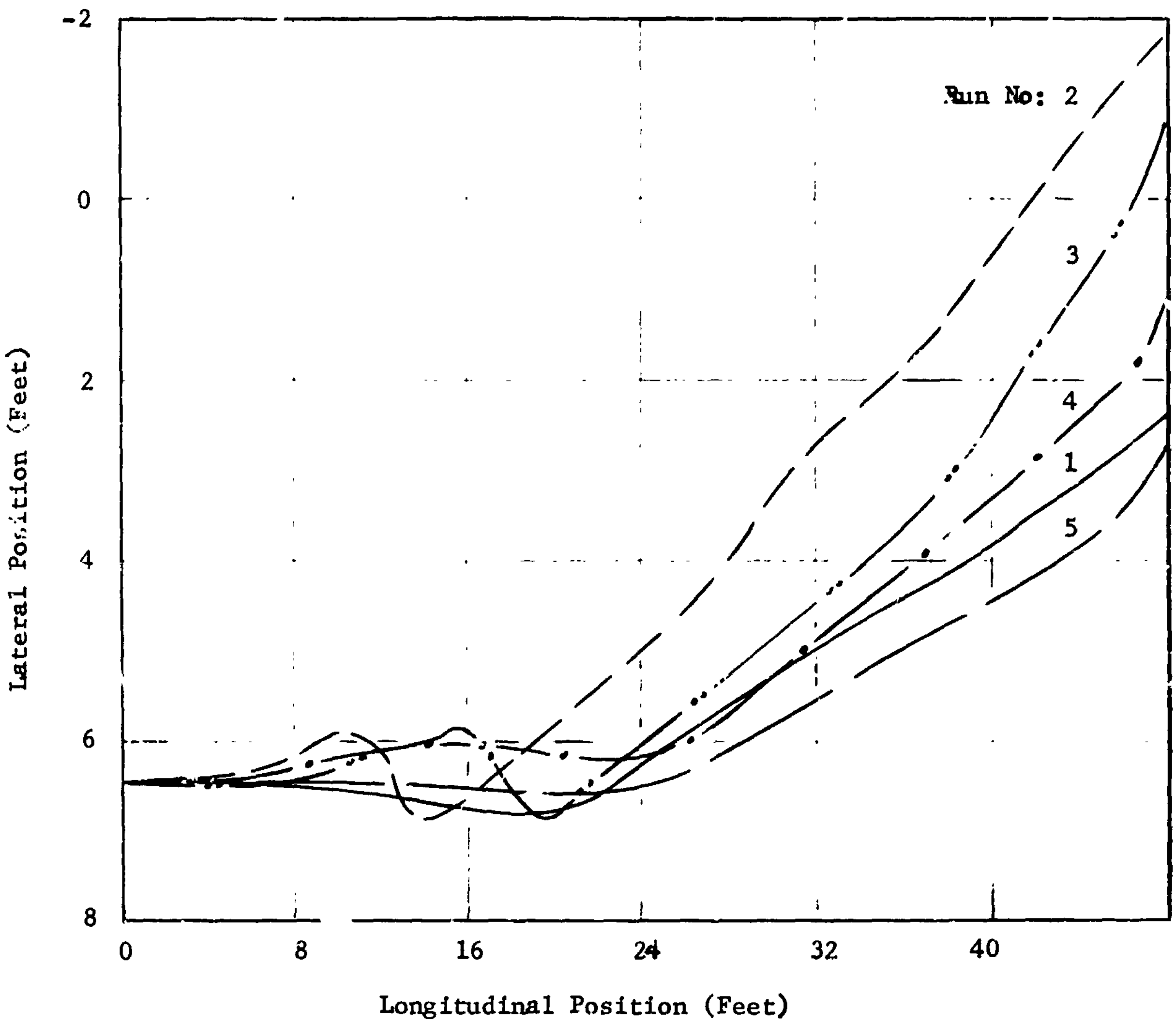

FIG. 10 - SUBJECT'S LATERAL POSITION HISTORY

The lateral position spread at contact was 5 ft. At contact in each run, the subject could reach the target with either his right or left hand. The largest contact velocity, 1.5 ft/sec, occurred in the longitudinal direction. The average lateral contact velocity was 0.6 ft/sec and the average vertical contact velocity was 0.2 ft/sec. The average angular rates at contact were: 0.14 rad/sec in pitch, 0.08 rad/sec in roll, and 0.01 rad/sec in yaw.

INTEGRATION OF RELATED HARDWARE INTO SIMULATIONS

Any mobility aid simulation technique or facility presents unique problems if the actual hardware or mockups of hardware and structural environments are included in the simulation. In the Martin Marietta SOS, operating maneuvering units are not required since all unit characteristics are synthesized on the hybrid computer. Therefore, taking a backpack as an example, only the control arms and mechanisms that extend forward from the subject's torso are included in the simulated backpack.

Mockups of segments of vehicles must be placed three feet or more above the floor of the SOS so that they provide clearance for the moving base. The mockups can, therefore, be up to twenty-one feet tall without contacting the simulation chamber ceiling. If a particular maneuver requires that the subject penetrate into a recessed area of a mockup, the recess must be at least two and one-half feet wide up to a depth of four feet to accommodate the gimbal yoke. At depths greater than four feet, the recess entrance must be at least six feet wide and eighteen feet high to allow entrance of the moving base's pedestal.

One large mockup currently being installed in the SOS consists of a half-section of the OWS forward tank area (cut along a longitudinal plane). The mockup is being installed with the longitudinal axis running vertically and the lower floor is absent to allow clearance for the moving base and pedestal. The test subject will be able to maneuver anywhere within this OWS half-volume.

Any worksite on or within a mockup can be mounted independently on load cells (Figure 11). For an experiment such as M-509, "Astronaut Maneuvering Equipment", where an astronaut will fly various maneuvering units within the OWS, ground-based simulations should include, eventually, both maneuvering unit and vehicle contact effects. The Martin Marietta simulation is capable of providing both of these in a full-scale mockup. This simulation also will lead to ease of installation of

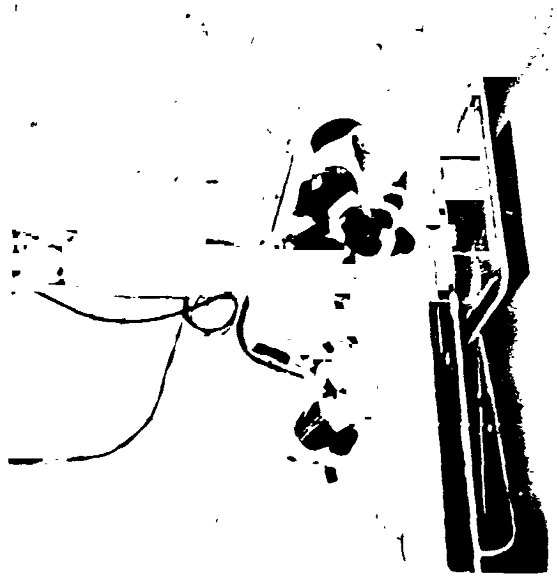

FIG. 11 - INSTRUMENTED WORKSITE

other experiment hardware such as motion picture cameras, astronaut position locators, OWS interior lighting, and equipment bays and racks so that these may be evaluated simultaneously with the mobility tasks.

CONCLUDING REMARKS

As the spectrum and complexity of EVA mobility tasks grows in the future, it is certain that virtually all dynamic parameters in the tasks will come under quantitative scrutiny. Therefore, considerable importance is attached to the development of a six-degree-of-freedom mobility aid simulation that will provide data for accurate prediction of flight performance. The Martin Marietta Space Operations Simulator, when exercised to its full capability, provides an accurate simulation of an astronaut's motions in space. Although the test subject senses the one-g laboratory environment, his motions are purely a function of his maneuvering aid thruster impulses, his contacts with other vehicles, and his limb-motion-induced rotations and inertia changes.

Full-scale mockups fixed in the simulator chamber provide realistic surroundings for mobility tasks. Clearance for the moving base is required behind the test subject but it is generally possible to orient the tasks so that the mockup does not interfere with the moving base. The mockups also provide for inclusion of equipment that is related to, or that might interfere with, the mobility task. The effect of this equipment on the mobility task can then be evaluated.

Since this is a computer-driven simulation, all dynamic parameters are available for processing and recording in appropriate formats. Typically, the greatest data generation problem is selecting an appropriate and practical set of parameters for recording from the total available set.

The initial "Jet Shoes" simulation verified another key factor; the subjective suitability of the simulation. The test subject felt comfortable and capable of controlling the problem after a total accumulated simulation flight time of five hours.

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EXTRAVEHICULAR ASPECTS OF THE AIR FORCE D021 AND D022 ORBITAL WORKSHOP EXPERIMENTS

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SUMMARY: Experiments D021 and D022 are two of the experiments aboard NASA's Orbital Workshop which depend upon EVA for their performance. D021 will evaluate the elastic recovery materials and design philosophy of an expandable airlock; D022 will provide samples of materials which have been chemically rigidized in space.

INTRODUCTION

During the past eight years the Air Force and NASA have conducted considerable research and development on expandable materials and structures technology for space application. This R&D has ranged from basic materials research through fairly sophisticated ground fabrication and demonstration of articles such as lunar shelters, crew transfer tunnels, and solar collectors. Examples of these are shown in Figure 1.

Simpler but less versatile expandable techniques were employed in the Echo and Pegasus satellite programs of the National Aeronautics and Space Administration. The two Echo passive communications satellites were large spherical reflector antennas. Their diameters (100 feet for Echo I and 125 feet for Echo II) were obviously too great for such vehicles as Saturn and Titan 3, required that the structure be expandable.

Pegasus, designed to determine meteoroid flux in near-earth orbits, required a structure with a large surface area to maximize exposed area for meteoroid impact. This satellite had a deployed wing span of 96 feet. Again, because of booster launch limitations, an expandable structural concept was used in its construction.

Although both the Pegasus and Echo programs utilized expandable structures, the type of structure employed was either extremely light weight (3 lb/1000 ft^2 surface area to 7.1 lb/1000 ft^2 surface area) or unfolded mechanically.

Chemically-rigidized and elastic recovery structures have a high degree of potential for application to crew quarters, reentry vehicles, airlocks, and lunar shelters. Therefore, a series of more advanced development-oriented and flight demonstration programs have been

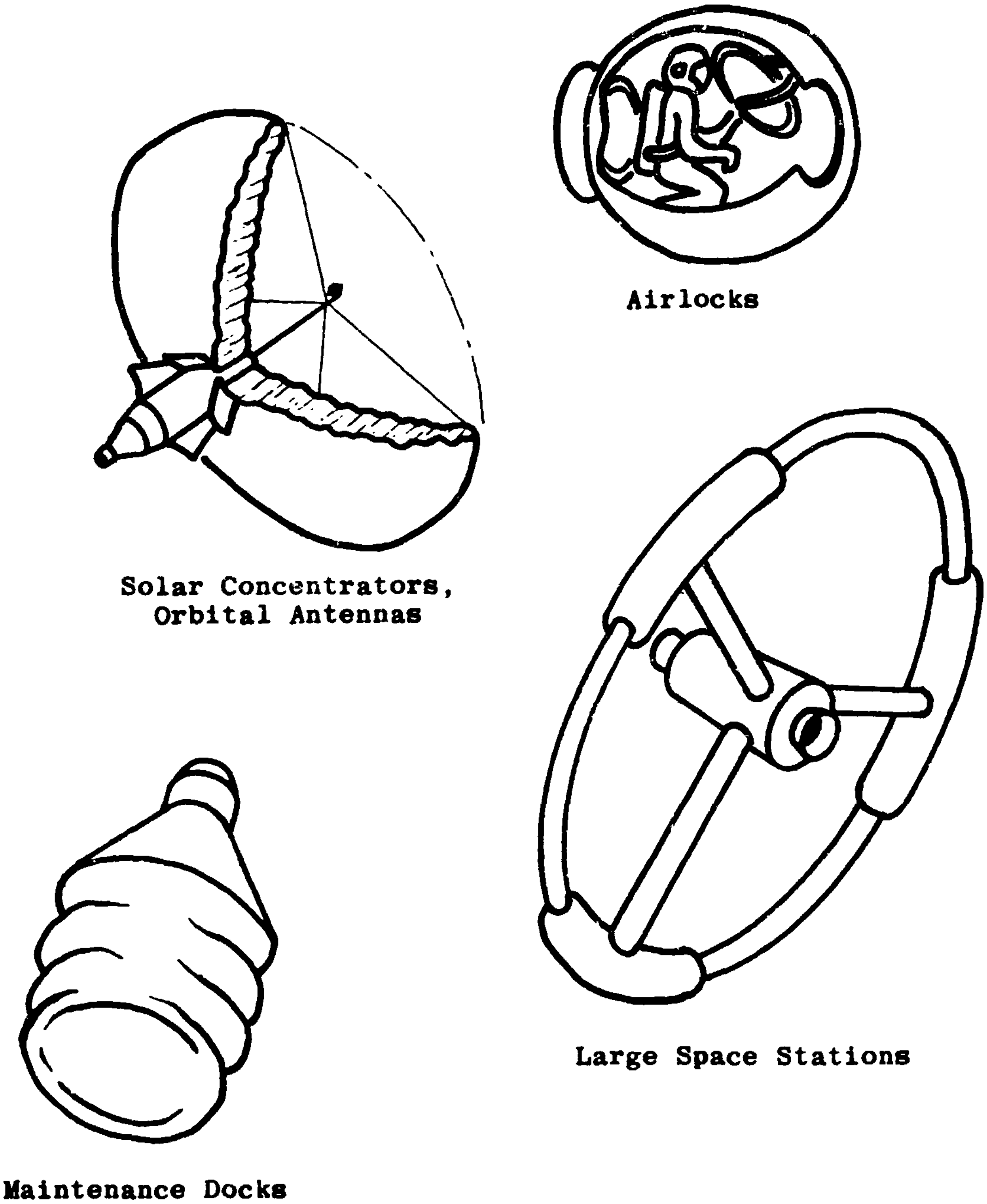

Figure 1
Expandable Structures Applications

planned to provide sufficient expandable structure data to permit systems engineers to utilize technology for the previously mentioned applications.

The D021 Expandable Airlock Experiment and the D022 Chemical Rigidization Experiments are the first examples of these programs. These experiments were defined for the Orbital Workshop (OWS) program and were formally approved by the Manned Space Flight Experiment Board, MSFEB. The orbital workshop presented an early opportunity to execute these expandable structures experiments and had the unique capability of permitting astronaut EVA participation in both experiments. This participation will add measurably to the overall experiment quality and probability of success.

EXPERIMENTS FOR THE WORKSHOP

The potential payoff inherent in expandable structures for space warranted an early evaluation aboard a manned orbiting vehicle. The elastic recovery principle was selected for use in a full-size airlock. Chemically rigidization, because of its greater complexity and sophistication, was not yet suited for use in a complete structure. Rather, an experiment was designed to provide for orbital deployment and rigidization of small, flat panels using two promising rigidizing systems. Both experiments were chosen for the first mission of NASA's Orbital Workshop (Figure 2), a part of the Apollo Applications Program, as Experiment D021, "Expandable Airlock Technology", and Experiment D022, Expandable Structures for Recovery".

OBJECTIVES OF EXPERIMENT D021

The D021 Expandable Airlock will obtain considerable information on the elastic recovery structural concept for simple airlock applications. Specifically, the experiment will (1) ascertain the ability of elastic recovery structures to withstand the boost and launch phase of a typical mission, with subsequent successful deployment in orbit; (2) validate the successful performance of these materials in operational use when subjected to the total orbital environment; (3) evaluate structure packaging techniques and deployment dynamics for a relatively small configuration; (4) evaluate effects of the space environment on elastic recovery materials after prolonged exposure (six months); (5) establish some of the basic design parameters and requirements for elastic recovery airlocks for future manned space laboratories; (6) provide a baseline from which to extrapolate the application of expandable structures technology to other uses, such as crew transfer tunnels, space shelters, maintenance stations, and storage depots; and (7) to demonstrate the compatibility of expandable elastic recovery materials in airlock designs with the dynamics of astronaut ingress/egress. The last objective is related most closely to extravehicular activity and is of major interest here.

The experiment procedures are divided into two phases and require only one EVA. The first phase is deployment and proof-pressure-testing of the airlock, followed by a 15 day space exposure test. This phase will be performed remotely from a control panel mounted in the airlock module. The second, extra-

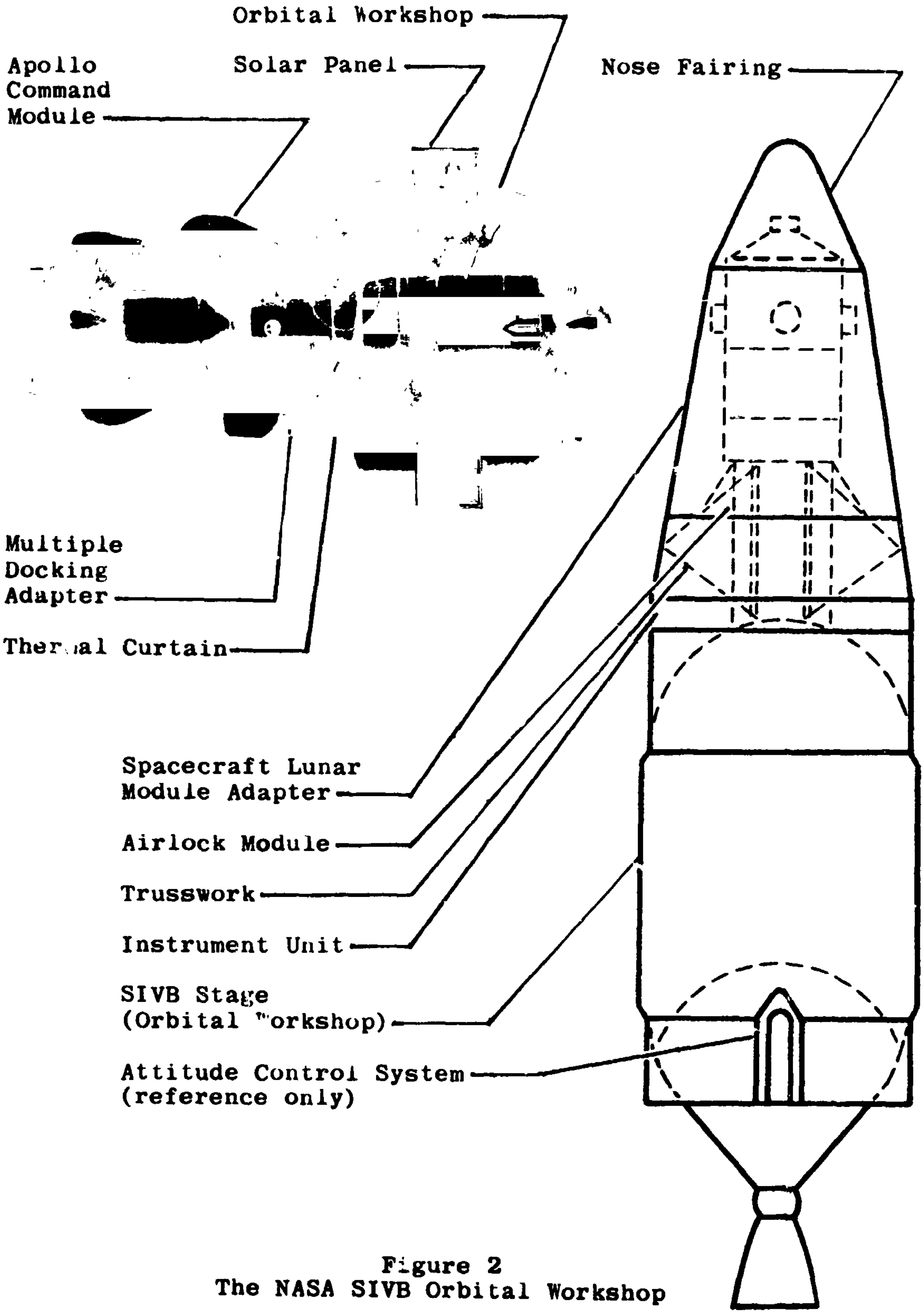

Figure 2
The NASA SIVB Orbital Workshop

vehicular phase will consist of two ingress/egress maneuvers by the astronaut, separated by a complete pressurization cycle. The astronaut will manually open, close, and lock the D021 hatch, but at no time will he be sealed inside the airlock. To conclude this second phase, the astronaut will retrieve several materials samples from the base of the experiment package. These will be placed in a protective container, which will be sealed for return to earth. Similar samples will be retrieved by another astronaut, during the revisitation mission several months later.

Photographic coverage will be either from the hatch of the Apollo Command Module or by a second extravehicular astronaut stationed in the thermal curtain opening. An automatic motion picture camera, mounted outside the thermal curtain is also under study. It is anticipated that the segment of the EVA requiring photography can be accomplished within the sunlight portion of one orbit. An artist's concept of the expanded airlock is provided in Figure 3.

OBJECTIVES OF EXPERIMENT D022

The objectives of the D022 chemical rigidization experiment are similar to those of D021: (1) To demonstrate the technology of chemically-rigidized expandable structures; (2) to establish the astronaut's ability, while working in the orbital environment, to deploy and rigidize an expandable structure; (3) to determine the effects of packing, storage, launch, ascent, and the orbital environment on samples of two types of chemically-rigidized, glass fiber, structural materials, by comparing their properties with those of control panels rigidized and stored on the ground; (4) to establish a baseline of experience and data from an actual orbital test (and possibly to uncover unforeseen problems), from which future developments of materials and structures may be undertaken with greater efficiency and confidence.

Experiment D022 will require two periods of extravehicular activity for completion. A crewman will leave the Workshop via the EVA hatch of the NASA Airlock Module and the thermal curtain opening, and translate to the worksite. Here he will open two sealed canisters, each of which contain four flexible, folded, material samples. These are withdrawn from the canisters on telescoping rods and inflated to the proper shape. The panels in one canister rigidize simply by exposure to the orbital vacuum; the others are cured by internal electrical heating. Two panels from each canister will be retrieved by an astronaut on a revisitation mission for return to earth. It is planned that the remaining panels will be recovered on a later mission, after several months exposure to the space environment. Figure 4 shows the experiment about half way through the deployment phase.

EXPERIMENTS-TO-WORKSHOP INTERFACE

Hardware for both experiments will be mounted on or between the large trusses which strengthen the connection of the Airlock Module to the end of the SIVB stage. These trusses also provide some support for a flexible, protective thermal curtain. (Figure 5 shows this area of the Workshop in greater detail.) A closeable flap in the curtain is located directly above the EVA airlock. The entire D021 airlock

Figure 3
D021 Artist's Concept

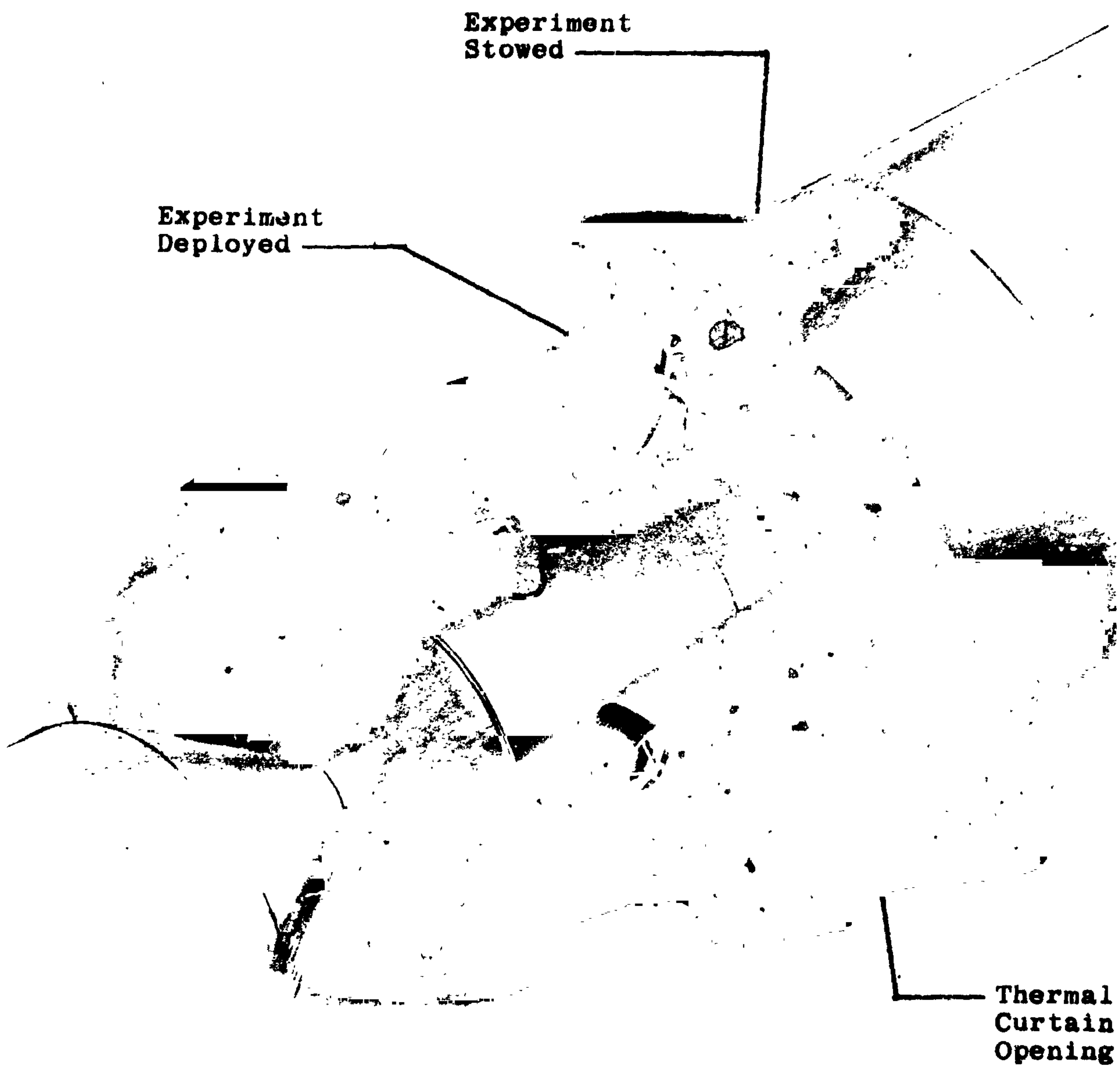

Figure 4
D022 Artist's Concept

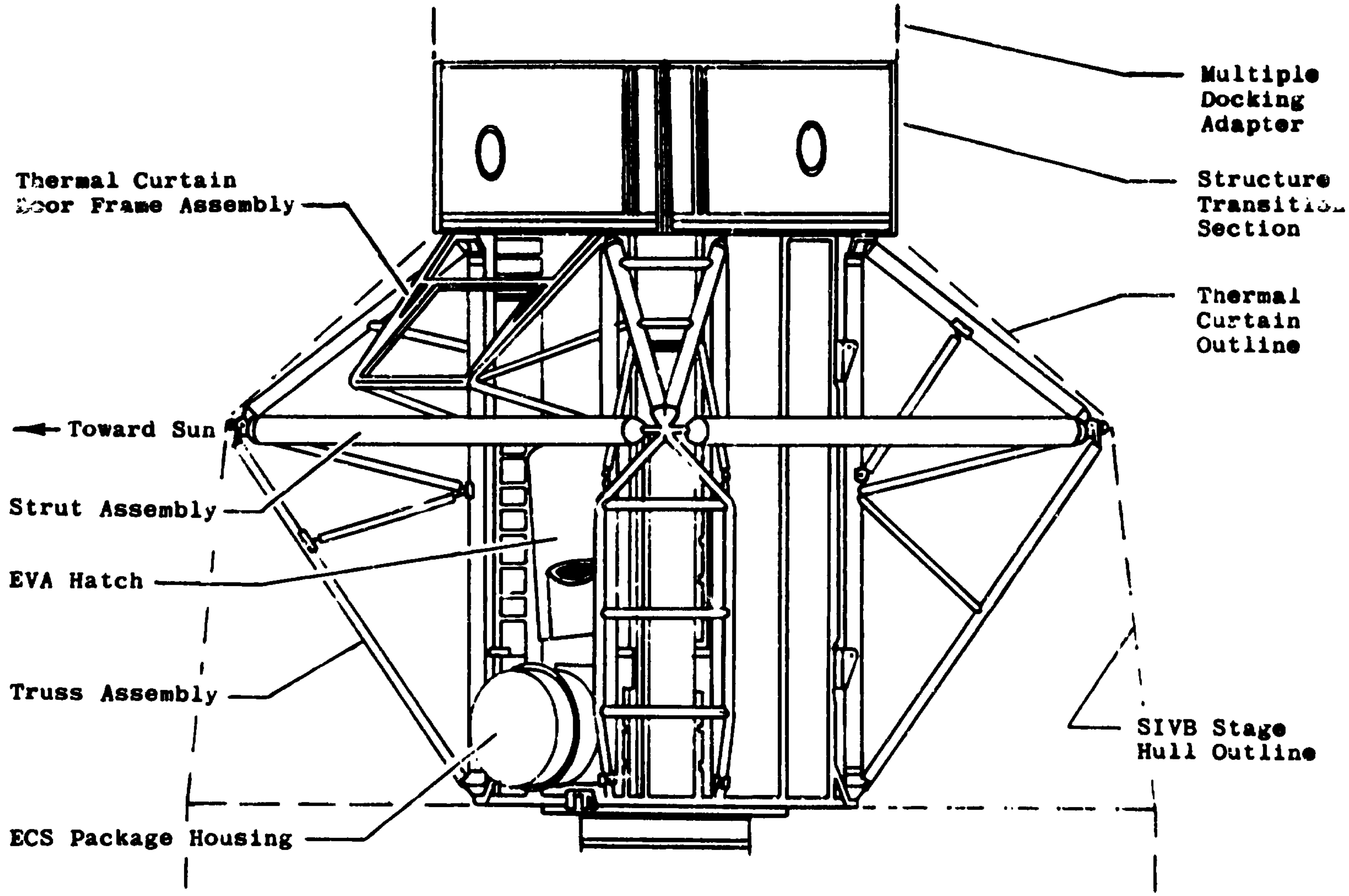

Figure 5
Airlock Module Detail

structure and the lids of both D022 canisters will be outside the thermal curtain. (The thermal curtain, the Multiple Docking Adapter, and the experiments are protected by shrouds and a nose cone during launch. These are jettisoned after orbital insertion.) Mounting alignment and position will permit observation of all experiment sequences from viewports of the Docking Adapter. Since the Workshop always keeps the same side toward the sun, proper mounting will also assure that about 50% of the surface of the D021 airlock and D022 deployed panels will receive solar radiation. This will provide realistic space exposure and will also make photographic coverage possible using only available sunlight. The two experiments will be discussed separately and in greater detail below.

D021 EXPANDABLE AIRLOCK

With the rapid advancement in space technology, the United States will be orbiting a series of manned laboratories from which astronauts can perform various experiments and tasks. Some of these experiments and tasks will necessitate astronaut EVA. An airlock system will be required to alleviate the repeated loss of oxygen due to the decompression and compression cycles imposed on the laboratory work area during egress and ingress maneuvers associated with EVA. An expandable airlock would minimize weight and volume requirements imposed on the vehicle and would permit maximum utilization of the internal volumes already available in these laboratories and spacecraft. Experiment D021 is the first preliminary attempt to evaluate the feasibility of such a structure under actual orbital conditions.

One of the prime objectives of the D021 experiment, in addition to the evaluation of materials, is to demonstrate the compatibility of a non-rigid structure with the dynamics of astronaut ingress/egress maneuvers and to establish minimum hatch diameter and airlock volume required in an orbital airlock. To fulfill these objectives it is necessary that the experiment be performed outside the Orbital Workshop, necessitating EVA. The only alternative would be to perform the astronaut ingress/egress phase inside the OWS and the materials evaluation phase outside the OWS. This would require that the airlock be repackaged, transported outside through several hatches, and remounted on the OWS exterior. Due to the narrow restrictions of the hatches and the fact that the D021 airlock could not be efficiently repackaged, this procedure was eliminated.

ESTABLISHMENT OF CONFIGURATION

An extensive in-house effort was initiated prior to contract award to establish the basic configuration of the D021 airlock structure. This in-house effort included underwater neutral buoyancy tests and zero-G aircraft tests, performed by the Air Force Aero Propulsion Laboratory and the 6570th Aerospace MedicalResearch Laboratory, Wright Patterson AFB. These tests established the shape and minimum hatch and airlock sizes required for an astronaut with a back pack to enter, close the hatch, turn around, and egress. Also established were proper hatch hinging operation and the location and type of mobility aids, hand-holds, and tethers. Since the experiment requires man's participation, NASA and USAF required thorough ground based testing of all EVA procedures to assure astronaut safety. All EVA human factors requirements were established during this test program; in addition, the preliminary timeline analysis for the EVA portion of the experiment was determined.

Neutral Buoyancy Testing

The preliminary configuration of the airlock structure was established in underwater neutral buoyancy tests conducted during the latter part of 1966. A fiberglass and wood mock-up was fabricated by WPAFB shops to evaluate minimum hatch size, interior dimensions, double-hinged hatch configuration, and latching mechanism. The guidelines for this first attempt at establishing a configuration were (1) minimum sweep clearance for hatch opening, (2) simple latching mechanism requiring only one-hand operation (3) hatch diameter sufficient to allow one pressure-suited astronaut with a back or chest-mounted portable life support system to adequately enter and leave the airlock, and (4) interior volume sufficient to allow a pressure-suited astronaut with a back-pack to perform turn-around maneuvers inside the airlock. This maneuver would be required to operate both hatches of an operational airlock.

Extensive underwater tests, similar to that shown in Figure 6, were performed with both one and two subjects, to evaluate each of the above EVA-imposed guidelines. Each subject made observations and comments regarding the validity of the proposed design, keeping in mind the EVA requirements of the experiment. From these tests it was determined that the double-hinged hatch design and latching mechanism were adequate and simple enough to give reliable operation in an extra-vehicular environment. The preliminary airlock design configuration was found to be on the conservative side by one to two inches. For instance, the original hatch was 36 inches in diameter; but testing indicated that a 34-inch diameter opening was sufficient.

The pressure-suited test subject determined the required minimum turn-around volume by first bending at the midriff and then rotating in the airlock. This was done with the suit pressurized to approximately 3 psig. The subject was able to use the wall of the airlock for leverage to allow him to reduce his crouched height and also as a means of moving himself inside the structure. These tests were performed with and without chest or back-packs. From these tests it was found that a subject would successfully rotate in a 60-inch diameter structure. In the actual airlock the non-rigid sides will flex somewhat, allowing the astronaut to perform this turn-around maneuver without difficulty. After considerable practice it was possible to perform the complete ingress/turn-around/egress maneuver in about 45 seconds.

In evaluating the hatch latching system, it was found that only one hand was needed for operation, although two hands are needed for maximum efficiency. The latching system consists of two handles located 180 degrees to each other and about 12 inches long. They move opposite to one another, negating torque imparted to the astronaut during the opening sequence. If only one hand is used

Figure 6
D021 First Neutral
Buoyancy Mockup

Figure 7
D021 Final Neutral
Buoyancy Mockup

to open the airlock, then the body should be stabilized with the other hand by grasping the toroidal metal tether ring. There is no stability problem inside the airlock as the friction between the astronaut's feet and the side walls of the airlock is sufficient to retain proper orientation. In this experiment the hatch will not be locked with the astronaut inside the structure.

Several emergency situations duplicating those which might arise inside a closed and latched D021 airlock were set up. These required a second subject to open the D021 hatch, enter partway, and remove an immobilized subject. Preliminary tests indicated the conservative design was more than adequate for this maneuver.

Based upon these preliminary results, a second D021 fiberglass mockup with a metal hatch was fabricated and subjected to the same type of tests as the original. This particular mockup was an updated, realistic structure, the dimensions and configuration of which would match the final airlock (See Figure 7). It incorporated a 34 inch hatch and had overall dimensions of 56 inches by 60 inches. Additional tests similar to those performed on the initial mockup verified the adequacy of the final design. These tests were performed in early 1967 at Wright Patterson AFB. Additional tests on this structure included evaluation of mobility aids, EVA tether ring, and knee tether attachment. As a result of this series of tests, the type and location of mobility and hand holds were defined.

Three fabric straps, each 2 inches in width, were sewn equally spaced inside and outside of the airlock expandable structure and extending around the airlock. The fabric straps mounted on the airlock will permit easy folding of the structure during packaging, but still provide adequate strength and rigidity for mobility aids. The three straps mounted outside the airlock will allow the astronaut to proceed easily from the base of the structure to the hatch. The straps inside the airlock provide additional aids to the astronaut, when performing his turnaround maneuver and also as an aid in reaching the rear of the structure. In addition, a toroidal metal tether ring and hand-hold was mounted circumferentially around the D021 hatch opening. During the second series of tests the basic experiment timeline was determined and the immobilized astronaut rescue procedure was established.

Tests to establish the immobilized astronaut rescue procedure were originally to determine whether it was possible to open the hatch with an immobilized, pressure-suited subject inside. Since a pressure-suited subject who has lost muscular control assumes a spread-eagle configuration because of suit pressure, it was believed such a situation could jam the hatch closed, making rescue very difficult if not impossible. In initial attempts the subject acting as rescuer, also in a pressure suit, would open the hatch and, because of the double hinge design, use the hatch door as a scoop and gently push the immobilized subject's feet or head aside. Sufficient clearance could be gained so that the rescuer could manually reach in and manipulate the immobilized subject sufficiently to fully open the hatch.

Several trials were run,

alternating the "disabled" subjects and varying the original position of the "disabled" crewman, i.e., feet against hatch, heat against hatch, etc. Various restraining methods were used to secure the rescuer to the airlock, ranging from a knee tether to no restraint whatsoever. Given the worst case, the larger subject "disabled" with feet behind the hatch, it was shown that maximum retrieval time was less than 140 seconds, well within the capacity of the emergency oxygen supply.

Zero-G Testing

These two early mock-ups were, of course, rigid structures. An expandable mock-up, fabricatd from the final D021 wall structure and incorporating all the latest design changes established in previous tests, was subjected to extensive zero-G tests aboard the KC-135 aircraft. This mock-up included the three circumferentially-mounted external and internal fabric handholds and the hatch tether ring. (See Figure 8) All mobility aids were evaluated in this zero-G environment to verify data gained previously in the underwater neutral buoyancy tests. The knee tether attachment evaluated previously was proven ineffective in zero-G tests and discarded. All other aspects of the D021 design were found to be acceptable, and at the conclusion of these tests the airlock design was frozen. The zero-G tests were performed in late 1967. In addition to verifying the design further tests were performed to establish the timeline and to perfect the immobilized astronaut rescue procedure. Figure 9 illustrates the final locations of the handholds and tether ring as mounted on the D021 qualification testing hardware.

Of particular interest to the Air Force during this test program was the possible application of unwanted perturbations to either the D021 structure or the test subject. Such perturbations could be imparted by a test subject making severe maneuvers within the nonrigid expandable airlock in a zero-G environment. Subsequent tests proved that this was very unlikely and should not be a problem during EVA.

DESCRIPTION OF D021 HARDWARE

The drawings of Figure 10 show the D021 airlock in packaged and expanded configurations, and illustrate its major subassemblies. They are: (1) the packaging system, (2) airlock structure assembly, (3) hatch assembly, (4) pressure bulkhead assembly, (5) pressurization system, (6) telemetry data system, (7) electrical system, (8) mounting structure assembly, and (9) experiment control system.

Packaging System

The packaging system consists of a series of flexible nylon straps, located around the periphery of the mounting base, to restrain the expandable portion of the airlock structure in a packaged configuration. The restraining straps terminate at a release fitting located at the apex point of the package. The D021 airlock can be deployed both manually and by remote control.

Airlock Structure Assembly

The airlock expandable structure assembly consists of the composite wall material which is bonded and joined to aluminum terminal rings at each end of the airlock. The function of these rings is to

Figure 8
D021 Zero-g
Testing Mockup

Figure 9
D021 EVA Aids
(Final Design)

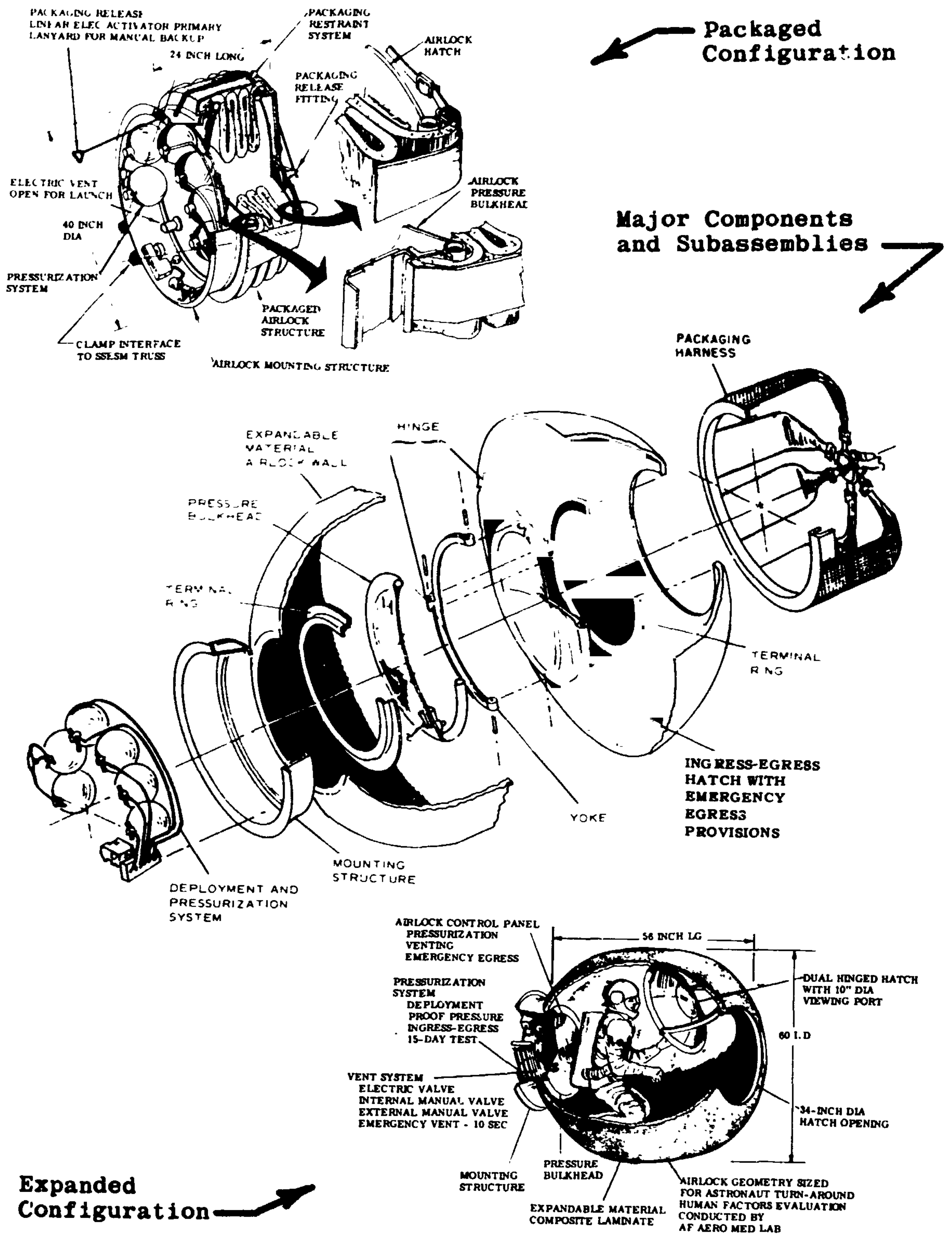

Figure 10
D021 Hardware

provide a rigid termination for the flexible material of the airlock and also to provide a smooth flat surface for hatch and pressure bulkhead seals. The airlock wall can be compressed by a vacuum technique, from the fully expanded thickness of about 1 inch to 1/4 inch. This expandable portion of the structure used the elastic recovery materials technique, to permit folding and packaging of the compressed structure into a small, compact configuration for launch. Once in orbit, the airlock is deployed to its full expanded configuration by the recovery action of the wall material, augmented by low-level pressurization for final shaping. After final shaping, the inherent stiffness of the wall structure will ensure the final shape is maintained under orbital conditions.

Basically, the nonrigid structure wall is a four-layer composite as shown in Figure 11 and described below. Final shaping of the airlock is provided by pressuring a bladder. This pressure bladder is a laminate of three individual sealant layers, with an inner layer of 0.3-mil aluminum foil. The inner sealant layer is a laminate of nylon film-cloth. This layer is bonded with polyester adhesive to a second layer of closed-cell EPT foam. The outer sealant is a nylon film-cloth laminate coated with a polyester resin. A filament winding manufacturing process is used for the structural layer and provides nearly the optimum in a lightweight, load-carrying, flexible structure. The structure layer will be would with three 0.0036 inch stainless steel wires interlaced with a rayon yarn. Micrometeoroid protection is achieved by a one-inch layer of flexible polyester foam. While the primary function of the foam is to act as a micrometeoroid barrier, it also serves as a deployment aid. The outermost layer of the composite wall structure encapsulates the wall, to provide a smooth base for the application of a thermal coating. This passive thermal control coating maintains material temperatures within acceptable limits. Inasmuch as the outer cover encapsulates the composite wall, it also serves as an aid in packaging the structure prior to launch.

Hatch Assembly

The hatch assembly consists of a basic dome and compression ring structure, a dual yoke-type hinge and latch hardware, hatch separation provisions for emergency egress, and a 10 inch diameter viewing port. The hatch latches and seals against the terminal ring and can be operated either from inside or outside the airlock. The pressure dome and compression ring structure is fabricated of aluminum and is separable from the overall hatch assembly, as a provision for emergency egress.

Pressure Bulkhead Assembly

The pressure bulkhead assembly consists of a basic dome and compression ring structure of aluminum, similar in concept to the dome structure of the hatch. The bulkhead assembly seal and connection is made at the 34 inch diameter terminal ring of the airlock structure and is also attached to this ring with six equally-spaced bolts. Provisions are incorporated into this assembly for a control panel; a future connection for an astronaut

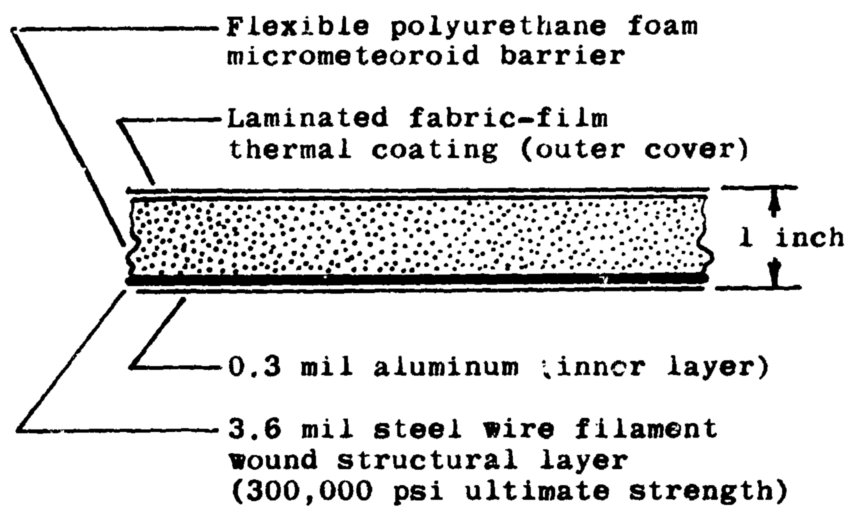

Figure 11
D021 Structural Materials Composite

umbilical; and subsystem connection requirements for pressurization, venting, electrical wiring, and instrumentation.

Pressurization System

The pressurization system for the D021 experiment consists of six 150 cubic inch high-pressure gas storage bottles charged with nitrogen. The gas is released from each storage bottle through a pyrotechnic valve to flood the airlock to a specific level of pressure, established by charging the bottle to a predetermined level. All elements of the pressurization system are supported off the mounting base structure assembly. Controls for gas release are located on the remote control panel in the Workshop Airlock Module.

Telemetry Sensors

Pressure and temperature data will be monitored during the experiment by the NASA Airlock Module telemetry system. Eight sensors will be provided, two for airlock pressure and six for interior and exterior surface temperatures of the expandable material wall structure.

Electrical System

Power for the D021 experiment will be supplied by a self-contained battery pack and the 28V power source of the Workshop Airlock Module. The self-contained power source will be a dual pack of nickel-cadmium batteries. These batteries will supply power only for the pyrotechnics included in the design. The Airlock Module will provide all remaining power requirements for the experiment.

Mounting Structure

The mounting base structure is constructed of light-gage aluminum sheet and provides the physical integrating function for all hardware components of the experiment. The airlock structure assembly is attached to one ring face of this structure. All subsystems exterior to the airlock itself are located within and supported on the mounting shell structure.

Experiment Control System

Controls for conducting the airlock experiment were to have been provided in three locations: (1) A remote control panel located inside the Airlock Module to provide the principal source of control for conducting the experiment; (2) a control panel mounted inside the D021 Airlock on the pressure bulkhead; and (3) Emergency and/or back-up controls mounted on the exterior of the D021 airlock mounting base structure. Because of subsequent simplification of the experiment, the second set of controls was eliminated and the third was reduced to an emergency pressure relief valve. A possible design for the Airlock Module control panel is shown in Figure 12.

D021 EXPERIMENT PROCEDURES

Extravehicular activity is required only during Phase II, the airlock ingress/egress cycles. This portion of the procedure was verified in underwater and zero-g aircraft tests. It is anticipated

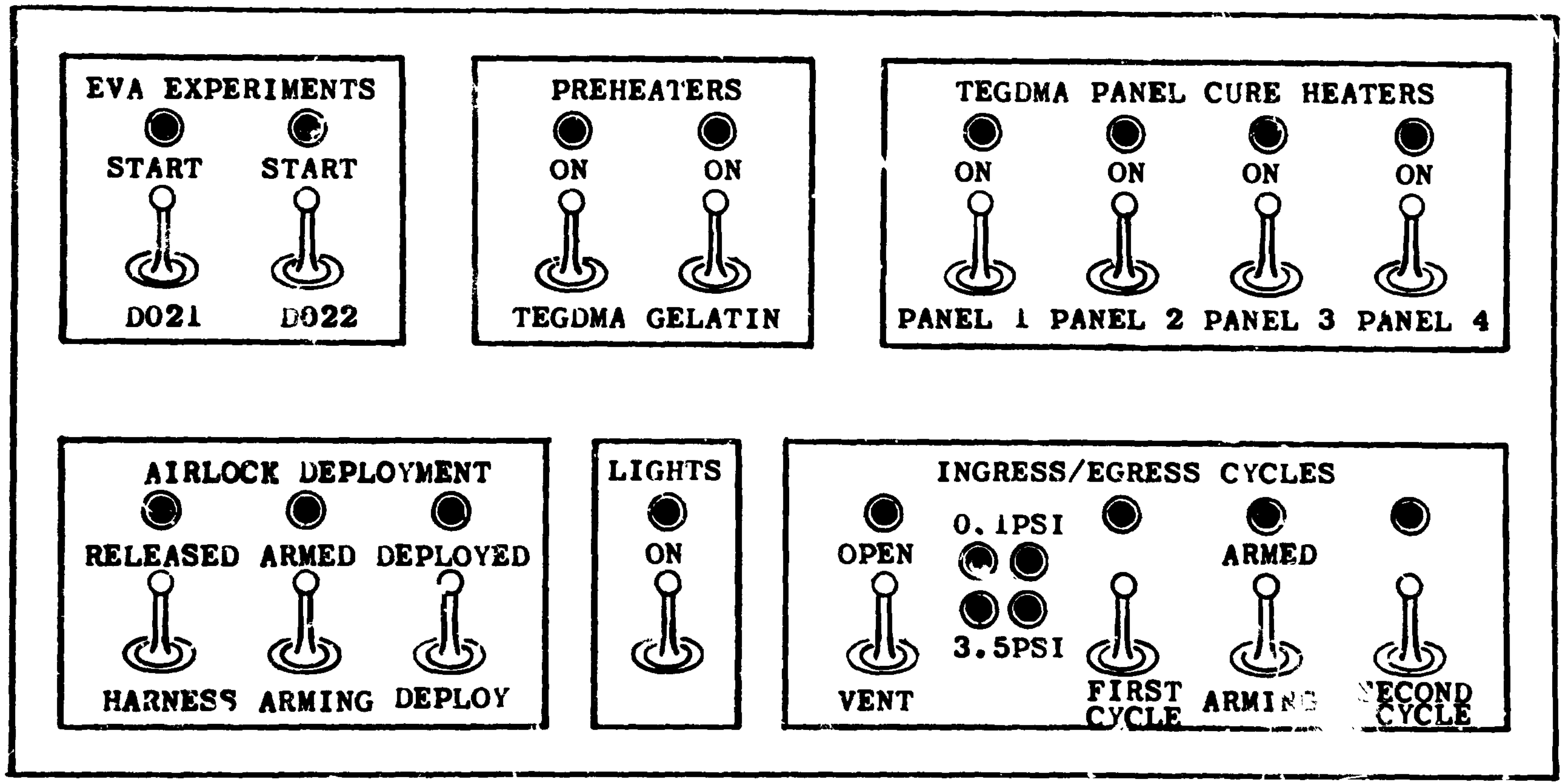

Figure 12
Conceptual Airlock Module
Control Panel for D021 and D022

that airlock deployment and initiation of the 15-day leakage and pressurization test will require about 2 minutes; the EVA ingress/egress cycles will take less than one hour.

Phase I

Deployment. There is no extravehicular activity during the first phase of the experiment. Crewman 1 remains at the D021 control panel in the Airlock Module. Crewman 2 is stationed at an observation port of the Multiple Docking Adapter, from which he may watch the progress of the experiment.

Step	Cumu Time, Sec.	Task Sequence For Crewman 1	Task Sequence For Crewman 2
1	000	Checkout control panel. Actuate experiment start switch and observe indicator light.	
2	030	Close vent switch and observe indicator light.	
3	045	Arm D021 airlock deployment switches.	
4	060	Actuate airlock restraint harness release switch and observe indicator light.	Observe and report harness release and partial deployment
5	090	Actuate final shaping switch and observe indicator light	Observe and report airlock shape and deployment progress.
6	120	Return to other duties.	Return to other duties.

The experiment start switch is left on, to allow for continued readout of the sensors monitoring airlock pressure and temperature. Step 5, the final shaping of the airlock, also initiates the 15-day leakage test.

Phase 2

Ingress/egress cycles. During Phase 2, Crewman 1 leaves the Workshop, via the Airlock Module EVA hatch; and Crewman 2 controls the experiment from the control panel in the Airlock Module.

Step	Cumu Time, Min.	Task Sequence For Crewman 1	Task Sequence For Crewman 2
1	000	Exit Airlock Module through EVA hatch.	Assist Crewman 1
2	001	Proceed to D021 airlock.	Turn on D021 airlock interior lighting switch and observe indicator light.
3	014	Perform interior examination of airlock through window.	
4	015	Move to safe distance from airlock and rest.	
5	016		Open vent valve to relieve pressure, and observe indicator light. Observe dual low-pressure (<0.1 psi) lights for proper signal.
6	017		Close vent valve and observe indicator light. Arm pressurization switches. Actuate pressurization switch for first ingress/egress cycle and observe indicator light. Observe dual high-pressure (>3.5 psi) lights for proper signal.

Step	Cumu Time, Min.	Task Sequence For Crewman 1	Task Sequence For Crewman 2
			Disarm pressurization switches.
7	019		Repeat Step 5 and report when low-pressure lights come on.
8	020	Return to D021 airlock.	
9	021	Observe exterior and perform interior examination through window and report.	
10	023	Release hatch and enter.	
11	024	Inspect interior and exit.	
12	027	Secure hatch and report.	
13	028	Repeat Step 4.	
14	029		Repeat Steps 5 thru 7.
15	033	Repeat Steps 8 thru 12.	
16	041	Remove two materials samples from base of airlock.	Turn off D021 airlock interior lighting switch and observe indicator light. Turn off experiment start switch and observe indicator light.
17	043	Return to Airlock Module and enter through EVA hatch.	Assist Crewman 1
18	056	Stow materials samples in protective container.	Assist Crewman 1

The final EVA task required in the experiment is the retrieval of two materials sample. One 8 x 8 x 1 - inch panel of the D021 expandable wall structure and another sample of material are to be returned in the Apollo Command Module at the end of the 28 day Saturn I Workshop Mission. A second set will be returned after the follow-on revisitation mission several months later. The samples will be returned in a sealed metallic container. It is presently envisioned that all experiments requiring return of samples may be placed in a single, compartmented, sample return canister. This would include Air Force experiments D021 and D022, and two NASA experiments. This specific task is undefined at the present time.

D022 CHEMICAL RIGIDIZATION EXPERIMENT

Unlike the Expandable Airlock, which by its very nature must be evaluated by man, Experiment D022 could have been designed for automatic, rather than manual performance. The basic requirement here was that the experiment take place outside the Orbital Workshop, to obtain true exposure to the space environment.

DESIGN TRADE-OFFS

The experiment has two primary phases, both of which occur outside the Workshop. The first phase is the deployment in space of the eight fiberglass panels and the initiation of the rigidization process. The second is the recovery of four of the rigidized panels, after prolonged exposure to the orbital environment. Each of these phases could be performed either automatically or by an extra-vehicular astronaut. ("Automatically" here is defined as an action or sequence of actions which does not require a member of the Workshop crew to leave his shirt-sleeve environment.) This generates four possible approaches to accomplishing the objectives of the experiment:

1. Automatic deployment, followed by automatic recovery.

2. Manned deployment, followed by automatic recovery.

3. Automatic deployment, followed by manned recovery.

4. Manned deployment, followed by manned recovery.

A fifth possibility, an automatic experiment aboard an orbiting reentry satellite, was not considered due to extreme cost and complexity.

Deployment Phase

Experiment D022 could have been deployed either automatically, or as finally was chosen, by an astronaut. Hardware in these two cases would have differed in four instances:

1. A pyro-valve, rather than a manual one, would be needed to vent each canister to space.

2. The hand-operated clamp which maintains the seal at the canister lid would be replaced with some type of pyro-disconnect.

3. The telescoping rods to which the canister lid and the four panels are mounted would be extended by a pyro-activated pneumatic system, rather than by having an astronaut pull them out.

4. A pyro-valve, rather than the manual valve now used, would allow the inflation system to deploy the panels.

Hardware weight would be approximately the same in either case. Although the use of pyrotechnics makes the design of experiment hardware somewhat simpler, it also represents a potential fire or explosion hazard. This is particularly true for one of the two canisters, which is pressurized with oxygen and contains panels impregnated with a vinyl acrylic resin.

The greatest drawback to automatic deployment, however, is the need it generates for some means to visually observe and record the extension of the telescoping rods and the inflation of the panels. A closed-circuit television system would be superior here to a motion picture camera in that it permits instantaneous observation of deployment. This is desirable to assure that each step of the experiment has been completed, before succeeding steps are initiated. Use of an automatic movie camera would also require an EVA to retrieve the camera and film. Also, a stationary camera of either type would probably be unsuited for this purpose, in that no more than one side of each of two panels per canister would be entirely visible to the fixed camera. For proper observation of the inflating and rigidizing panels, a movable camera or several fixed cameras would be required.

Recovery Phase

The two approaches using automatic recovery would require an auxiliary airlock to bring the rigidized panels into some pressurized area. Such an airlock could be installed in the Workshop hull, the Multiple Docking Adapter, or the EVA Airlock Module. Of these three, the last never receives exposure to solar and background radiation - the aft end of the module is shielded by a metal shroud, formed by an extension of the OWS hull, while the forward end is covered by the thermal curtain.

Incorporation of an auxiliary airlock into the hull of the Workshop itself is possible, but not practical. The inner portion of the airlock would be subjected to the cryogenic hydrogen fuel during time spent on the pad and during launch and ascent. The intense cold would require the use of both considerable insulation and of full-time electric heaters in the experiment canisters. Apart from these design drawbacks, such an arrangement would represent a potential hazard: The vinyl panels must be stored in at least a 9 psi oxygen partial pressure atmosphere. to prevent premature curing of the resin. The presence of a powerful fuel (hydrogen), an oxidizer (oxygen in the canister), and an ignition source (the electric circuitry), separated by a potentially leaky barrier (the inner door seals of the auxiliary airlock), might become dangerous.

The small airlock required for panel retrieval could have been designed into the Multiple Docking Adapter, which is pressurized during most of the mission. As presently envisioned, the inner wall of the MDA is to be lined with flat sheets of lightweight metal grid. Experiment packages and those items to be taken into the OWS after passivation will be mounted on these grids with releasable fasteners. Such an arrangement is not only well suited to the abilities of the crewman, but allows great flexibility of design. It permits rapid, relatively straightforward modification of the interior layout to meet changing mission profiles. The greatest drawback to incorporating the retrieval airlock into the Docking Adapter would be, then, the constraint which it would impose on the overall design of the MDA interior.

Summary

Manned deployment of the experiment suffers in comparison

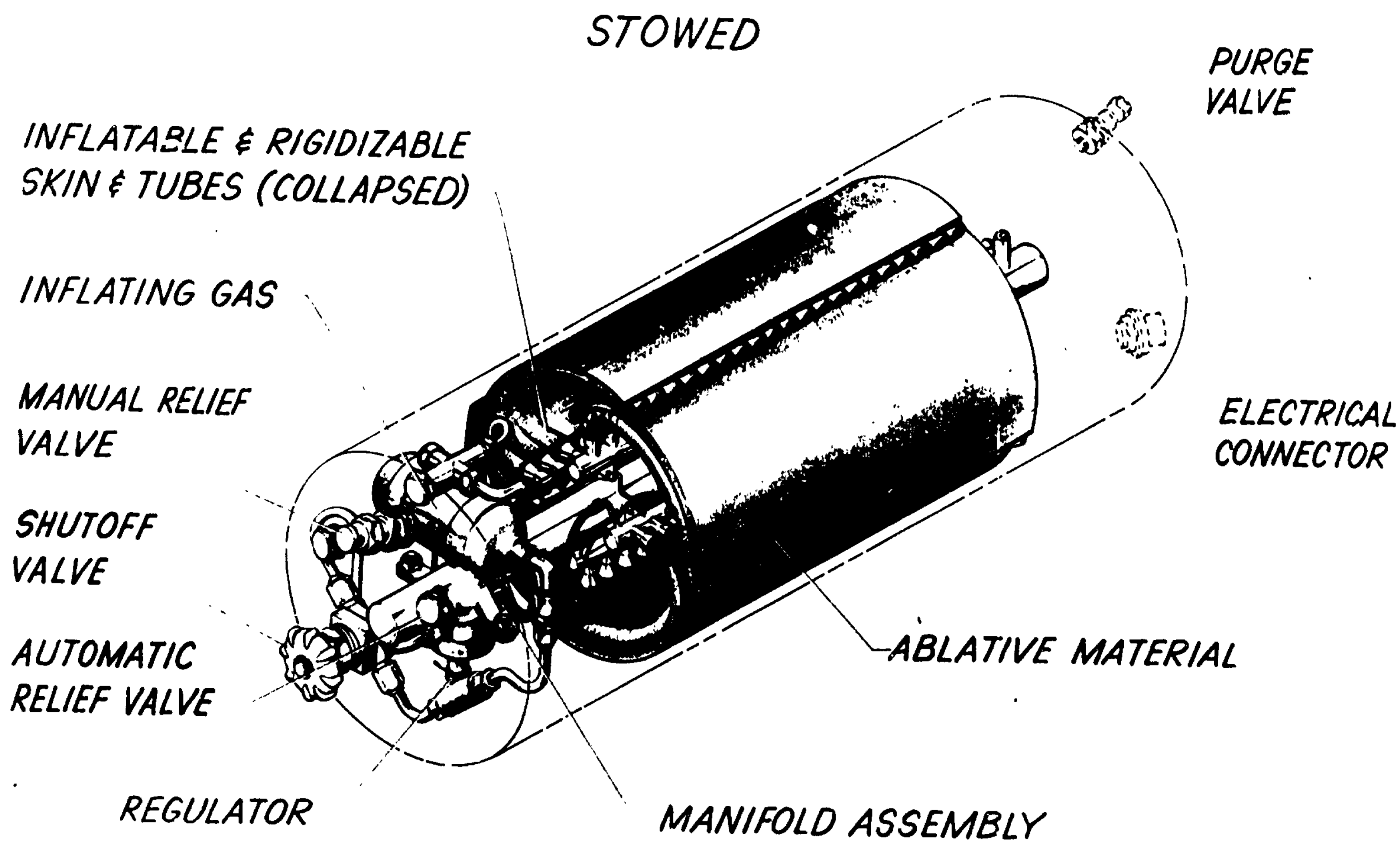

Figure 13
D022 Stowed Configuration

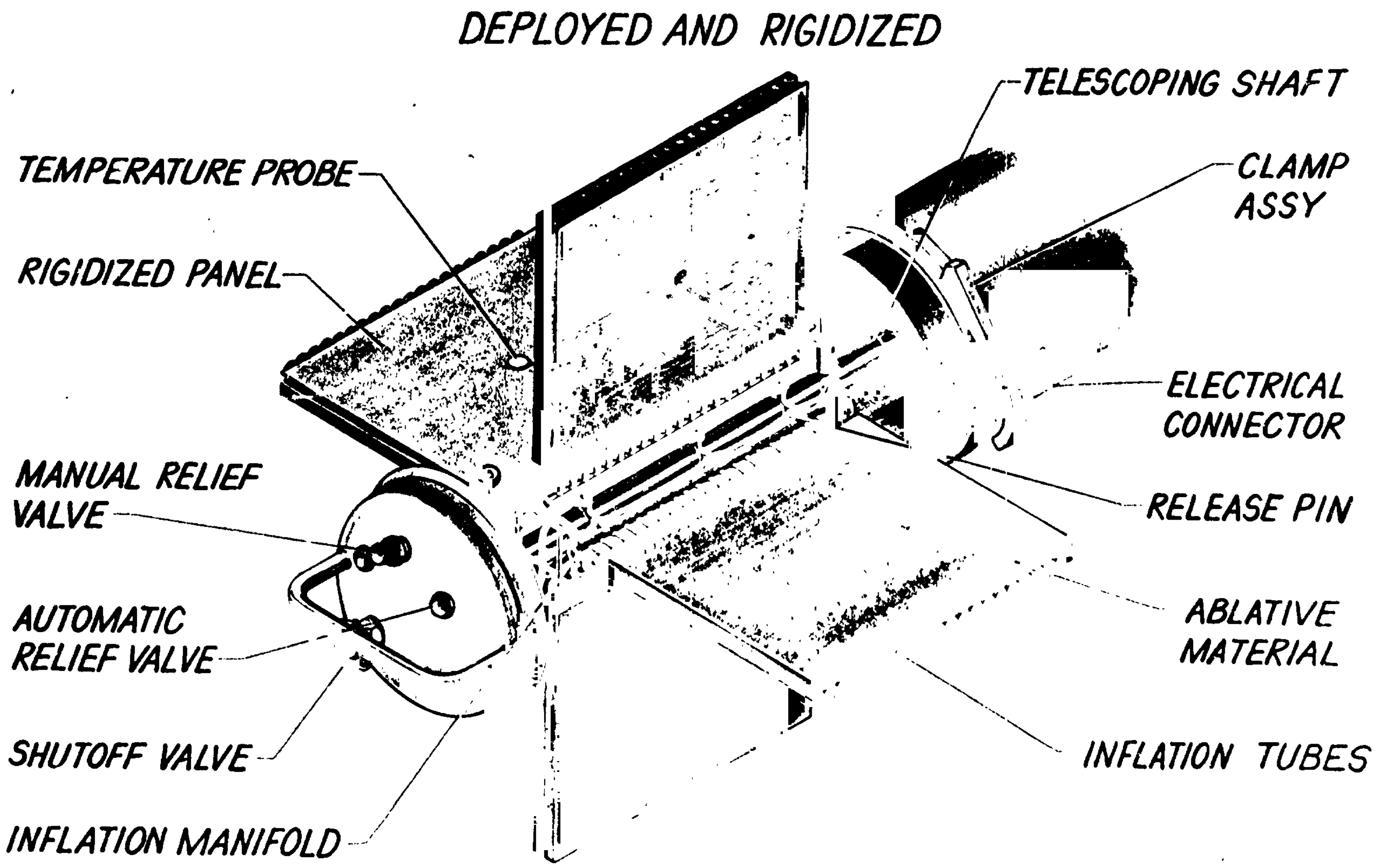

Figure 14
DU22 Deployed Configuration

with automatic deployment, solely in that requires a period of extra-vehicular activity. To offset this liability are the unmatched capabilities of man the observer - his mobility and his ability to draw conclusions. Manned recovery of the rigidized panels is superior to automatic recovery in many respects - weight, volume, complexity, etc. Of even greater importance - and one not limited to Experiment D022 - is the flexibility which EVA gives to the mission and hardware designer.

DESCRIPTION OF D022 HARDWARE

Five basic pieces of hardware are involved in the experiment: (1) eight panels of the sample materials, for eventual return to earth; (2) canisters to store and protect the panels; (3) a junction box which provides the electrical interface between the canisters and the Workshop; (4) a control panel in the Airlock Module; and (5) a container to protect the panels during their return to earth. Total experiment weight is 50 pounds. These hardware items are described in greater detail below. D022 hardware in both packaged and deployed configurations are shown in Figures 13 and 14 respectively.

Panels

All the panels are similar in construction, size, and shape. Each consists of a fluted-core fiberglass structure measuring 9 x 11 1/2 x 1/2 inches, coated on one side with a 1/4 inch thick flexible silicone ablator and weighs about 2 1/4 pounds. Rubber tubes are placed in the flutes of the panel for inflation. Four panels are impregnated with a gelatin resin, which rigidizes by plasticizer (water) boil off. The other four panels are impregnated with a tetraethylene glycol dimethyacrylate resin and are rigidized by resistance heating of wires imbedded in the fiberglass. A thermocouple is mounted on each panel.

Panel Canisters

Two panel canisters are used, one for the gelatin panels and one for the vinyl panels. Each canister, 28 inches long by 12 inches in diameter, contains a panel pressurization system and an electric heater and thermostat for heating the panels prior to deployment. The final venting of the canister, deployment of the experiment, and pressurization of the panels are done manually by an astronaut, using controls on the canister.

Electrical Junction Box

A small electric junction box is mounted between the two canisters, under the thermal curtain, to interface the electrical and instrumentation subsystem of the experiment with those of the Airlock Module.

Control Panel

An experiment control panel will be mounted in the Airlock Module. It will contain controls for operation of the experiment heaters, which will be turned on 12 hours before panel deployment, to bring the panels to the proper temperature; vinyl panel curing heaters; and pressure and temperature readouts. Figure 12 shows a possible design for such a panel.

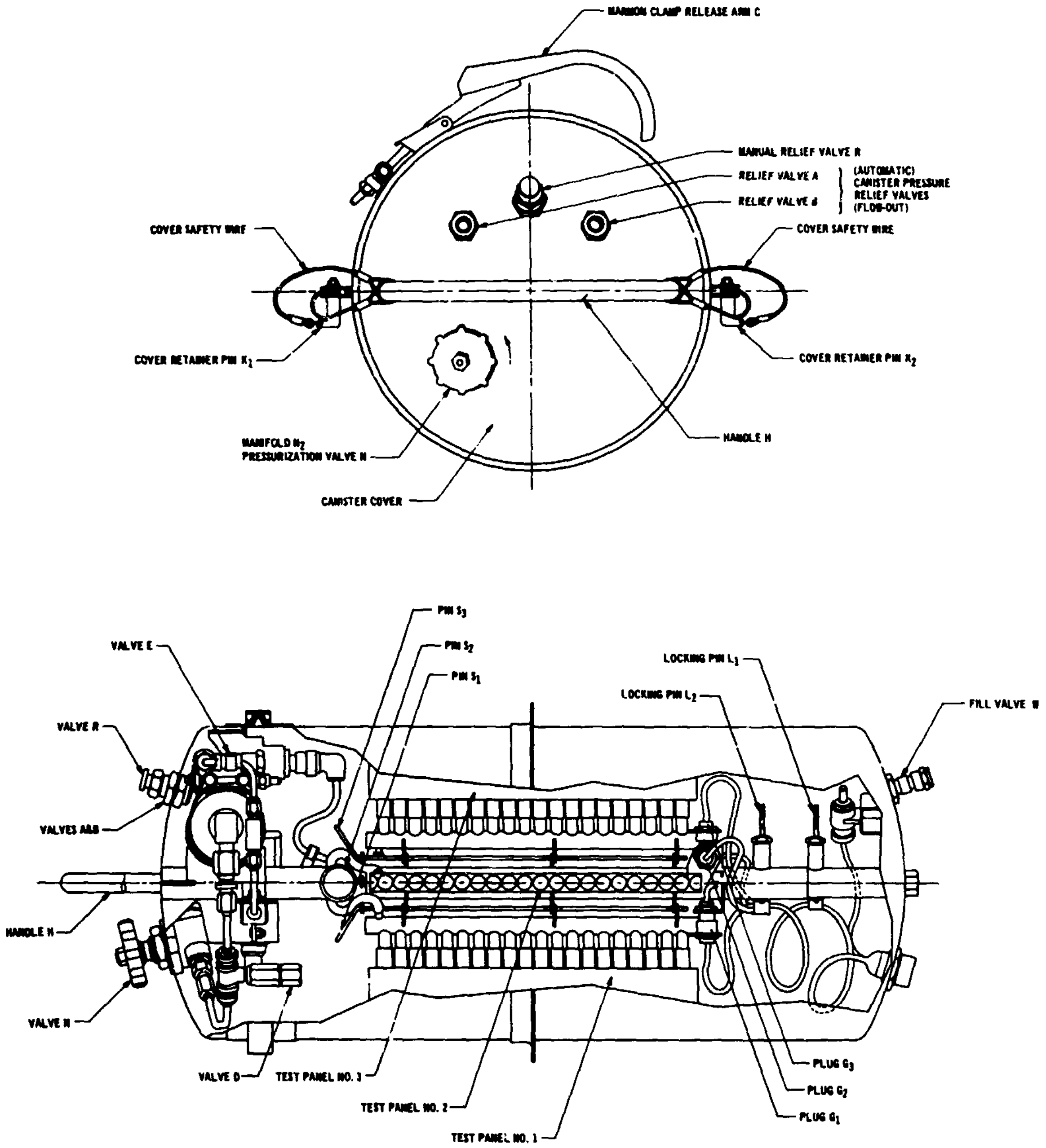

Figure 15
D022 Operating Controls

Panel Return Container

A container will be provided for returning the four rigidized panels (two impregnated with gelatin and two with vinyl resin) to earth for analysis. The container will be stored in the cabin of the Apollo Command Module during return from orbit. As discussed earlier, the container may also be used for samples from other experiments. The four remaining rigidized panels will be returned by the crew of the revisitation mission.

HUMAN FACTORS

Human factors experience gained in the Gemini program has been used as a guide in developing the D022 space experiment. All controls, knobs, valves, clamps, and panel release mechanisms are designed for operation by a pressure-suited, extra-vehicular astronaut. The operation of the experiment is kept simple, requires no great dexterity on the part of the astronaut, and involves no lengthly tasks.

A failure Mode and Effects Analysis was performed on the experiment. This analysis takes all failures and places them in one of three categories, depending on whether the failure affects (1) crew safety, (2) mission success, or (3) experiment success. These are in order of decreasing criticality. One potential hazardous failure mode was discovered: If the experiment canister is pressurized when the canister clamp is released, the canister cover would deploy rapidly, possibly striking the astronaut. Two safety tethers were added to prevent the canister cover from deploying more than a few inches. These tethers are released by the astronaut after the canister clamp is opened.

All restraint devices at the work site and translation devices from the Airlock Module to the experiment worksite are being developed by NASA.

D022 EXPERIMENT PROCEDURES

The experiment will require several periods of EVA, each of which will involve two astronauts. One crewman (identified herein as Crewman 1) will perform the actual experiment in space. The other (Crewman 2) will remain in the Airlock Module, to act as an observer and to operate the remote controls. Both sequences begin with the two astronauts in the unpressurized Airlock Module, and end with the crewmen back in the Airlock, ready to initiate pressurization. All controls mentioned are shown in Figure 15. Total times for Phases 1 and 2 are about 40 and 30 minutes, respectively.

Phase 1

Panel Deployment. A preliminary time-line analysis is given in the following table. Note that during Step 8 Crewman 2 will actuate either one switch, to begin simultaneous cure of all four TEGDMA resin panels, or will actuate one switch for each panel every 7.5 minutes, for consecutive panel curing. Which of these two tasks is required will depend upon the power available from the workshop.

Step	Cumu Time, Min.	Task Sequence For Crewman 1	Task Sequency For Crewman 2
1	000	Open AM hatch and egress.	Assist Crewman 1
2	003	Translate to location of experiment canisters on AM.	
3	007	Install camera on mounting bracket. Prepare worksite for experiment.	
4	011	Position self in front of TEGDMA resin panel canister and rest.	Switch off preheater power for both canisters from AM control panel.
5	013	Depress manual relief valve R, to relieve canister internal pressure.	Observe and take pictures of complete sequence.
6	014	Release canister Marmon Clamp C, move it to rear, close clamp. Release two cover retainers pins K_1 and K_2, and gently pull out canister cover and attached panel assembly, until pins lock into position.	
7	016	Inflate panels by turning manifold pressurization valve N counterclockwise.	
8	018	Rest. Report observation of panel curing.	Actuate panel cure switch (es) on AM control panel.
9	022	Position self in front of gelatin panel canister.	
10	023	Repeat Steps 5 thru 7.	
11	028	Rest and report observations.	
12	032	Remove camera from mounting bracket.	
13	036	Translate to AM hatch	
14	039	Ingress through AM hatch, close hatch, repressurize AM.	Assist Crewman 1.

Phase 2

Panel Recovery. The second phase of the experiment is the retrieval of four rigidized panels, two impregnated with the TEGDMA resin, and two with gelatin. This must be done no sooner than 24 hours after completion of Phase I and for best results should not be done for a week. A time-line similar to that below would, presumably, be followed during recovery of the remaining four panels on a later mission.

Step	Cumu Time, Min.	Task Sequence For Crewman 1	Task Sequence For Crewman 2
1	000	Open AM hatch and egress.	Assist Crewman 1.
2	003	Translate to location of experiment canisters on AM.	
3	006	Prepare worksite for experiment.	
4	010	Position self in front of TEDGMA Resin panel canister and rest.	
5	012	Turn off manifold pressurization valve N.	
6	013	Attach two safety lines on utility harness to Panels 1 and 2.	
7	015	Pull Panel 1 release pin S_1 in axial direction. Pull panel away from central manifold tube disengaging electrical connector plug G_1 and manifold hose H_1. Repeat for removal of Panel 2.	
8	017	Place Panels 1 and 2 in separate pockets of plastic bag.	
9	018	Position self in front of gelatin panel canister.	
10	020	Repeat Step 5 thru 8	
11	026	Return worksite to original condition.	

Step	Cumu Time, Min.	Task Sequence For Crewman 1	Task Sequence For Crewman 2
12	029	Translate to AM hatch	
13	030	Ingress through AM hatch, close hatch.	Assist Crewman 1
14		Place four panels in protective container, seal container.	Assist Crewman 1
15		Repressurize AM	Assist Crewman 1

CONCLUSIONS

1. The D021 and D022 experiments as described in this paper will fulfill the experiment objectives.

2. Both experiments will make valuable contributions leading to the eventual application of expandable structures technology to future Air Force and NASA missions.

3. Astronaut EVA participation in these experiments is essential to the successful execution of the experiments.

SIMULATION OF PACKAGE TRANSFER CONCEPTS FOR SATURN I ORBITAL WORKSHOP

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SUMMARY: Objective of this program was to investigate the problem of manually transferring massive packages in a zero "g" environment with no mechanical aids except a handrail.

INTRODUCTION

The Saturn I Orbital Workshop (OWS) experiment packages will be stowed in the Multiple Docking Adapter (MDA) during launch. To activate the OWS, these packages must be transferred from the MDA through the Structural Transition Section (STS), the Airlock Module (AM), and a portion of the S-IVB stage LH_2 tank into the crew quarters area of the stage (see Figure I).

Initial design of the OWS provided only a single handrail (fireman's pole) to assist the astronaut in translation and in transfer of the packages. The fireman's pole extended from the S-IVB stage LH_2 tank entry hatch through the crew quarters ceiling hatch and into the crew quarters.

A preliminary task analysis revealed several critical areas requiring simulation and further

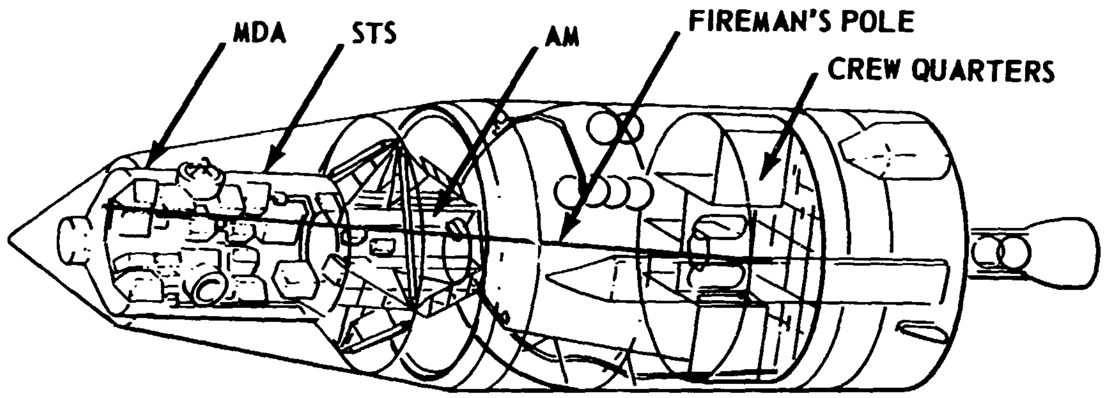


FIG. I - OWS PACKAGE TRANSFER ROUTE

study in order to assure successful performance of the transferral task:

a. The effectiveness of a fireman's pole as a translation aid needed verification.

b. Limitations relative to package mass and configuration had to be established.

c. Effectiveness of package handholds (design and location) required investigation.

d. A basic technique of man/package transfer had to be developed.

SUMMARY OF SIMULATION METHODS EMPLOYED

In order to accomplish the above, a preliminary simulation program utilizing a six-degree-of-freedom simulator and the KC-135 zero "g" aircraft was developed. Techniques, conclusions, etc. established using the six-degree-of-freedom simulator were verified aboard the KC-135.

Data obtained from the preliminary program was sufficient to verify the effectiveness of the fireman's pole as a translation aid, to determine handhold design and location, and to establish a basic man/package transfer technique. Only gross limitations of package mass and configuration could be established from these tests, however.

Analysis and correlation of data from the above simulations indicated that the most important single characteristic of a package which affects its maneuverability is package moment of inertia about the handle. For this reason, an additional neutral buoyancy/KC-135 zero "g" program was designed to obtain data on ease of handling for a wide range of package masses and moments of inertia and to determine the limiting moment beyond which control of the package cannot be maintained.

PRELIMINARY SIMULATIONS

Rationale

This phase of the study was designed as a general investigation of the problem of manual translation and package transfer for the purpose of supporting OWS design. Objectives included developing a basic transfer technique, determining adequacy of the fireman's pole design, investigating necessity of tethers and mobility aids, and determining package size and weight limitations. Developmental work for this phase was conducted using a negator spring six-degree-of-freedom simulator at MSFC. Results were verified under zero "g" conditions in the KC-135 during approximately 150 parabolas.

Method

a. Subjects

A total of three MSFC test subjects, each equivalent to an astronaut in size and physical condition, participated. One of the three took part in both the six-degree-of-freedom simulator and the KC-135 work. Additionally, four members of the MSC flight crew participated briefly in the KC-135 tests. All subjects operated in shirt-sleeves.

b. Apparatus

Apparatus for both the six-degree-of-freedom and the KC-135 simulations was similar, in order that there might be consistency between the tests. Due to stress requirements, however, the KC-135 mockups were considerably more sturdy. Test setup included two 40-inch-diameter hatches separated by a distance of approximately 20 feet and connected by a handrail (fireman's pole) of elliptical cross section (1.25" x 1.75"). Entrance to one of the hatches was enclosed by a "cage" approximately five feet in diameter and four feet long to provide an envelope similar to that of the aft compartment of the AM (see Figure II). Three packages were employed for each simulation:

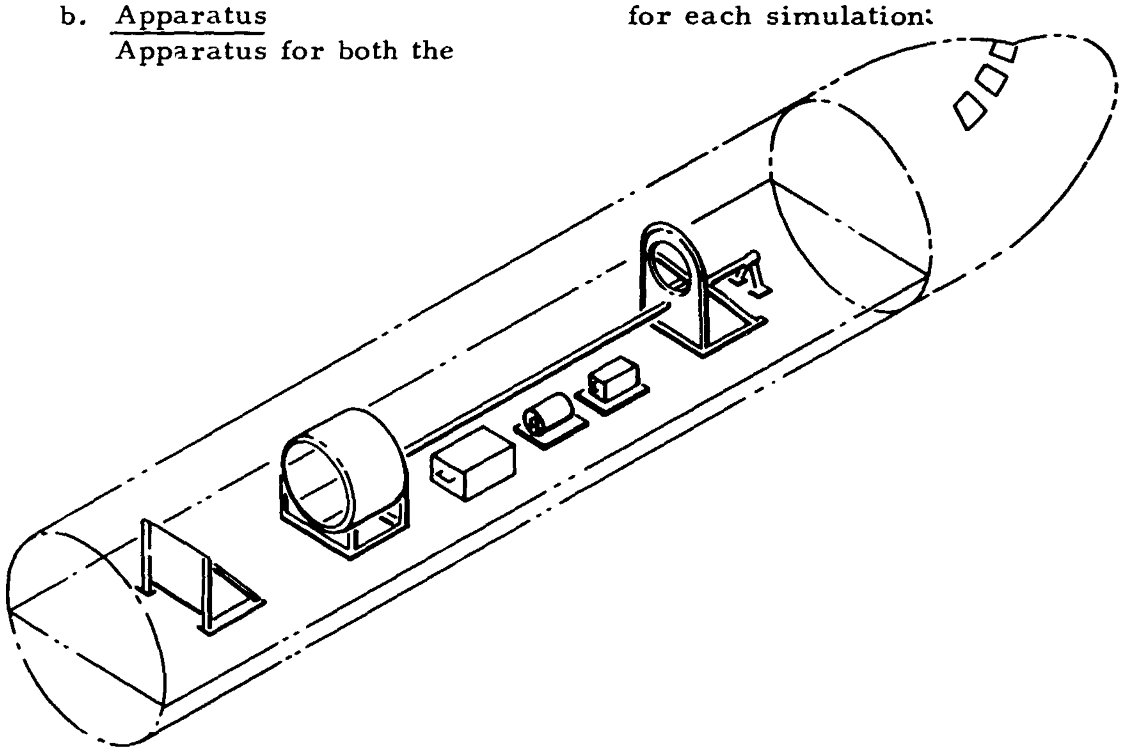


FIG. II - TEST EQUIPMENT INSTALLATION IN KC-135

Package #1 - 12" x 12" x 24" box on a 20" x 30" pallet

Package #2 - 15" dia. x 26" cylinder on a 20" x 30" pallet

Package #3 - 20" x 30" x 40" box on a 30" x 40" pallet

During the six-degree-of-freedom simulations, helium-filled balloons were attached to the packages to approximate weightlessness. For the KC-135 tests, the packages were weighted to provide the following masses:

Package #1 - 60 lbs.

Package #2 - 80 lbs.

Package #3 - 150 lbs.

Tethers, tether rings, temporary storage hooks and other auxiliary equipment were similar between test setups. Still and motion picture cameras were used for data collection.

c. Procedure

In each simulation, the subject initially transversed the pole empty-handed to become acquainted with the equipment and to determine a basic translation technique. Afterwards, for each iteration, the subject transferred one package on a "round trip" through both hatches and along the length of

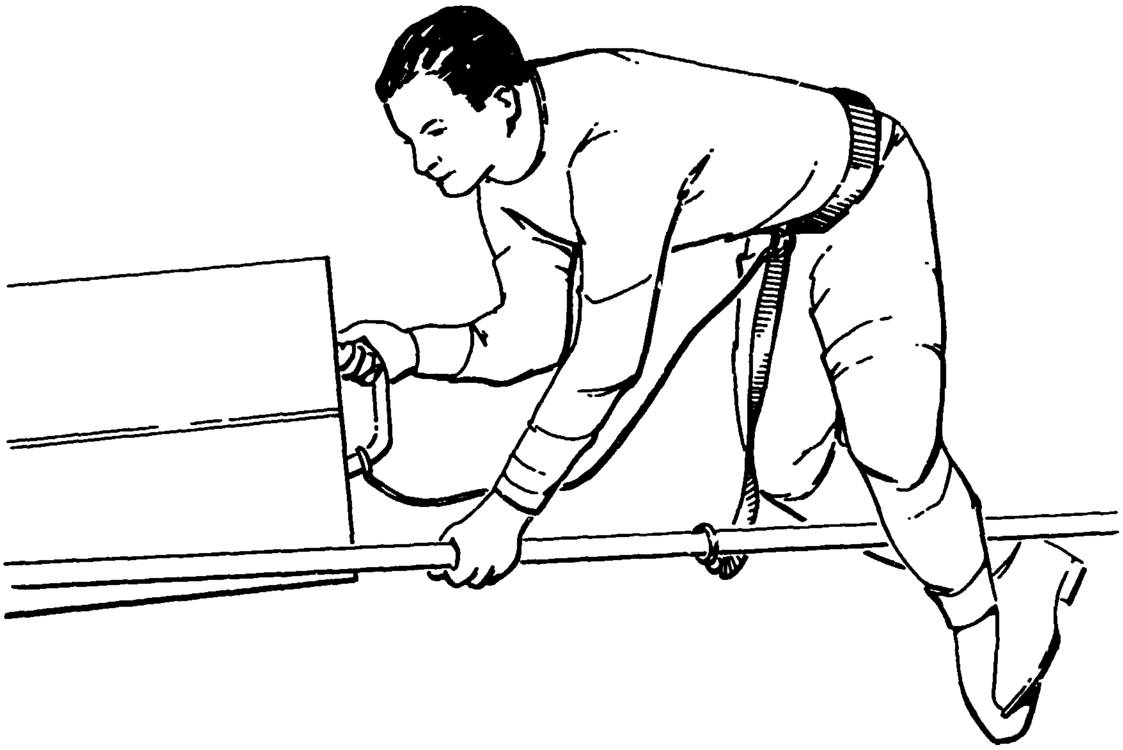


FIG. III - TRANSFER TECHNIQUE (NOTE: TETHERS PROVIDED FOR SAFETY.)

the fireman's pole.

Results

a. Transfer technique

The preferred method of translation, developed in the six-degree-of-freedom simulator and verified aboard the KC-135, is illustrated in Figure III. Major considerations of the method are as follows:

(1) Grasp fireman's pole in one hand.

(2) Grasp package in other hand.

(3) Push package in front of subject for optimum guidance and control.

(4) Lock feet lightly around pole for directional stability. (NOTE: This suggests that translation under suited conditions will be much more difficult than in shirt-sleeves. Also, in the KC-135 the subject's legs tended to involuntarily come unlocked and float apart, which resulted in a loss of directional stability.)

b. Handrail design

Cross section of the fireman's pole was a 1.25" x 1.75" ellipse. This proved to be a definite advantage in preventing rotation around the pole and is the recommended configuration.

c. Tethers

Tethers for connecting man to package, man to handrail, or package to handrail are not required except as a safety measure.

d. Package handhold design and location

Handholds should be located on the package in such a manner that, as the package is "pushed" in front of the man, the force exerted against the handle passes directly through the package center of mass. Generally, this would mean locating the handle at the center of the package side that is directly in front of the man. A contoured pistol grip handle configuration is recommended for adequate control of the package in both pitch, roll, and yaw.

e. Package weight limitations

Subjects were able to safely and accurately transfer the 60 lbm and the 80 lbm packages. The 150 lbm box, however, was too massive to be handled safely by one man within a desirable time. Subjects suggested that approximately 90-100 lbm appears to be a reasonable maximum for one man manual transfer, provided the package center of mass is not more than 14-16 inches from the handhold.

f. Package size limitations

In order that visibility should not be obstructed, the side of the package directly in front of the subject's face should be limited to no more than 20" x 25". Length of the package should be such that package moment of inertia about the handle is acceptable.

Summary

Most of the objectives of this phase of the study were accomplished. As stated previously, however, it was discovered that package manueverability is not a function of either package mass or package size but a combination of both parameters. Thus the hypothesis was formed that the greater the moment of inertia of the package the more difficult its translation and/or maneuverability. Additional testing was suggested to test this hypothesis and to establish a maximum allowable moment about the handle.

PACKAGE MOMENT OF INERTIA SIMULATIONS

Rationale

The purpose of this portion of the simulation program was to explore and refine results of the preliminary tests by investigating package maneuverability in terms of package moment of inertia about the handle. Data was obtained using both a neutral buoyancy simulator and the KC-135 during 77 parabolas.

Method

a. Subjects

A total of five MSFC test subjects participated in these tests, three in the neutral buoyancy simulator and two in the KC-135. Each was equivalent to an astronaut in size and physical condition, and the neutral buoyancy subjects were qualified Scuba divers. One of the KC-135 subjects had participated in the earlier KC-135 tests. Neutral buoyancy subjects operated in Scuba equipment and KC-135 subjects in shirt-sleeves.

b. Apparatus

Test equipment setup for each simulation mode basically consisted of a fireman's pole and a package "receptacle" arranged in such a manner that the package had to be transferred along the pole and maneuvered into the receptacle. Clearance between the inside of the receptacle and the exterior of the packages was approximately 1-2 inches. KC-135 mockups used in the earlier phase of the study were modified to include an extension to the handrail and the receptacle mounted on the simulated airlock. This required the subject to turn the package through a full 180° just prior to insertion into the receiver. (see Figure IV)

Neutral buoyancy packages used were of the following configurations:

Package #1 - 10" x 10" x 20" - 70 lbm - MOI about handle 41 in-lb-sec^2

Package #2 - 10" x 10" x 30" - 110 lbm - MOI about handle 137 in-lb-sec^2

Package #3 - 10" x 10" x 40" -

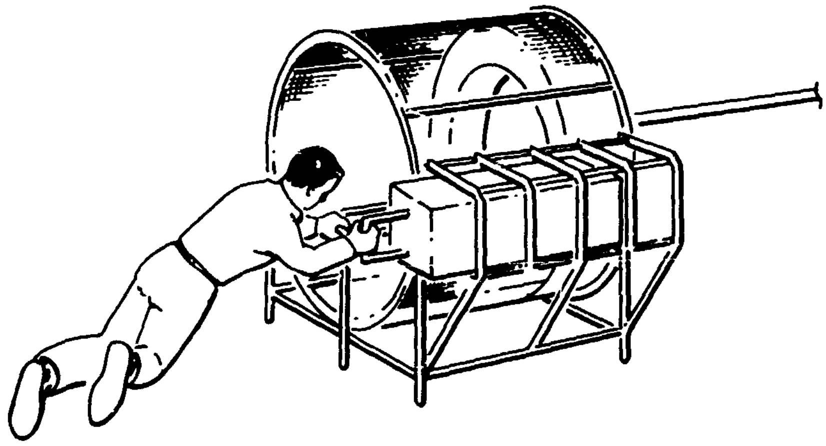
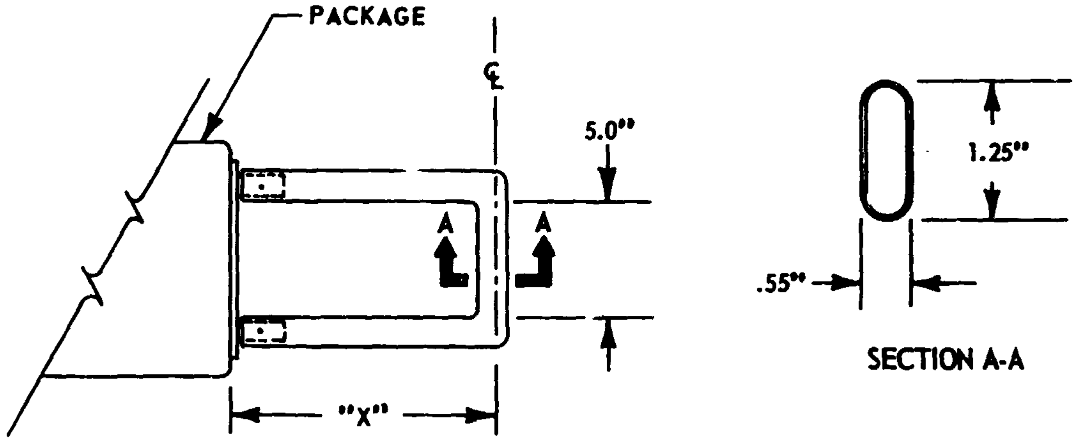


FIG. IV - SUBJECT AND MODIFIED KC-135 MOCKUP

140 lbm - MOI about handle
306 in-lb-sec^2

Each neutral buoyancy package was modified to receive each of five interchangeable handles of different "grip area to mounting surface" lengths. (see Figure V) This enabled the packages to be rapidly modified to vary the moment of inertia while maintaining the same mass. Table I summarizes the combinations which could be achieved.



LENGTH "X" (INCHES)

A= 3
B= 6
C=10
D=16
E=24

FIG. V - INTERCHANGEABLE PACKAGE HANDLES

PKG. SIZE	HANDLE LGT. / PKG. WGT.	PACKAGE MOMENT OF INERTIA (IN-IB-SEC^2) 3"	6"	10"	16"	24"
10"X10"X20"	70 LBS.	35	51	77	127	214
10"X10"X30"	110 LBS.	130	162	215	311	470
10"X10"X40"	140 LBS.	266	318	400	545	775

$MOI = I_0 \bullet + \frac{MK^2}{12g} \bullet I_0$ DETERMINED EMPIRICALLY ON AIR BEARING TABLE

TABLE I - PACKAGE MOMENTS OF INERTIA

c. Procedure

In each simulator subjects were allowed two practice runs to familiarize themselves with the equipment and the transfer technique. Afterwards each iteration required the subject to translate approximately ten feet of the fireman's pole, and insert the package into the receptacle. On the KC-135 this included maneuvering through the simulated airlock and guiding the package through the 180° turn.

d. Data

Data recorded included movies, time required to complete each task, and subjective evaluation of each iteration. The subjective evaluation was based on a five point rating scale:

1 . . . Excellent

2 . . . Good

3 . . . Fair - some reservations

4 . . . Poor - many reservations

5 . . Unacceptable

Results

This phase of the testing program verified that a combination of the package parameters size and mass (moment of inertia) is a very significant factor affecting package maneuverability. Average subjective ratings for the various moments investigated are plotted in Figure VI. This plot assumes a reasonable package configuration for adequate vision and a reasonable amount of time within which to

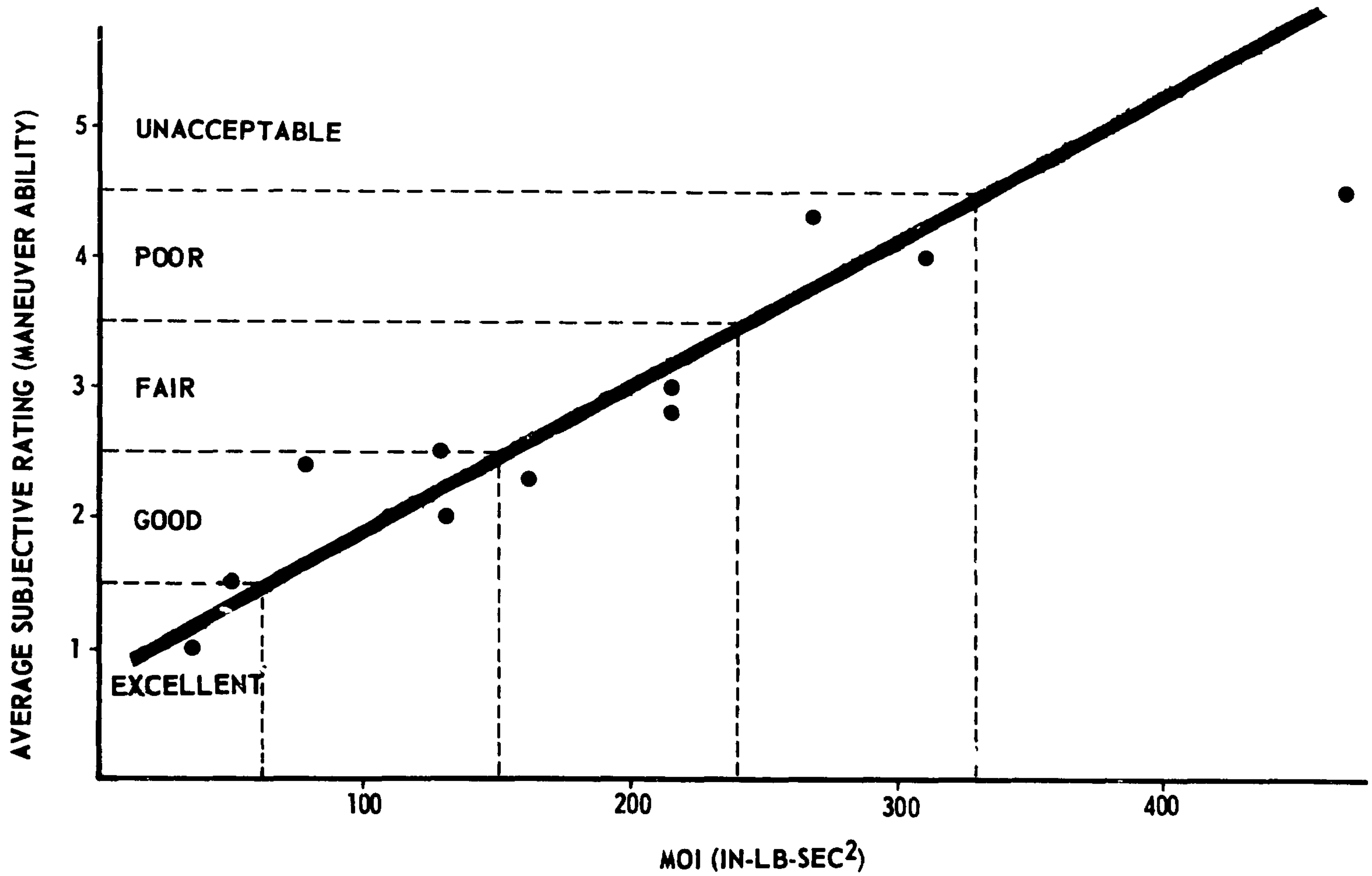


FIG. VI - SUBJECTIVE RATING VS MOMENT OF INERTIA complete a transfer operation. (With unlimited time restraints and with a package of small dimensions, the upper limit on package mass would be considerably higher.) The moment of inertia limits presented in Table II were derived from the data plotted in Figure VI. These limits suggest that a package with MOI less than 65 in-lb-sec^2 can be easily maneuvered and precisely positioned under zero-g conditions. If the requirement exists to transfer a package through a large open area where few positioning constraints exist, approximately 250-300 in-lb-sec^2 can be tolerated. A value of 300-350 in lb-sec^2 appears to be an upper limit for one man transfer within a reasonable time frame.

MANEUVERABILITY	MOI
EXCELLENT	0 - 65
GOOD	66 - 150
FAIR	151 - 240
POOR	241 - 330
UNACCEPTABLE	331 →

TABLE II - MOI LIMITS

PLANNED WORK

Experiments conducted to date have provided significant data on man's ability to transfer massive packages in a zero-g environment. However, many additional tests, as well as experience gained from actual EVA/IVA, are required before a reasonable degree of confidence can be attached to results obtained. Planned experimental work at MSFC includes the following:

a. An investigation to determine maneuverability of a very heavy (approximately 300-500 lbm) package with its handle at the package center of mass.

b. More work to validate the recommended pistol grip handle configuration.

c. A study to determine upper limits of moments of inertia about each of the pitch, roll, and yaw axes.

EVA REQUIREMENTS IN SUPPORT OF SCIENTIFIC AND TECHNICAL EARTH ORBITAL SPACE ACTIVITIES

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SUMMARY: EVA requirements in support of earth orbital experiment activities of the scientific and technical community projected for the 1970 - 1980 timeframe indicate a potential for extravehicular activity to support stated goals. The requirements for the EVA activities have been addressed by the NASA and was the subject of a 14-month contractual effort by the Space Division of North American Rockwell Corporation.

This study indicated the broad end objective of the following disciplines: astronomy, biosciences, physical sciences, communications and navigation, meteorology, biomedical behavior, earth resources, advanced technology, and orbital operations. These objectives were used to determine the supporting scientific and technological experiments in each discipline; and criteria were established to reduce the large number of identified experiments to a representative, manageable number to be analyzed in detail for establishing EVA requirements. As a result of the detailed analyzes of representative experiments from each of the scientific/technical disciplines, EVA astronaut task and functions were identified.

The functional task was made to eliminate redundancy and to develop EVA requirements in terms of procedures and equipment. These, in turn, were used as a basis for two experimental EVA programs on NASA Form 1346. These experiments are directed toward development and verification of operational EVA capability required to support the S/T experiments.

This paper will also discuss the ground simulation plan and experimental approach.

INTRODUCTION

This paper will discuss a study designed to determine the EVA capability required to support the earth orbital scientific and technical experiments projected for the 1968 - 1980 time period. The objectives of the study were: (1) to identify the scientific and technical experiments requiring EVA support in the 1968 through 1980 period, (2) to define the astronaut EVA functional capabilities in terms of techniques and equipment necessary to provide that EVA support, and (3) to define the experimental EVA program that must be conducted to develop and demonstrate the EVA functional capability.

The major phases of the study were organized according to the study objectives and were accomplished in a logical sequence which began with establishment of a scientific and technical experiments baseline and proceeded through definition of two proposed experimental EVA experiments.

The experiments baseline analysis, which occupied a major portion of the study, consisted of listing all the proposed earth-orbital scientific and technical experiments. Sources for the proposed experiments included previous NASA in-house and NASA contractor studies. The experiments were assembled by scientific and technical discipline, time period, and potential requirement for EVA support. It was found necessary to reduce the large number of proposed experiments requiring EVA during the principal period of interest (1971-1974) to a manageable number for detailed analysis. Accordingly, 16 of the best defined and most realistic were selected as representative and analyzed in depth to establish the validity of the EVA requirements and to define EVA functional capabilities in terms of techniques and equipment required to perform the EVA.

Because the EVA functional descriptions contained large quantities of detailed data, a special analytical technique was developed. Called EVA Building Blocks, the technique consisted of assembling EVA functional descriptions by means of decimally coded numbers in such a manner that a complete EVA task could be described by a set of building block numbers. The building block technique, which permitted statistical treatment of EVA capabilities data, was one of the more significant analytical tools developed in the study. This analysis resulted in delineation of the EVA techniques and equipment required to support the scientific and technical experiments. Existing capabilities were established in accordance with current equipment status, and the demonstrated EVA techniques were defined and compared with required capabilities. The difference or "delta" then represented the capability that must be developed for adequate support of earth orbital scientific and technical experiments.

The delta capability requirement formed the basis for definition of two experimental EVA experiments that are designed to develop and demonstrate the necessary EVA capability. These two experiments, which were defined in accordance with the third study objective, represent a major output of the study. They were structured about the three basic EVA functions: translation, egress and ingress, and work performance. One experiment specifically involves translation of EVA astronauts and astronauts with cargo but includes demonstration of astronaut rescue techniques. The second experiment is designed to demonstrate the EVA functions of egress and ingress and work performance. Both experiments are written in accordance with NASA Form 1346 and define ground simulation and training requirements as well as in-orbit experiments. Each experiment consists of a set of subexperiments, or experiment options, so designed that the options may be performed separately on different missions, or sequentially during a single mission.

EXPERIMENTS BASELINE PROGRAM

Baseline Selection Criteria

The primary objective of the experiments baseline activity was to produce a framework of earth orbital scientific and technological space activities upon which a time-phased baseline extravehicular activities program (EVA) could be developed. To be considered valid, the scientific and technological experiments contained within the baseline must have their relationship to the overall national space goals and objectives identified in terms of their contribution or usefulness. In addition, the experiments baseline must reduce the large number of candidate manned earth orbital experiments to a smaller, more manageable number of scientific and technological experiments exhibiting typical EVA operational requirements in order that EVA task requirements may be analyzed in sufficient detail. The traceability of experiments through the various levels of reduction in the experiments baseline must also be clearly evident and logical for credibility to be maintained.

Top-Down Approach

To accomplish this objective, goals and objectives were delineated in a "top-down" manner, extending from national space goals through discipline and subdiscipline objectives and knowledge requirements and finally to intestigation objectives. Measurement requirements were then established to satisfy the investigation objectives. In order to keep redundancy and overlapping activities to a reasonable minimum, space objectives relating to national space goals were organized according to uses or "user" groups such as scientific or technical disciplines, societies or organizations. This also allowed maximum use of existing reference material. This resulted in nine scientific and technological disciplines, as shown in Figure 1, for which the end objectives were defined. Discipline end objectives, together with subobjectives, were defined for each discipline.

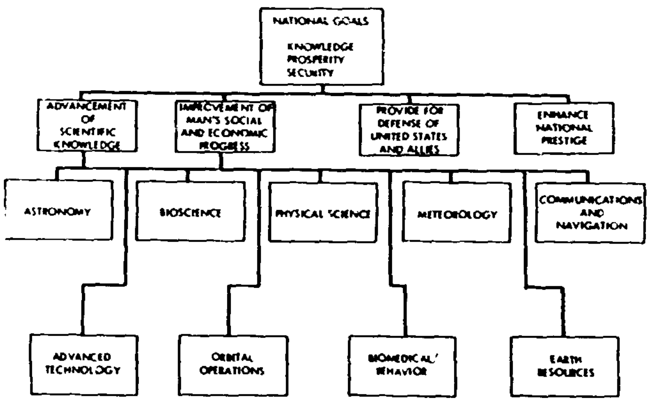

FIG. 1 - NATIONAL SPACE PROGRAM GOALS

Potential experiments and associated equipment were identified through use of experiment data sources. The observable phenomenon or measurement capability was defined for each experiment. The correlation between the experiments and the investigational objectives was established by relating the observable phenomena or measurement capability of the experiments to the measurement requirements of the investigation objectives.

The first level experiment baseline resulting from the top-down analysis was, however, insufficient to serve as a means of defining EVA requirements within reasonable limits of the study. The number of experiments was unwieldly, and many of the experiments were not defined in sufficient detail to readily allow more detailed analysis. Therefore, the experiments baseline was defined in three additional levels of detail.

Table I. Representative Scientific/ Technical Experiment Packages

DISCIPLINE	NUMBER	EXPERIMENT TITLE
ASTRONOMY	AE0201RE	1-METER NON-DIFFRACTION-LIMITED TELESCOPE
	AE0502RE	1-METER TELESCOPE, ADVANCED PRINCETON EXPERIMENT PACKAGE
	AE020 RE	ATM/80 CENTIMETER SOLAR TELESCOPE
	AI0304RE	X-RAY FOCUSING TELESCOPE
	AI0401RE	EMR PAYLOAD (OPTION B)
	AG0101YY	10-KM INTERFEROMETER
BIOSCIENCES	BA0101ID	SOFT CAPTURE AND CULTURE MICROORGANISMS IN NEAR-EARTH ORBIT
	BB0102ID	EFFECTS OF SPACE ENVIRONMENT ON BACTERIAL SPORES
	BI0111KZ	PRIMATE BIOMODULE ("B10 A")
	BI0110KZ	MULTI-PURPOSE BIOLOGICAL PACKAGE
COMMUNICATIONS AND NAVIGATION	CA0601ID	30-45 METER ANTENNA TECHNOLOGY AND LARGE POWER SUPPLY
	CA0101DD	LONG-BOOM INTERFEROMETER
PHYSICAL SCIENCES	SI0202AZ	SPACECRAFT PHYSICAL ENVIRONMENT
	RE0103EA	CONJUGATE AURORA AND AIRGLOW
ADVANCED TECHNOLOGY	LG0000IA	LARGE STRUCTURES DEPLOYMENT
ORBITAL OPERATIONS	SG4012LZ	EARTH ORBITAL SPACE STATION

Each level progressively contains fewer experiments, each defined in greater depth with respect to prospective experiment tasks and associated EVA functions. The final fourth level experiment baseline contains a manageable number of experiments (Table I.) defined in sufficient detail so that they can be verified as representative of operational EVA requirements and can, therefore, serve as a baseline for the subsequent more detailed analysis of EVA tasks with associated EVA techniques, equipment, and performance requirements.

EXPERIMENT DEFINITION

Each of the 16 scientific and technical experiments selected as representative was analyzed in detail to identify the specific EVA requirement and to establish whether the requirement was valid or whether the experiment function requiring EVA support could be accomplished in some other logical manner. The details of the experiment definition analysis are explained in a following paper, so will not be elaborated on here. Briefly, the analysis consisted of examining the basic experiment to assure that the experiment concept was realistic for the time period and that the equipment would satisfy the measurement requirements of the scientific objective. The experiment operational configuration then was analyzed to determine its physical interfaces with the spacecraft and spacecraft systems and to assure that the experiment could be performed in the space environment. The in-orbit experiments procedures necessary to conduct the experiment were defined to identify man's role, particularly man's EVA role, in performing the experiment. Once the EVA support requirement had been identified, the experiment procedures were reexamined in depth and possible procedural and equipment modifications were postulated which would eliminate the necessity for EVA to provide assurance that the EVA support requirement as defined in the original scientific and technical proposal was valid. In many instances, a modification of experimental procedure resulted in reduction of the number of EVA's, the number of EVA hours, or more efficient use of astronaut EVA time.

EVA OPERATIONS ANALYSIS

Because of the large volume of descriptive detail involved in the preceding experiment definition analysis, it was necessary to develop a special technique for handling and analyzing the data. The analytical method consisted of identifying basic sets of EVA functions, subfunctions, techniques required to accomplish the functions, equipment necessary to perform the EVA functions/technique, and, finally, a set of gross performance measures. These functions, techniques, etc., were grouped into logical combinations that could be used to describe a discrete EVA activity.

Analysis of EVA Tasks

Figure 2 illustrates the analytical procedure used in the EVA operations analysis of individual

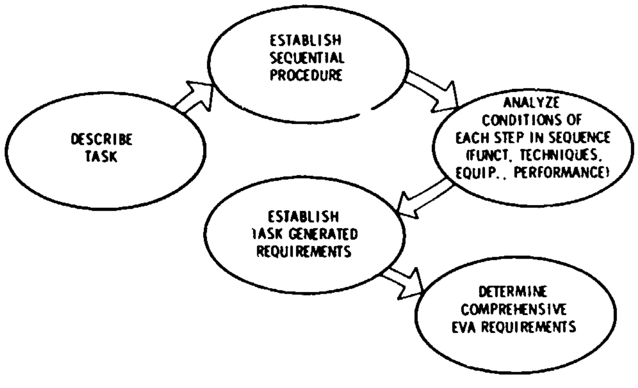

FIG. 2 - TYPICAL ANALYTICAL PROCEDURE

EVA tasks. This analysis is initiated with the description of the EVA tasks to be performed. To further define this task, a functional astronaut procedure required to perform the task must be specified. For each of the steps in this sequential procedure, the conditions which apply to that step must be specified. This specification requires listing of the astronaut function being performed, the technique the astronaut uses to perform that function, articles of EVA equipment required to support the astronaut, and the astronaut performance capability. Following the analysis of each step in the sequence for a specified EVA task, the requirements generated by this analysis can be collected. Finally, after completion of all analyses for all EVA tasks considered, a comprehensive listing of EVA requirements may be generated.

Data Retrieval Task

As an example of the application of this procedure, the retrieval-of-data task in support of the one-meter telescope will be considered. Figure 3 illustrates the sequential steps required to perform this data

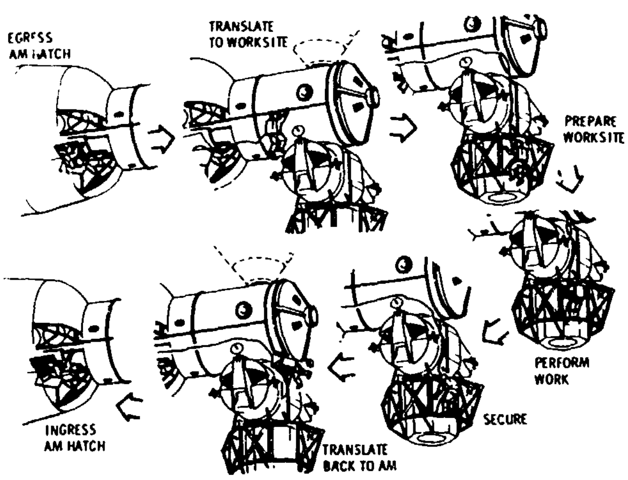

FIG. 3 - EVA ACTIVITIES - MODULE REPLACEMENT

retrieval. The astronaut first egresses an airlock on the space station and then translates to the worksite. He prepares the worksite for the data retrieval operation; then performs the actual work. Following retrieval of the data, he secures the worksite for further experimental work and returns to the airlock.

The first step in this sequential procedure, the egress of the airlock with two film cassettes, contains three separate substeps, as shown in Figure 4.

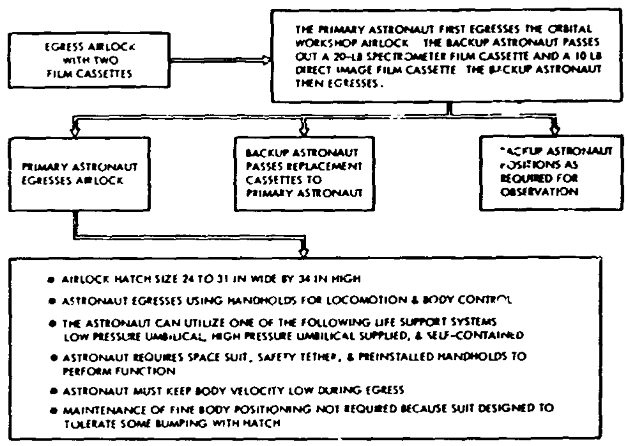

FIG. 4 - ANALYZE CONDITIONS OF STEPS

The first substep deals with the egress of the airlock by the primary astronaut, the second with transfer of two film cassettes, and the third with back-up astronaut positioning. The first of the three functions, the egress of an airlock by an astronaut, was encountered a large number of times during the analysis of many EVA's. The analysis of the conditions of this step, as prescribed in Figure 4 requires specification of the function performed, astronaut technique, EVA equipment, and performance level. The analysis of this specific step includes specifications of the airlock hatch characteristics, determination of the technique used for the egress, and specification of astronaut equipment, such as life support, spacesuit, etc. In this case, the appropriate performance emphasis refers to astronaut body velocity and body positioning maintenance. As is shown by this discussion, the analysis of this rather simplified substep results in the generation of large amounts of data.

This type of analysis leads to three basic problems. The first of these is that the generation of the large amounts of data during analysis is a rather tedious procedure. Also, there is repeated usage of much of the data in the analysis, such as that data pertaining to egress of an airlock. Finally, there is a very large accumulation of data. There are three possible solutions of these problem areas. A shorthand system eliminates some of the tedium associated with this analysis. Consistency during the analysis is necessary so that each time similar conditions are generated. Finally, the large accumulation of data requires that a categorization system be developed so that the analyst can very rapidly scan like amounts of data to determine EVA requirements upon completion of the task analysis portion of his effort.

Building Blocks Concept

Standardized data blocks, based upon the requirements for analytical tools, provide a necessary solution. Also, these data blocks, since they apply to analysis of sequential steps in an EVA task procedure, can be used sequentially to identify possible ways to perform an EVA. During this study, the term "building blocks" has been applied to the standardized data blocks generated.

The building blocks format contains four items of information required in the analytical procedure; these are primary astronaut function being performed, the technique used to accomplish the function, specifications of EVA equipment used, and estimates of the astronaut capabilities.

Because the building blocks are used to describe the functional astronaut performance of all possible EVA operations, it is mandatory that these building blocks be comprehensive in nature. Three basic categories of astronaut functions have been used to describe astronaut functions. These categories, as shown in Figure 5, are: egress and ingress, translation, and work performance. The categorization system

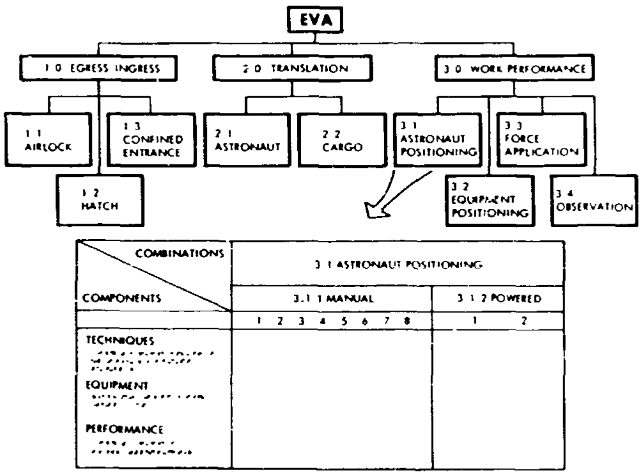

FIG. 5 - BUILDING BLOCK CONSTRUCTION

used assigns decimal numbers 1.0, 2.0, and 3.0 to these three basic areas. To further describe the astronaut functions being performed, these basic functions were subdivided. In the egress and ingress area the next subdivision used is airlock hatch, or confined entrance. In the area of translation the next subdivision differentiates between translation of an astronaut and translation of cargo. Finally, in the area of work performance, the next level subdivision refers to astronaut positioning, equipment positioning, force application, or astronaut observation. These latter four areas are assigned decimal descriptors of 3.1, 3.2, 3.3, and 3.4, respectively.

Since these latter descriptors are not unique descriptions of astronaut functions which may be performed, further subdivision is necessary. Illustrated in Figure 5 is the subdivision of area 3.1, astronaut positioning. It is shown that astronaut positioning may be performed in either a manual mode or a powered mode. These modes carry the decimal descriptors of 3.1.1 and 3.1.2. Also, as shown in this matrix, there are various specifications of particular techniques and equipment which may be used to perform this astronaut function. For each combination of techniques and equipment, there is a specified astronaut performance level. In this case there are eight unique specifications of techniques and equipment which may be used to perform manual astronaut positioning.

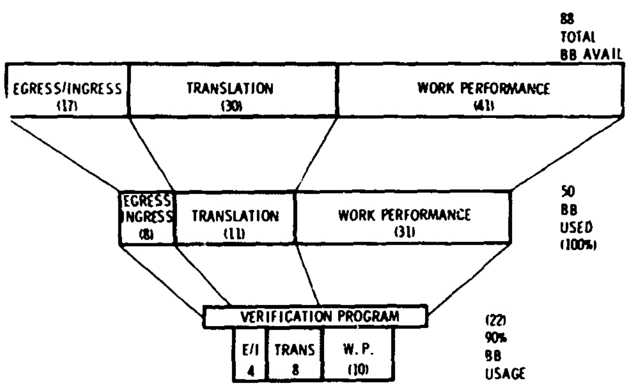

FIG. 6 - BUILDING BLOCK USAGE

An example of one of these eight different building blocks in the area of manual astronaut positioning is shown in Table II. This building block specifies the technique used and the specific items of EVA equipment in Building Block 3.1.1.3. For this building block, the performance requirements specified include a 0- to 4-foot radius positioning reach and fine positioning accuracy.

Table II. Manual Astronaut Positioning, Building Block 3.1.1.3

COMPONENT	DESCRIPTION
TECHNIQUE	MANUAL ASTRONAUT POSITIONING
EQUIPMENT	SPACESUIT ASTRONAUT SAFETY TETHERS LIFE SUPPORT SYSTEM RESTRAINTS PREINSTALLED HANDHOLDS OR HANDRAILS WAIST RESTRAINT - FLEXIBLE (DUAL) PREINSTALLED FOOT RESTRAINTS
PERFORMANCE REQUIREMENTS	POSITIONING REACH (0 TO 4 FEET RADIUS) POSITIONING ACCURACY LINEAR-FINE 0 TO 2 INCHES ANGULAR-FINE 0 TO 5 DEGREES

Application of Building Blocks

Because the building blocks provided for description of EVA tasks by means of a numerical code, they were readily amenable to analysis and were utilized to generate the overall EVA requirements which were categorized according to a requirement that the astronaut have a certain capability to perform functions and to the requirements for EVA equipment that permit the astronaut to perform the functions. These requirements were then statistically analyzed to determine the frequency of occurrence for each requirement according to the number of times a capability was required in the performance of the representative EVA tasks. This statistical analysis was used as a basis upon which to structure experimental EVA programs designed to demonstrate and develop the EVA capability required to support early requirements. Because it was impossible to demonstrate that all requirements could be met, it was determined that those requirements which would satisfy at least 90 percent of the requirements generated by analysis of the early experiments would be used in defining the experimental EVA program.

Eighty-eight building blocks were assembled to describe all possible EVA's; however, the EVA operations analysis revealed that only 50 building blocks were actually required to satisfy all the EVA support needs of the 16 representative scientific and technical experiments. Furthermore, by substitution of building blocks to eliminate those less frequently used, it was found that 22 building blocks accounted for more than 90 percent of EVA usage. Accordingly, the two proposed experimental EVA experiments were structured to develop and demonstrate the EVA capabilities, techniques and equipment making up these 22 building blocks.

EVA REQUIREMENTS

EVA requirements may be categorized according to specification of functional performance requirements or equipment requirements. The functional performance requirements impact heavily upon the generation of equipment requirements and will be described initially. EVA functional performance requirements are categorized according to the same functional descriptions which were used in the generation of the building blocks. These areas are egress and ingress, translation, and work performance. The functional performance requirements can be generated by referring back to the EVA task analyses which were prepared and by specifying the required levels of performance which were most critical in these analyses. An example of one technique for displaying these data is shown in Table III for two astronomy experiments.

Egress/Ingress

Two basic egress/ingress categories were used for specification of astronaut performance requirements. These are astronaut-only egress/ingress and equipment egress/ingress. These functions must be performed from an airlock and from a command module hatch. The most difficult problem in astronaut-only egress/ingress is presented by the hatch situation. The volume available within the CM is quite limited, and the astronaut and his backup must don their equipment prior to initiation of the EVA. It has been determined that the actual emergence of the astronaut will present no problem.

In the category of equipment egress/ingress, it was found that light equipment was utilized most frequently. The technique utilized for this procedure requires that the back-up astronaut pass the equipment out to the primary astronaut after his egress. Equipment egress/ingress from the command module hatch is very limited because of the weight and volume limitations of the system. Generally, the equipment weight will not exceed 25 pounds. For the airlock, cases were found in which the equipment to be handed out weighed up to 70 pounds. The types of restraint system to be utilized in this process must be evaluated in the experimental program.

Work Performance

Among the most common astronaut functions required in support of operational EVA are astronaut positioning and repositioning at worksites. These functions include positioning using handholds only, waist restraints only, and combinations of waist restraints and foot restraints. Repositioning of the

Table III. EVA Data Matrix

PERIOD 71 - 74	75 →	DISCIPLINE: EXPERIMENT LIST (ASTRONOMY)	EVA REQUIREMENTS: REQUIREMENTS	FIRM - ●	POTENTIAL - O	DURATION - HOURS	FREQUENCY/MISSION	TRANSLATION: DISTANCE - FEET	MASS - POUNDS	MANUAL AIDS	SOARING	MANEUVERING DEVICE	TRANSPORTER	SUIT: SOFT	HARD	LIFE SUP.: UMBILICAL	SELF-CONTAINED	EGRESS/INGRESS: CM HATCH	AM HATCH	OTHER	WORK PERFORMANCE: POSITIONING - FEET	REACH - FEET	MASS HANDLING - LBS	HANDS USED	FORCES (1)	MANIPULATION (2)	ACCESS (3)	MEASUREMENT (4)	AIDS (5)	TOOLS (6)	WAIST RESTRAINT (7)	FOOT RESTRAINT	EQUIPMENT RESTRAINT	HANDHOLD RESTRAINT	PRESENT STATUS: GEMINI AAP-1-4 O	NEED: DELTA NEED □	COMMON AF: EVA AREA (8)	WITHIN EXPERIMENT	WITHIN DISCIPLINE	BETWEEN DISCIPLINES	PRIORITY
X		A10304RE, X-Ray Focussing Telescope	Correct deployment malfunction		O	4		30	10	X					X		X	X			4	4	10	2	H	F	A	.1	M	P	R	X	X	X							
			Remove turret restraints	●				10	<20	X					X		X	X			2	4	<20	2	L	F	A	.1°	O	H	W	X	X	X							
			Align laser system	●				10		X					X		X	X			4	4		2	L	F	E	.1°	O		V	X		X		Trans					
			Free wheel & c/o turret	●				10		X					X		X	X			2	2		2	M	F	E	.1°	O		V	X		X		&					
			Check out lens alignment	●		2.3		10		X					X		X	X			4	4		2	L	F	E	.1°	O		V	X		X		work	MH				
			Monitor data link response	●				10		X					X		X	X			2	2		2	L	F	E	.1	O		V	X		X	0	perf.	O				
			Replace ACS units	●				30	125	X			X		X		X	X			>18	>4	82	2	M	F	A	.1	M		W	X	X	X			AY				
			Replace solar panels	●		15.2		30	43	X			X		X		X	X			>18	>4	43	2	M	F	A	.1	M		W	X	X	X			R				
			Replace sensors	●				30	88	X			X		X		X	X			2	2	88	2	L	F	E	.1	M		V	X	X	X							
			Replace batteries	●				20	100	X			X		X		X	X			2	2	100	2	M	F	A	.1	M		F	X	X	X							
X		AG0101YY, 10 Km Interferometer	Correct Deployment malfunction		O	4		15	31	X			X		X		X	X			4	4	31	2	M	F	A	.01	M		V	X	X	X							
			Correct flight control malfunction		O	4		10	<20	X					X		X	X			2	2	<20	2	L	F	E	.01	M		V	X	X	X		Trans					
			Replace 8 solar panels	●				35	85	X			X		X	X		X			2	2	60	2	M	F	A	.01	M		R	X	X	X		& work	MH				
			Replace 6 dipole modules	●		8.5		15	31	X			X		X		X	X			4	4	31	2	M	F	A	.01	M		V	X	X	X	0	perf.	O				
			Refuel ACS tanks	●			1	15	25	X		X			X		X	X			4	2	25	2	M	F	A	.1	M		V	X	X	X			AY				
			Replace batteries	●		2		10	25	X			X		X		X	X			2	2	25	2	M	F	E	.01	M		V	X	X	X			R				
			Replace ACS regulators	●		4		10	<20	X			X		X		X	X			2	2	<20	2	L	F	E	.01	M		V	X	X	X							

LEGEND

(1) FORCES: LIGHT - L, MEDIUM - M, HEAVY - H

(2) MANIPULATION: FINE - F, GROSS - G

(3) ACCESS DEPTH: ARM - A, ELBOW - E, HAND - H

(4) MEASUREMENT: ERROR, 1/100TH IN., 1/10TH DEG.

(5) AIDS: OPTICAL - O, MECH. - M, ELECT. - E, RADIATION - R

(6) TOOLS: POWER, MECH., HAND

(7) WAIST: FLEXIBLE - F, RIGID - R, VARIABLE - V, PLATFORM - W

(8) AREA: LOCOMOTION - L, MAT'L HDL'G - MH, ASSEMBLY/DISASSEMBLY - AY, ALIGN, TEST, OPERATE - O, RESCUE - R

astronaut at the worksite generally was an angular repositioning about a foot-fixed position. The most frequently encountered angular movement was a pitch movement of approximately 30°. Roll and yaw repositioning were found to be required for the astronaut to perform work at a large area worksite. These movements also were found to be approximately 30°.

The astronaut must be capable of restraining himself at several different worksites during the course of a single EVA. Consequently, the worksite restraints must not be one-time applications; and the attachment and detachment must be simple operations. Finally, it was found that for contingency worksites the astronaut should possess the capability to install his foot restraints while using either waist restraints only or, more commonly, handholds only. The astronaut foot-restraint system is required in these cases to be transported on an astronaut cargo harness from the point of egress to the worksite.

Astronaut positioning functions utilizing a powered mode were also required and were accomplished with a stabilized maneuvering unit (SMU). After positioning himself near the worksite the astronaut must be able to attach worksite restraints for the SMU if he is to apply forces during his work. The specialized restraints will generally be used to attach the unit at a contingency worksite, i.e., one at which no provision has been made for restraint anchor points.

In most of the EVA tasks analyzed, the astronaut must be able to temporarily position various equipment at the worksite. Some of this equipment may be attached at worksites that have been previously prepared with attachment points. Other worksites will be contingency sites with no preparation. In a manual mode it was found that the items of equipment to be temporarily positioned ranged from very small packages to bulky racks containing approximately 60 pounds of equipment. Equipment restraints considered range from adhesive or velcro patches to variable or fixed mechanical connections. The patch type restraints require less effort on the part of the astronaut but are limited in capability. Where equipment weights exceed 20 pounds, mechanical restraints should be used.

During tasks in which the astronaut utilizes the SMU for transporting both himself and his equipment, the equipment can generally be restrained on the SMU until needed.

The majority of requirements for operational EVA capability pertain to the development of unassisted astronaut force application techniques. Generally, the astronaut is required to exert light-to-medium forces while manipulating light modular packages within confined areas. The packages to be so manipulated usually weigh between 10 and 15 pounds, and the astronaut may utilize dual waist restraints and foot restraints while performing this activity. Critical to the development of this capability is determination of astronaut reach limitations and ability to perform manipulative actions at extreme reaches.

It has been found that the astronaut must have much freedom in his arm and body movements while performing the activities analyzed. This need for freedom of movement is noted primarily in the removal of modules from storage. In many of the long-duration EVA tasks the astronaut is required to possess great endurance also. Some of these EVA's approach four hours, and the astronaut is required to perform light-to-medium force applications as well as repeated translations. This work involves metabolic heat loads of approximately 2000 Btu's per hour.

Manual Translation and Cargo Transfer Requirements

Manual translation and cargo transfer requirements are occasioned by: (a) EVA support requirements of scientific/technical experiments and space station operations in the early time period, (b) the requirements to support S/T experiments in separable modules (occurring in the mid and later time period, and (c) by the necessity to have an EVA astronaut rescue/retrieval capability.

The manual translation and cargo transfer functions were found to be required at least once for each of the 16 S/T experiments analyzed (see Table IV). There were 53 total manual translations required of the astronaut travessing up to 60 feet. Thirty-nine of these translations required him to transfer cargo packages weighing up to 85 pounds.

Table V provides a summary of the manual translation tasks which require the astronaut to transfer cargo. The information in this table refers to just the experiments in the astronomy discipline and indicate, in addition to performance requirements, equipment requirements which are specific to the translation task. The translation column refers to the traverse path (L. S. CM-WS-CM is command module to work station to command module). The other columns are self-explanatory.

Tables VI and VII are summaries of the performance factors and equipment required to accomplish the manual translation and cargo transfer function to support the 16 scientific/technical experiments analyzed in this study.

EVA EQUIPMENT REQUIREMENTS

The EVA functional performance requirements definition formed the basis for establishing EVA equipment requirements. Each basic function, together with the function performance level, was examined from the standpoint of the equipment required by the EVA astronaut to perform the function at the necessary performance level. Many of the equipments found to be required also were found to be non-existent or at least not to exist with the capability level necessary to adequately perform the required EVA function. Therefore, equipment requirements were defined largely in terms of general equipment concepts and associated performance requirements.

Spacesuits

The requirements for the development of spacesuits have been separated into two basic categories. The first category is basic work functions, such as data retrieval, which are not exceedingly complex or of long duration. In the performance of basic work functions, it is required that the astronaut spacesuit allow significant hand dexterity. The present pressurized gloves allow the astronaut at best a 65 percent capability to manipulate one-half inch pins compared to the same manipulation with a bare hand. This type of limitation impacts on the design of the scientific experiments which the astronaut can support. These experiments must provide sufficient size objects that the astronaut can handle with the present glove system.

The suit should be compatible with umbilical supply provisions to reduce the expendables required to support EVA missions. Generally,

Table IV. Translation Types Versus Discipline

DISCIPLINE	ASTRONOMY	BIO. SCI.	PHYS. SCI.	COMM/NAV.	ORB. OPS.	LARGE ST.
Experiments	6	4	2	2	1	1
EVA's Analyzed	12	5	3	4	3	1
Manual Translation Occurrences (53 total) (Up to 60 Feet)	27	9	2	11	4	-
Manual Translation With Cargo (39 total) (Up to 85 Pounds)	19	3	4	3	3	2

Table V. Astronomy Task Translation Summary

Experiment	Task	Translation	Carried	Performance	Special Equipment
1 M APEP	R&R film	CM-WS-CM	A + 25 lb	20', 20', 1 fps	Cargo harness
	R&R elec.mod.	CM-WS-CM	A + 10 lb	20', 20', 1 fps	Cargo harness
	Rep. drive	CM-WS-CM	A + 10 lb	20', 20', 1 fps	Cargo harn., pwr. tool
1 M NDL	R&R film	AL-WS1-WS2-AL	A + 20 lb + 10 lb	30', 8', 38', 1 fps, struct. members	Cargo harness
ATM 80 SOLAR	Act. focus	AL-WS-AL	A + 5 lb	40', 40', 1 fps	Cargo harn., tool kit
	Return film	AL-WS1-WS2-AL	A, A + 14 + 26.5 + 13.5 lb	43', 5', 43', 1 fps	Cargo harness
EMR	R&R film	AL-WS1-WS2-AL	A + 10 + 10 + 10 lb	48', 20', 68', 1 fps, frag.str.	Cargo harness
X-RAY TEL	Correct malfunction	CM-WS1-CM	A + 10 lb + Batt.	30', 30', 1 fps thru struct.	Cargo harness, power tool
	Act. & C/O Turret	CM-WS-CM	A, A + 20 lb	10', 10', 1 fps	Cargo harness
	Rep. ACS	CM-SM-WS-SM-WS-CM	A, A + CL	15', 35', 35', 35', 20', 1 fps thru struct.	Clothesline
	Rep. battery	CM-WS1-WS2-SM-WS2-CM	A, A + CL	20', 5', 40', 40', 25', 1 fps thru struct.	Clothesline
10 KM INT	Replace solar panels	CM-SM-WS1-WS2-WS3-WS4-SM-CM	A, A + 60 + 25 lb	10', 25', 3', 3' 3', 25', 10', 1 fps, asfps	Cargo harness, portable foot restraints

Table VI. Performance & Measurement Factors

PERFORMANCE	MEASUREMENTS
Astronaut only to 60 feet	Distance - 0 - 60 ft and 0 - 200 ft (1)
Multiple translations	Velocity - 1.0 fps, 0.5 fps, 0.5 fps
Proximity to fragile structures	Weight - 0 - 20 lb, 0 - 100 lb, 0 - 300 lb
Bridge Discontinuities	Body Control - fine, gross
Astronaut + cargo - 85 pounds to 25 feet	
Cargo Transfer:	
Rail type: 300 pounds, 50 ft	
Clothesline: 100 pounds, 50 ft	

Table VII. Equipments Required

BASIC ASTRONAUT SUPPORT	TRANSLATION ORIENTED OPERATIONS
Spacesuits: Soft - early High mobility - advanced Life Support Systems: High pressure umbilical Self-contained Safety Tethers: 60' general purpose 10' worksite Restraints: Flexible Rigid Variable	Mobility Aids: Preinstalled handholds and handrails Portable handholds and handrails Harnesses: Special purpose, 100 pound cargo General purpose, rescue Transporters: Inflatable rail Clothesline

with a self-contained life support system and spacesuit it is necessary that the life support be reprovisioned at the end of each task. These suits should conform with maneuvering units even for the basic work functions. It is possible that the maneuvering unit will be used in the early time period to provide means of translation of astronauts and cargo and for inspection type EVA's.

The second category of spacesuit requirements relates to advanced work functions which require significant translations, movement of heavy objects, fine manipulative dexterity, or long-duration tasks. Where the astronaut is required to perform work in an advanced category, it is necessary that high-mobility type suits be developed. Current softsuit designs tend to increase EVA task time and effort over shirt-sleeve operations by a factor of two or greater, depending on suit pressure levels. The astronaut may be required to perform EVA's of up to four hours' duration, and the high-mobility suit will enhance his overall performance capability. The requirements for greater freedom of torso and arm movements to perform cargo removal operations are expressed in terms of mobility capabilities. Increased dexterity will allow the astronaut to perform more contingency-type work operations. This suit, life support system, and maneuvering unit may all be required to be worked simultaneously by the astronaut. Therefore, these units must be compatible. Additionally, the life support system should be separable from both to ease the servicing of the life support system and to allow the astronaut at a work-site to leave a maneuvering unit to perform work functions. As before, the suit should be compatible with umbilical supply provision to reduce expendables required to support EVA tasks. Reduced mass and packaging over the presently developed concepts are required so that the suits may be easily stored within the various spacecraft.

Life Support System

In the development of the life support systems, the systems should be developed separable from both the hard-suit and the maneuvering unit. The separability of the suit allows ease of maintenance of the life support system upon completion of an EVA task. Separability from the maneuvering units allows the astronaut capability to leave the unit at a worksite to perform various work functions. Higher density packaging is required to ease both the problems associated with storing life support systems within spacecraft and to reduce the astronaut difficulties associated with egress and ingress. Reduced packaging may be achieved through the provision of advanced oxygen generation concepts, sophisticated heat-rejection concepts, and advanced packaging design. The unit should further be developed to provide higher metabolic heat rejection capability. The EVA's which have been analyzed require average metabolic heat rejection capability in excess of 2000 Btu's per hour.

Restraint Systems

There are three types of restraint systems which have been considered in the analysis of EVA tasks. These are astronaut restraints, equipment restraints, and cargo harnesses.

Astronaut worksite restraints include foot restraints to waist restraints. The present concepts of Dutch shoes as foot restraints are generally satisfactory for work performance operations. Dutch shoe restraints require that the astronaut insert his feet within the restraint to provide an anchorage at the work-site. Using this type of foot restraint only, the astronaut can apply short-duration forces of 35 pounds. Within limits the astronaut can also perform pitch-, roll-, and yaw-type repositioning movements using this type of foot restraint. However, if this repositioning is done in a pressure suit, a continuous expenditure of energy is required to maintain position. Therefore, for the advanced work performance activities it is required that the foot restraints themselves be developed to have capability for small roll, pitch, and yaw repositioning. This will allow the astronaut to assume an optimal work position with the least expenditure of energy.

Three types of astronaut waist restraints were considered: rigid, flexible, and variable. The variable restraint is that type of restraint which can achieve unusual shapes and through tensioning adjustment be made to retain that shape while the astronaut exerts force. Any of these waist restraint systems must be compatible with the astronaut repositioning requirements at the worksites and must be developed to permit the astronaut to attach and detach his restraint system with minimum expenditure of energy.

Three basic requirements for temporary restraint of equipment at a worksite were found to be required. With a 300-pound item of equipment and consideration of a prepared worksite, fixed mechanical equipment restraints are necessary. The pip-type pin can provide the necessary restraint in this category. This requires two-hand installation where the astronaut maintains position of the equipment with one hand and performs mating operations with the other. The second category of equipment restraints is applicable for cargo weighing up to 100 pounds. This category also applies where the worksite does not have preinstalled provisions for temporary equipment restraints. Latch-type devices which can be secured to available handholds, etc. are applicable in this area. The final category of temporary equipment restraint is that equipment weighing approximately 20 pounds which is to be secured at nonprepared worksites. The adhesive family or velcro-patch type equipment restraints is applicable to this requirement. The adhesive types equipment restraints require development of adhesives which are relatively unaffected by either the high-temperature or low-temperature extremes that could be encountered at a worksite.

Cargo Harnesses

Two types of astronaut-transfer harnesses are required. The first of these harnesses is an astronaut rescue harness which should be capable of moving a 300-pound astronaut by manual translation techniques. This type of harness should provide rather snug trunk restraint which can be rapidly attached by the rescuing astronaut. The second type of cargo transfer harnes should be capable of handling equipment weighing up to 100 pounds. These cargo transfer harnesses will generally be of special-purpose design for the particular modules which are to be transported.

Transfer Aids

In many cases it will be impossible for the astronaut to translate certain types of cargo modules upon an astronaut cargo harness. These modules will either exceed the weight capability of the astronaut or will be extremely unusual in shape. Two types of transfer aids are considered, manual and powered. Manual transfer aids for cargo modules weighing up to 300 pounds are required to have a 5 degree-of-freedom constraint over movement of the cargo. The only degree of freedom for cargo module movement is in a linear direction. Any transfer aids should also be manually deployable, as they may be required to be directed between the logistics resupply vehicles and a space station after docking. The transfer aids should be capable of operating over approximately 50 foot distances. An inflatable two-rail type device with a module latching system was considered. For equipment weighing up to approximately 100 pounds, a limited 4 degree-of-freedom restraint of equipment on a transfer aid is sufficient. These types of transfer aids must also be manually deployable and capable of transferring equipment up to 50 feet. Two classes of device can be considered. The first is a hand-held STEM boom which can quickly translate light equipment items from one astronaut to another. For operational transfer of light equipment, a double-clothesline type device with adjustable tensioning pulleys and stand-off bars is sufficient to meet these cargo-transfer requirements.

Powered Maneuvering Unit Requirements

Maneuvering unit requirements are occasioned by: (a) EVA support requirements of scientific/technical experiments and space station operation in the early 1970 time period, (b) the requirement to support S/T experiments in separable modules (occurring in the mid and later time periods), and (c) by the necessity to have an EVA astronaut rescue/retrieval capability.

Experiment Support

Powered EVA astronaut maneuvering aids were found to be required in direct EVA support of 4 of the 16 scientific/technical experiments investigated in detail and which were projected for operation in the early- to mid-1970 time period. The experiments were those concerned with operation of an earth orbital space station (EOSS), astronomy (X-ray and radio), and with communications and navigation (large parabolic antenna). The basic EVA tasks common to support of these four experiments and which require the aid of powered maneuvering devices all involve relatively long translation distances (from 100 to 300 feet) and the requirement to translate with cargo of small to medium mass (20 to 100 pounds). EVA time at the worksite is of prime importance. Because it is necessary for an EVA astronaut to pace his work output to avoid over expenditure of energy, manual translation over long distances was found to require too much of the astronaut's time in the extravehicular environment leaving too little time at the remote worksite.

This is especially true in long translations involving transfer of cargo - even of small masses. Cargo management tends to increase manual translation time. Finally, the necessity for manual translation aids, i.e. handrails, handholds, either portable or preinstalled, requires that these aids be numerous and installed in unlikely places. Maneuvering devices were found necessary to reduce time and energy expenditure in long translations (over 60 feet), to aid in transfer of small cargo (tools, modules, etc.), and to reduce the necessity for large numbers of preinstalled manual transfer aids.

Separable Module Support

Several of the S/T experiments, particularly those occurring in the mid (and later) 1970 time period involve the use of separate modules flying in formation with the EOSS. These experiments, primarily in the bioscience and astronomy scientific disciplines, require extremely low acceleration environments (10^{-5}g bioscience experiments), or no on-board disturbances such as would be created by human occupants (astronomy).

As initially conceived, these experiments required the use of the Command Service Module (CSM) for experiment deployment and set-up and as a base from which the astronaut would conduct any EVA required in these tasks. In addition the CSM would be used to perform experiment maintenance, refurbishment and updating activities subsequent to deployment and initial operation. This latter experiment support activity would involve occupying the CSM, undocking from the EOSS, translation to the free-flying module, docking to the module, performing the necessary experiment support activity, and then returning and docking the CSM to the EOSS. This procedure is a fairly complex operation and, because of the many precise docking and undocking maneuvers, relatively hazardous.

An alternative technique - that of using a powered translation device - appears the most feasible technically and much safer operationally. The separable modules all have attitude and reaction control systems, and all could be initially separated from the EOSS or logistics vehicle by automatic means without use of the CSM and stabilized in the vicinity of the EOSS. Further, when experiment procedure dictates a refurbishment or up-grading task, the module could be returned to the vicinity of the space station (300 to 500 feet separation). The EVA astronaut could then utilize a powered maneuvering device for translation to the module.

Rescue/Retrieval

An analysis task under the EVEA Program Requirements Study was concerned with rescue and retrieval of an astronaut engaged in EVA. This analysis considered a number of possible EVA operational hazards and revealed that for hazards occasioned by certain equipment failures, response time (to return the astronaut to a point of safety) was too short to be accomplished by manual means, even with the aid of an assisting or rescuing EVA astronaut. Response times varied from 15 minutes to 1.5 hours. At the lowest response times rescue and retrieval will be difficult even with a powered maneuvering device; and the rescue requirement gives rise to certain maneuvering device performance requirements, i.e. rigid docking, stabilization, and the large cargo-carrying capacity (300 pounds, 8 ft^3) represented by a disabled EVA astronaut.

Operational Requirement

The requirement for EVA support of earth orbiting S/T experiments and of an EOSS in both the early and later 1970 time period plus the requirement to rescue and retrieve a disabled EVA astronaut dictate the use of a powered maneuvering device. The specific operational requirements are derived from examination of the specifics of the operational support and rescue functional requirements and are given in Table VIII.

EXPERIMENTAL EVA PROGRAM

The EVEA Study resulted in preparation of two recommended experimental EVA experiment proposals to be performed in earth orbit for the purpose of developing and demonstrating the EVA task functions and associated equipments. The experiments are designed to satisfy the majority of EVA requirements which have been generated for the 1971-1974 time period. One experiment entitled "Transfer, Locomotion and Rescue" deals primarily with translation of astronauts and astronauts plus cargo. The other experiment entitled "Work Performance" is designed to demonstrate both the egress/ingress and work performance functional performance capabilities.

Table VIII. Maneuvering Unit Operational Requirements

PARAMETER	EXPERIMENT SUPPORT		RESCUE	OPERATIONAL REQUIREMENTS
	1971 - 1974	1975 Plus		
Duration	4 hours (not critical)	4 hours (not critical)	1.5 hours	1.5 hours to 4 hours
Range	300 feet	3000 feet	300 feet	3000 feet
Stabilization	5°/min all axes	5°/min all axes	5°/min all axes	5°/min all axes
Cargo Transfer:				
Mass	80 pounds	300 pounds	300 pounds	300 pounds
Volume	3 ft^3	8 ft^3	8 ft^3	8 ft^3
Thruster	Cold gas	Hot gas	Cold gas	Hot/cold gas
LSS	Separable	Separable	Separable	Separable
Maneuvers	Translation local, i.e., station keeping, accel-decel worksite dock, small cargo	Translation local, i.e., station keeping, accel-decel worksite dock, small cargo	Translation local, i.e., station keeping, accel-decel worksite dock, small cargo	Translation local, i.e., station keeping, accel-decel worksite dock, large cargo
Docking	Rigid	Rigid	Rigid	Rigid
Storage	28 days	60 - 90 days	28 - 90 days	28 - 90 days

Using data from the EVA operations analysis the diagram of Figure 6 was structured to demonstrate the applicability of building blocks to describe the tasks investigated during this study. It is of interest to note that of the total 88 building blocks available only 50 were identified as required to support the representative scientific and technical experiments analyzed in this study. Certain specialized characteristic building blocks were eliminated or were found to contain activities redundant with the 50 used. The extensive cost of simulation and orbital experiment programs motivated a next-level analysis to identify the most commonly used building blocks. It was thus determined [illegible]2 building blocks comprised approximately 90 percent of all EVA activity projected for the early 1970's. The EVA orbital experiments were structured to satisfy at least 90 percent of the operational EVA requirements for the 1971-1974 time period.

As has been seen throughout the study, EVA activities basically fall into three categories: egress and ingress functions, translation functions, and work performance functions. During the course of the study, it was determined that the egress and ingress functions were similar in many respects to the work performance functions in that they contain similar tasks which are accomplished within a limited work area. Accordingly, an experimental EVA work performance experiment was established to demonstrate both the egress and ingress functions and the worksite work performance functions.

The EVA function of translation includes translation of cargo as well as translation of astronauts. The largest piece of cargo projected for the early 1970 timeframe was found to have a mass of approximately 300 pounds or roughly equivalent to a disabled EVA astronaut. The retrieval of a disabled astronaut logically leads to demonstration of a rescue capability. Accordingly, a second experimental EVA program to demonstrate translation functions involving both astronauts and cargo was established. This second experiment includes demonstration of a rescue capability and is entitled "Transfer, Locomotion and Rescue." The conduct of these two experiments will satisfy the basic requirement for demonstration of the EVA translation functions, egress and ingress functions, and work performance functions found to be required by analysis of the scientific and technological experiments planned for the early timeframe. These experiments would demonstrate the building blocks identified in the preceding section which satisfy [illegible] percent of the required performance capability.

Both experimental programs involve significant amounts of experimentation and test in ground simulation as well as in orbit experiments. The simulation program will serve to test the experiment procedures to gather data on individual test subjects (some of whom also will be in-orbit test subjects), as training in EVA procedures and to provide a data base for correlation of in-orbit experiment data with simulator data. The latter factor will enable future required EVA techniques and equipment designs (including equipments requiring EVA support) to be developed and tested with a high degree of confidence in simulation programs rather than the more expensive in-orbit test.

Transfer, Locomotion, and Rescue Experiment

The in-orbit transfer, locomotion and rescue requirements can be defined as a series of discrete tasks which must be accomplished to effect the transfer in a given situation. These tasks can be verified by a series of tests structured about the type of locomotion available. Accordingly, the test series is grouped into a set of three separate experiment options or subexperiments: (1) transfer and rescue using a stabilized maneuvering unit (SMU), (2) transfer and rescue under conditions of manual locomotion, and (3) rescue involving the use of various types of personnel/cargo transfer aids. The experimental procedures for investigation of each of the foregoing concepts are identical such that they can stand alone as individual experiment options which can be performed separately on consecutive space flights or ground simulator tests, or they can be performed all in one mission.

For reasons of safety and to obtain controlled experiment conditions, it is proposed that the experiment be conducted in that portion of the orbital workshop (OWS) configuration located just ahead of the crew quarters and known as the experiment area. It is 20 feet in height and slightly more than 20 feet in diameter and will provide the volume necessary to perform the maneuvers and other astronaut translations required in each of the experiment concepts (Figure 7). Ideally, the in-orbit experiment should be performed in an unpressurized environment; however, it is felt that a valid test can be obtained in a pressurized environment provided certain conditions found in the EVA environment are duplicated. For example, the test subjects must use the appropriate spacesuits pressurized to a nominal 3.7 psi above ambient pressure. Probably it will not be possible to obtain a valid test of life support systems in a pressurized environment largely because of difficulties in checking such items as leak rate and water heat exchanger operation. Accordingly, the test subject must be supplied by umbilical from the spacecraft life support systems; and EVA life support systems must be duplicated in the form of a mass and volume mockup worn by the in-orbit test subjects. In the OWS test area the in-orbit experiments will make use of such station habitation items as screen flooring, circumferential handrails, and the fireman's pole which are installed as part of the station activation phase prior to conduct of the experiments. Other than experiment instrumentation and monitoring equipment, spacesuits, and locomotion aids, relatively little equipment will be required to conduct the experiment.

Work Performance Experiment

The proposed work performance experiment is designed to develop and demonstrate the EVA technique and equipment requirements of the egress/ingress and worksite work performance functions. Like the translation experiment, the work performance experiment consists of a series of tests grouped into a set of three separate experiment options each designed to test a discrete series of EVA tasks. The options are: (1) astronaut positioning/repositioning activities, (2) equipment positioning, and (3) worksite performance. The functions of astronaut egress/ingress are accomplished as part of the first test option. The work performance requires the use of a piece of special test equipment, called a work performance analyzer, which includes several different work stations - one specifically suited for each of the experiment options. In addition, the work performance analyzer provides for the test of various forces, dexterity and work performance measurements to test for capabilities found necessary in this study.

Like the transfer, locomotion, and rescue experiment, the work performance experiment options can be performed singly on separate missions or all may be performed in a single mission. One advantage in

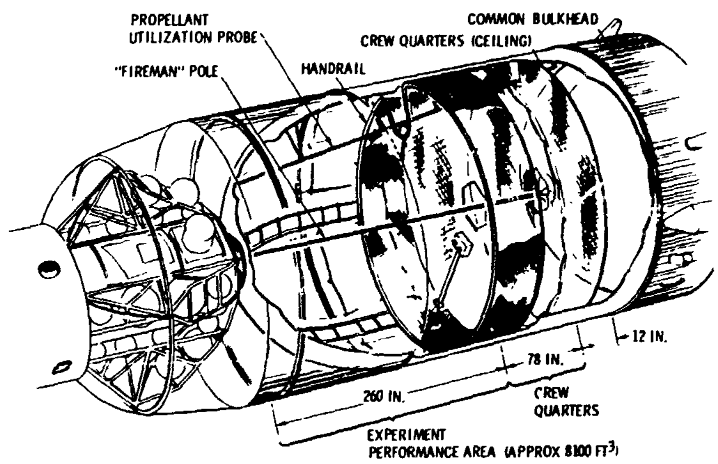

FIG. 7 - ORBITAL WORKSHOP EVA EXPERIMENT AREA - GENERAL CONFIGURATION

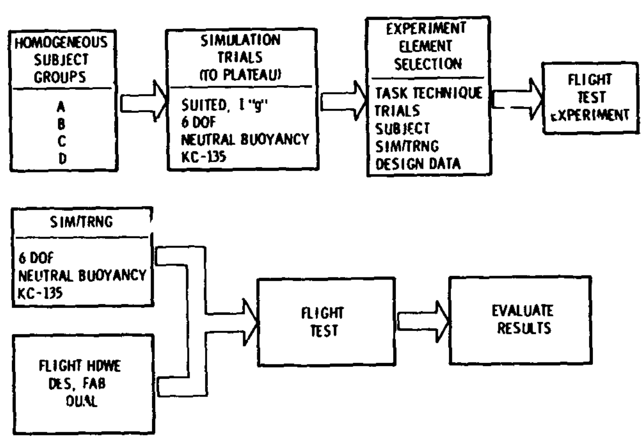

FIG. 8 - EXPERIMENTAL EVA PROGRAM

performing the test options separately (in either experiment) is that baseline data obtained from the first option will serve to verify the test method as well as provide early data correlation with ground simulators. The test procedures of the following test options could be modified as necessary, thereby enhancing the quality of the overall test data. This approach is considered advisable in that it will provide higher confidence in training or evaluations using ground facilities to simulate future EVA concepts.

Simulation Program

The in-orbit experiment program is only part of the overall experimental EVA program necessary to demonstrate the required EVA capabilities and equipment. One of the major purposes of the proposed experimental EVA experiment is to determine the metabolic energy costs of performing the EVA translation tasks. Because of the complexity of some of the tasks and because of differences in test subjects (i.e., size, strength, dexterity, motor response, etc.), experimental test results from a number of different test subjects will be required to obtain meaningful experiment data. The bulk of this work involving a large number of test hours can be most readily accomplished in ground simulators. The in-orbit experiments must be performed to demonstrate and prove the basic EVA translation techniques developed in simulation and to obtain experiment data which can be correlated with ground simulators. Accordingly, the proposed EVA experiments are designed as part of an integrated experiment program utilizing both simulator and in-orbit experimental tests. Figure 8 indicates in diagramatic form the scope of the overall program. A minimum of four groups of test subjects, five subjects to a group, will be required. The subjects are to be homogeneous as between groups; that is, the subjects must be homogeneous in terms of size, weight, strength, and probably motor-response characteristics. Preferably, test subjects who will perform the in-orbit experiments should be included in one or more of these groups. Initially, each member of each group will perform a series of EVA tasks in the pressurized shirtsleeve environment to gain task proficiency. A minimum of five trials per task for each subject is recommended. Subsequently, each of the groups will perform the same set of experiment tasks in one of the ground simulators, performing sufficient trials to reach a performance proficiency plateau in simulation. It will not be necessary at this time for each subject group to perform the simulation trials in each of the different types of simulators. One subject group per simulation type will be sufficient. With data from the simulation program it will be possible to define the in-orbit experiment tasks in detail, estimate the number of trials per task, select experiment subjects, and define the simulation training required for the in-orbit experiment. In addition, experiment equipment design data will be obtained from simulation; and any necessary modifications to equipment as well as experiment procedures can be made prior to performing the in-orbit experiment. With the flight experiment defined in detail and simulation training facilities available, in-orbit experiment training can commence. Subsequent to training, the flight test can take place, followed by evaluation of results from the total program - including simulation test results. After flight tests it may be necessary for some of the test subjects to repeat the experiment in simulation to verify unexpected in-orbit test results.

To perform the experiment as outlined using all four 5-man groups would require approximately 1200 experiment hours. Even if the subject groups were reduced to two and each group contained no more than two men, the time required for the total in-orbit experiment program run to the desired number of trials would be approximately 480 elapsed experiment hours. An in-orbit experiment program requiring 480 test hours would occupy the better part of a 28-day manned orbital mission. Further, it is doubtful that four men will be available for in-orbital experiment, at least in the timeframe of interest (1971-1974). It becomes necessary to reduce the number of subject groups to one, use just two test subjects, and carefully select the in-orbit experiment trials. By so doing, reducing the number of in-orbit test trials to no more than ten, and carefully selecting the astronaut test subjects (to represent the fifth through ninety-fifth anthropometric range) it is possible to reduce the in-orbit experiment to 70 elapsed test hours. It should be recognized that these 70 hours represent total test times. For many of the tests two astronaut test subjects will be working concurrently with data being taken on each. Therefore, the 70 hour total actually represents approximately 40 hours of elapsed mission time.

Once a valid correlation has been established between the in-orbit experiment operations and simulator experiment operation, much of the future EVA techniques development can be accomplished in simulators without need for verification of in-orbit experiments. The EVA experimental program will require the use of several different types of simulators; i.e., neutral buoyancy, six degree-of-freedom (6 DOF), and KC-135. However, performance of the transfer, locomotion, and rescue experiment required long experiment time continuity as well as the capability for distance-translation continuity. Accordingly, it is anticipated that the neutral buoyancy simulator will become the prime simulation device for this type of EVA activity.

RESULTS

In the approximately 1200 orbital experiments that were considered it was found that more than half had firm requirements for EVA support. A detailed systems analysis of a representative group of experiments occurring in the period of principal interest (1971-1974) revealed the need for a clear and definite understanding of EVA capability. This analysis highlighted much required EVA equipment development. Spacesuits utilized by the astronauts must be made much more mobile, and dexterity capability must be increased. As the EVA astronaut is required to perform more tasks in support of orbital experiments, he will be required to spend more time in EVA. Astronaut endurance is a problem that must be solved to maximize the useful astronaut work time. Development of suits should provide these capabilities, while reducing the weight and storage volume requirements.

Life support system weight and size must be reduced through better packaging techniques and use of advanced technologies. System capacity should be somewhat increased for support of longer EVA's. Cargo transfer aids are needed to allow the astronaut to transfer large equipment modules from a storage point to a worksite. Various design concepts are needed to fulfill the requirements of the different types of loads to be transported. Powered translation aids should be developed to increase the effective range of the astronaut. Manual translation is not practical in activities that require inspection of large vehicles. Astronaut worksite restraint systems need modification to minimize astronaut energy expended in attaching and detaching the restraint system and in repositioning at a given worksite. An assortment of general astronaut tools should be developed to provide the astronaut the capability to perform contingency modular maintenance.

Experimental data will be necessary in the development of the required EVA equipment and in the verification of astronaut techniques to effectively utilize the equipment. Two proposed experimental EVA programs therefore are proposed to obtain these data. The flight portions of the proposed experiments will be held to the minimum. Flight data will be collected to demonstrate EVA techniques and to provide a basis for correlation of an intensive ground simulation program. It is anticipated that the flight tests for both experiments can be conducted in a total of 170 hours.

CONCLUSIONS

This study has shown that the use of man in an extravehicular mode of operation has a potential to support scientific and technical orbital experiments.

Development of the necessary EVA equipment and techniques could make EVA a competitive operation for support of orbital experiments in the following areas: (1) increased life through maintenance and repair, (2) increased utilization through modular replacement, and (3) increased capability for data retrieval.

Training and simulation programs also must be conducted in preparation for EVA to gain experience and obtain design data.

Conduct of both proposed experimental EVA programs will result in the verification of approximately 94% of the capability required for support of early orbital experiments.

ADDITIONAL PAPERS

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EXTRAVEHICULAR MAINTENANCE: APPLICATIONS, TECHNIQUES, TASKS AND TOOLS

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SUMMARY: This paper identifies those spacecraft subsystems to which EVM is applicable; applicable EVM techniques; associated EVM tasks and task times; and associated tools. The results reported are presented in tabular form and are directly applicable to a broad range of future studies on the application of EVM to specific subsystems, vehicles or missions. The supporting data (EVM time, tools and test equipment weight, power etc.) provides a basis for making these future studies.

INTRODUCTION

The purpose of the study reported here was to identify those spacecraft subsystems to which extravehicular maintenance (EVM) is applicable, the techniques associated with EVM, the tasks associated with EVM, and the tools required to perform EVM. The approach utilized in performing the study consisted of three interrelated substudies:

1. EVM Applications/Techniques

2. EVM Techniques/Tasks

3. EVM Tasks/Tools

A generalized approach was followed throughout each of these studies in order to obtain results which are directly applicable to a broad range of future studies in the application of EVM to specific subsystems, vehicles or missions.

FUNCTIONAL AND DESIGN REQUIREMENTS

Specific functional and design requirements for EVM must be based on the reliability requirements for specific spacecraft subsystems. These requirements are in turn based on mission success and crew safety considerations. It is beyond the scope of the present study to determine specific spacecraft reliability requirements and, therefore, there are no specific functional and design requirements applicable to this tradeoff study. It has been assumed that all spacecraft subsystems are potential candidates for the application of EVM.

ASSUMPTIONS AND TRADESTUDY GROUND RULES

Applications of EVM. A generalized spacecraft was assumed. In the 1968-1972 time period it appears that such a spacecraft will employ Apollo/LM/MOL state-of-the-art subsystems or minimum modifications thereof.

Maintenance Techniques. Seven general maintenance techniques were assumed to be applicable. A complete discussion of these techniques is beyond the scope of this paper but equipment, time and logistics aspects of all of these techniques are discussed in depth in Reference 1 and the advantages and disadvantages of the various techniques are covered in References 1 and 2.

In addition to access and time, considerations of weight, cost and initial nonredundant system reliability are also important factors in recommending a specific maintenance technique for a specific subsystem. A comparison of the reliability improvement resulting from maintenance (enclosed by box) or nonmaintenance techniques, as a function of initial reliability, is shown in Figure 1 (from Reference 3). It should also be noted that no one redundancy or maintenance technique is 100% applicable to any given subsystem, as illustrated by Figure 2 (from Reference 4). It is beyond the scope of this study to investigate maintenance level and test system tradeoffs for specific subsystems and the reader is referred to Reference 2 for a detailed discussion of these factors.

While it is recognized that EVM will require a specific design for EVM (see Reference 4 for specific design considerations), it has been assumed that EVM tasks will parallel maintenance tasks currently associated with ground and airborne systems, but require additional time and energy expenditures due to pressure suit constraints and zero-g conditions.

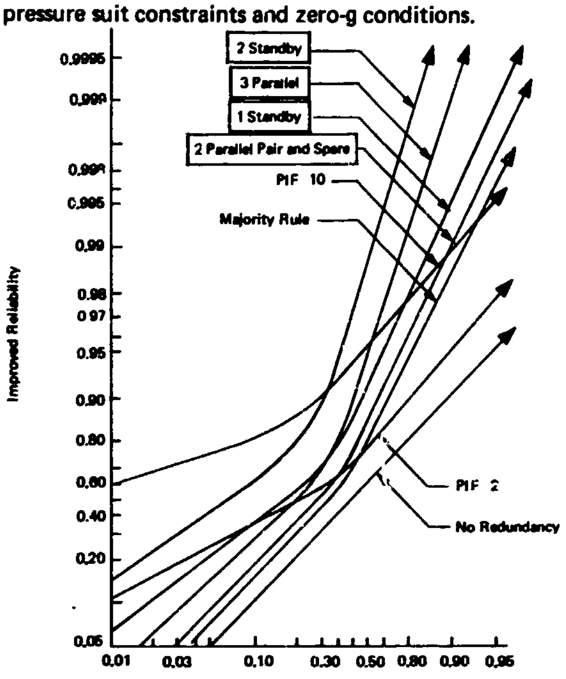

Figure 1. (U) Mission Success Probability as a Function of Various Redundancy Techniques or Parts Improvement Factors

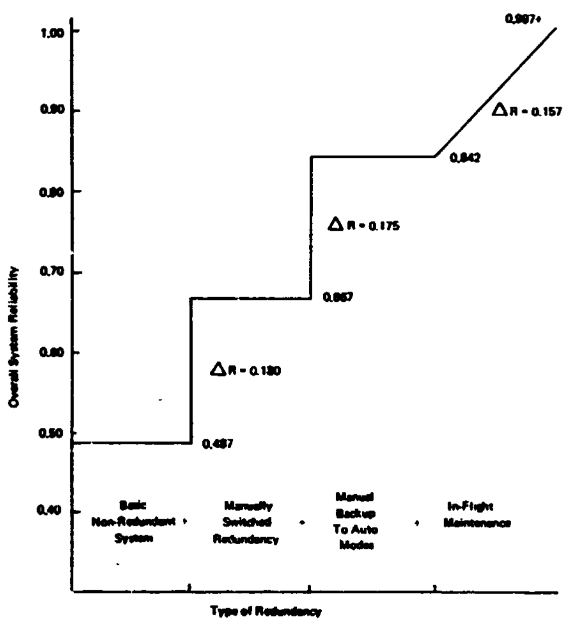

Figure 2. Contribution of Various Types of Redundancy to System Reliability

Maintenance Tools. In the area of maintenance tools it appears that modified conventional tools (References 5 and 6) or special tools now being developed (see References 7, 8 and 9) by the USAF and NASA will satisfy all maintenance tool requirements in the time period under consideration. However, recent Gemini experience (Reference 10) indicates that extensive investigation of maintenance restraint systems will be required before EVM can become a reality. Some preliminary investigations of these problems have been accomplished by Bell Aerosystems and others and the reader is referred to References 11 and 12 for further discussion. It was assumed that space qualified maintenance tools and restraint systems will be available in the 1968-1972 time period.

IDENTIFICATION OF POSSIBLE DESIGN ALTERNATIVES

Maintenance Techniques. Seven general maintenance techniques were considered:

1. Automatic monitoring and switching of built-in-redundancy (BIR). This technique is most applicable to inaccessible, time critical maintenance actions.

2. Automatic monitoring and manual switching of built-in- redundancy. This technique is most applicable to maintenance actions which have severe accessibility constraints.

3. Manual monitoring (using operational displays and controls) and manual switching of built-in-redundancy. This technique includes interchanging connectors between black boxes or manually operating valves, etc., as well as actuating control panel switches. It is most applicable to non-time critical maintenance actions where limited maintenance access and limited maintenance time constraints (i.e., repair times of less than five minutes) exist.

4. Modular replacement of subsystem components. This technique is most applicable to maintenance actions where there are no maintenance access limitations and where repair times of one or two hours are permissible.

5. Bits and pieces repair. This technique is most applicable to maintenance actions

where there are no maintenance access limitations and where repair time of many hours are permissible.

6. Component adjustment. This technique is most applicable to analog subsystems subject to drift or subsystems employing fluids or gases where optimum performance can be obtained by proper adjustment of flow rates. Limited access is required and repair times of greater than 5 minutes are anticipated.

7. No maintenance. In any spacecraft design, situations will occur which preclude consideration of any of the maintenance techniques considered above. In this case, parts improvement programs, active parallel redundancy (i.e., load sharing), or other techniques are the only feasible means of achieving specified reliability requirements.

Maintenance Tasks. As previously noted, it has been assumed that EVM tasks will parallel those currently being experienced with airborne systems under one-g conditions. Based on this assumption, Procedure I of MIL-HDBK-472 (Reference 13) has been selected as a means of specifying space maintenance tasks. Using this approach, all possible EVM tasks are assumed to be described by one or more of the 53 "elemental maintenance activities" incorporated in six categories:

1. Preparation
2. Malfunction Verification
3. Fault Location
4. Part Procurement
5. Repair
6. Final Test

Elemental maintenance activities and associated mean activity times for both one-g shirtsleeve and zero-g pressure suit conditions are shown in Table 3. Activity descriptions and one-g data were taken from Reference 14 except as noted. A literature search was conducted to obtain the necessary factors to determine zero-g, pressure suit maintenance times. Zero-g conditions have been shown to cause a 20 to 50% degradation in performance (References 6 and 15) while the pressure suit causes an additional 50 to 132% degradation (References 6, 15, 16 and 17). An average factor of 2.8 was used in this study to transform from one-g, shirtsleeve to zero-g, pressure suit conditions; i.e., zero-g pressure suit time is equal to 2.8 times one-g, shirtsleeve time. It should be noted that this factor of 2.8 may be somewhat conservative with respect to the time period under consideration since increased suit mobility and improved cooling are anticipated with the introduction of "hard suits" and water-cooled thermal systems.

Maintenance Tools. The maintenance tools considered in this study fall into four general categories:

1. Standard test equipment including:
   a. Stimulus generators
   b. dc power supplies
   c. ac power supplies
   d. Measurement equipment
2. Special test equipment specifically designed for a given subsystem and integrating the functions (as required) outlined under standard test equipment.
3. Standard hand tools
4. Special tools including:
   a. Martin/Black & Decker minimum-reaction tool
   b. Electron beam welder
   c. Wire wrap tool, etc.

A complete breakdown of the tools considered is given in Table 5.

Maintenance Applications. As previously noted, all systems and subsystems in both manned and unmanned spacecraft have been assumed to be potential candidates for the application of one or more of the seven maintenance techniques considered. It has been assumed that the parent spacecraft will be manned but that the remote vehicle can be manned or unmanned. If the remote vehicle is manned, it is further assumed that the same maintenance techniques, tasks and tools applicable to EVM of the parent vehicle are also applicable to the remote manned vehicle. The unmanned remote vehicle represents a special case since the design criteria for unmanned vehicles are significantly different in many areas than those employed for manned spacecraft. In particular, maintenance access may be severely constrained or nonexistent. The type of repair actions anticipated for unmanned vehicles can be hypothesized from a review of unmanned spacecraft failures. The results of one such review are given in Table 1 (from Reference 18).

DISCUSSION

Maintenance Applications/Techniques Tradestudy. A generalized spacecraft was developed based on Apollo/LM/MOL technology. This generalized spacecraft consists of 16 major subsystems and each of these subsystems was further subdivided into its major components as shown in Table 2.

For each subsystem an investigation was undertaken to determine: (1) the applicability of EVM to a given subsystem component in both the parent and remote spacecraft and (2) the applicability of one or more of the previously defined maintenance techniques to EVM of a given subsystem component. Four criteria and two techniques were used in making these determinations. The criteria were:

1. Accessibility considerations; i.e., within the current and projected state-of-the-art would the component under consideration be accessible from either the inside

TABLE 1
SUMMARY OF UNMANNED SPACECRAFT MALFUNCTIONS

	SATELLITE	LAUNCH DATE	DECAY DATE	MALFUNCTION	NOTES
1.	Midas 2	5-2-60	A.	Data Link Quit Second Day	A. Still in orbit as of 12-31-65
2.	Discoverer 20	2-17-61	7-28-62	Programmer Failure, No Capsule Ejection	B. In-Jun functioned until 3-6-63, Solrad until late 1961.
3.	Transit 3B/Lofti 1	2-21-61	3-30-61	Second Stage, Satellites Failed to Separate	C. Impacted on moon
4.	In-Jun 1/Solrad 3	6-29-61	A.B.	Failed to Separate	
5.	Discoverer 31	9-17-61	10-26-61	Capsule Separation Failed	
6.	Midas 4	10-21-61	A.	Dipoles Failed to Disperse	
7.	Discoverer 34	11-5-61	12-7-62	Malfunction Prevented Capsule Ejection	
8.	Traac 11-15-61	11-15-61	A.	Gravity Gradient Experiment Boom Failed to Extend	
9.	Transit 5A	12-18-62	A.	Power Failure First Day	
10.	Syncom 1	2-14-63	A.	Communication Lost at Orbital Injection	
11.	Geophysical Research Satellite	6-28-63	A.	Space Gas Experiment Ceased After 13 Orbits	
12.	Ranger 6	1-30-64	C.	T.V. System Malfunctioned	
13.	Tiros 9	1-22-65	A.	One Camera Operational	
14.	Secor 2	3-11-65	A.	Failed to Operate as Planned	
15.	Secor 4	4-3-65	A.	Failed to Operate as Planned	
16.	Surcal (65B)	8-13-65	A.	Failed to: Separate from Second Stage; Deploy 200 ft. Antenna	

(pressurized or unpressurized sections) or outside of the spacecraft; particularly with respect to modular replacement, bits and pieces repair or component adjustment.

2. Repair time considerations; i.e., is the nature of the component under consideration such that immediate maintenance is required or can extended repair times be tolerated or delayed maintenance be performed.

3. Component state-of-the-art considerations; i.e., does the current or projected state-of-the-art for the component permit modular replacement, component adjustment or bits and pieces repair or is a standby redundancy approach feasible when the monitoring and switching requirements are considered.

4. Component reliability considerations; i.e., will the initial reliability of the component under consideration be such that one maintenance technique will permit achievement of anticipated reliability goals at negligible cost and weight penalties as opposed to large penalties for other techniques.

The techniques employed were:

1. Reference to existing literature on previous studies conducted by Bell and other aerospace contractors (References 2, 3, 4, 5, 15, and 18 - 29) for recommended maintenance techniques.

2. Review of a draft of Table 2 by a panel of three senior Bell aerospace systems engineers with extensive Gemini, Apollo, LM, Apollo X, AAP and MOL spacecraft design and operational experience to determine the "face" validity of the original determinations.

Maintenance Techniques/Tasks Tradestudy. The technique employed in this substudy was a straightforward comparison of a generalized set of maintenance tasks against the previously defined techniques and then the identification of those tasks associated with a given technique. Mean task identify those tasks associated with a given technique. Mean task activity times for both one-g shirtsleeve and zero-g pressure suit conditions are also presented, so that preliminary time estimates for EVM of any subsystem component/maintenance technique combination (derived from Table 2) can be generated by simple summation of applicable elemental task times. (Note: In addition, suit don/doff and maintenance restraint system attachment/detachment times must be added to this sum to obtain total EVM block time.) The results of this substudy are given in Table 3.

Maintenance Tasks/Tools Tradestudy. A generalized set of maintenance tasks, such as tightening, cutting, drilling, etc., was developed from a review of the elemental activities listed in Table 3. A list of tools required to perform these tasks was developed from a review of the literature (References 1, 5, 7, 8, 9, 15, 19, 20, 21, 26, 30 and 31). These two lists were used to form the tradestudy matrix, Table 4, which identifies related tools and tasks. The characteristics of these tools are given in Table 5, and the use-frequency data given are from Reference 33.

Figure 3 shows in-flight test system weight as a function of the number of test/monitor points required. A complete description of these systems will be found in References 2, 15 and 32. The data from Figure 3 were used to obtain the weights given in Table 5 for test systems.

RESULTS AND CONCLUSIONS

Table 6 presents a summary of the results of the three sub-tradestudies. It identifies the subsystems and system components to which EVA maintenance is applicable and the associated applicable maintenance techniques, tasks and tools. Except for the launch escape system, EVA maintenance is applicable to (at least some components within) all major spacecraft subsystems.

As previously noted, no one maintenance technique is 100% applicable to a single subsystem, let alone all subsystems. This fact is verified by the "Applicable Maintenance Techniques" columns in Table 6. With few exceptions, at least two and in most cases three or four of the six techniques considered are applicable to all subsystems and subsystem components. Where applicable maintenance techniques are limited to automatic monitoring (with either automatic or manual switching), maintenance tasks are limited to preparation. Where switching of built-in-redundancy (BIR) is employed, part procurement and repair tasks are eliminated. The only techniques which require consideration of all six maintenance task categories are modular replacement and bits and pieces repair. Even

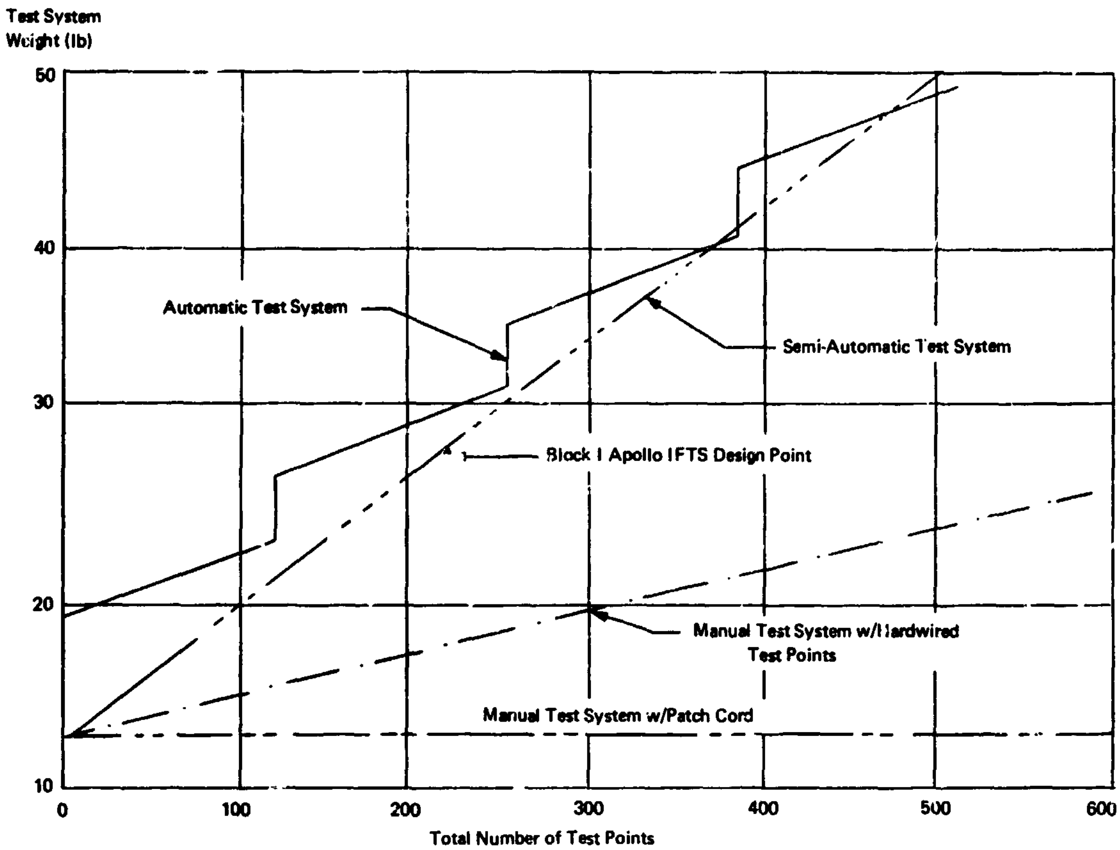

Figure 3. In-Flight Test System Weight as a Function of Required Number of Test Points

here, there are concepts currently being studied (Reference 34) which would eliminate the parts procurement task category as a requirement under the modular replacement technique.

It appears that standard test equipment and hand tools (suitably modified for operation under space and pressure suit conditions) will find wide application in support of EVA maintenance. Special tools such as the "Space Power Tool" significantly enhance the accomplishment of EVA maintenance by increasing the astronaut's ability to apply forces under zero-g and pressure suit conditions. A recommendation for power tools as opposed to hand tools depends on consideration of all possible applications within the framework of a specific mission and set of mission equipments and thus cannot be made at this time. The development of special test equipment should be approached with caution, since the principal justification for such equipment lies in the weight and fault isolation time reductions associated with integrating and/or automating the functions performed by standard test equipment. Unless specific weight (or maintenance time) constraints can be identified, the possible weight reductions are directly dependent on the extent, frequency and level of maintenance predicted for all subsystems. Thus, an extensive tradestudy is required to establish the requirement of a special test set for each specific program within the framework of a fairly well defined set of mission profiles and mission equipment.

ACKNOWLEDGEMENT

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TABLE 2

MAINTENANCE APPLICATIONS/TECHNIQUES TRADESTUDY MATRIX

Major Subsystem	Type Maintenance				Applicable Maintenance Techniques							Remarks
	Parent S/C		Remote S/C		Auto. Mon. • Sw. of BIR	Auto. Mon. Mnl. Sw. of BIR	Mnl. Mon., Mnl. Sw., of BIR	Mod Replacement	Bits and Pieces Repair	Component Adjustment	No-Maint., PIF or Others	
	IVM	EVM	IVM	EVM								
1 STRUCTURE												
External												
Primary		X		X					X		X	
Secondary		X		X					X		X	
Internal												
Primary	X		X	X					X		X	
Secondary	X		X	X					X		X	
2 LAUNCH ESCAPE	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	
3 LANDING AND RECOVERY												
Parachute or paraglider		X			X	X	X				X	
Deployment		X					X		X		X	Repair of Jammed Mech. or Meteorite Damage If Accessible
Control	X						X	X			X	
Impact attenuation		X									X	
Flotation bags		X									X	
Heat Shield		X									X	Can Possibly be Inspected by EVA
Beacon and communications											X	
RCVR/XMTR	X	X					X	X			X	
Antenna		X										
4 CREW SYSTEMS												
Space suits	X						X		X		X	Use Space Suit or Patch Major Damage Areas Failure Would Only Result in Inconvenience or Crew Discomfort
Personal equipment	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	
Waste management	X	X					X	X	X	X	X	
Medical and first aid	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	
Food preparation equipment	X							X			X	
Food supplies	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	
Personal hygiene	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	Failure Would Only Result in Crew Inconvenience or Discomfort
Couches	X							X	X	X		
Restraints and tethers	X						X	X	X	X	X	
Rescue and survival equip	X							X			X	
Portable, emergency life support	X	X	X	X	X	X	X	X	X	X	X	
Lighting	X	X	X	X	X	X	X	X	X	X	X	
5 ENVIRONMENT CONTROL AND LIFE SUPPORT												
Oxygen (including storage)	X	X	X	X	X	X	X	X	X	X	X	Time Critical
Diluent gas (including storage)	X	X	X	X	X	X	X	X	X	X	X	Time Critical
Humidity	X	X	X	X		X	X	X		X	X	Principal Adjustment Would Involve
Temperature	X	X	X	X		X	X	X	X	X	X	Control of Gas or Fluid Flow Rate
Cabin pressure	X	X	X	X	X	X		X		X	X	Time Critical
Contamination control and Removal	X	X	X	X	X	X	X	X		X	X	
Equipment cooling	X	X	X	X	X	X	X	X	X	X	X	
Displays and controls	X		X				X	X			X	

TABLE 2 (CONT)

Major Subsystem	Type Maintenance				Applicable Maintenance Techniques							Remarks
	Parent S/C		Remote S/C		Auto. Mon. and Sw. of BIR	Auto. Mon. Mnl. Sw., of BIR	Mnl. Mon., Mnl. Sw., of BIR	Mod Replacement	Bits and Pieces Repair	Component Adjustment	No-Maint., PIF or Others	
	IVM	EVM	IVM	EVM								
6 ELECTRICAL POWER												
Primary source(s) (fuel cell)	x	x	x	x	x	x	x			x	x	
Secondary source(s) (battery)	x	x	x	x	x	x	x	x		x	x	
Cryogenic storage		x		x		x	x			x	x	
Inverter(s)	x	x	x	x	x	x				x	x	
Power distribution (ac and dc busses)	x	x	x	x	x	x	x				x	
Displays and controls	x		x				x	x			x	
Solar panels and orientation mech.	x	x	x	x			x	x		x		
7 STABILIZATION AND CONTROL												
Sensors (gyros, horizon scanner, etc.)	x	x	x	x	x	x	x	x		x	x	
Control electronics	x		x	x		x	x	x		x	x	
Displays and controls	x		x				x	x			x	
Control moment gyros	x	x	x	x				x		x	x	
Inertial wheels	x	x	x	x				x		x	x	
8 SECONDARY PROPULSION												
Engines		x		x	x	x	x	x			x	
Propellant storage and distribution		x		x	x	x	x				x	
Pressurant storage and distribution				x	x	x	x		x		x	
Displays and controls	x		x				x	x			x	
9 COMMUNICATIONS AND DATA MANAGEMENT												
Antenna(s)		x		x	x	x	x	x		x	x	
Antenna orientation and control	x	x	x	x	x	x	x	x	x	x	x	
Antenna switching	x	x	x	x				x		x	x	
Transmitter (s)	x	x	x	x	x	x	x	x		x	x	
Receiver(s)	x	x	x	x	x	x	x	x		x	x	
Intercom	x		x				x	x	x	x	x	
Central timing	x		x		x	x				x	x	
Data mgmt. computer	x		x		x	x		x			x	
T.V. camera	x	x	x	x	x	x	x	x		x	x	
Tape recorder	x		x		x	x	x	x		x	x	
Signal conditioning	x	x	x	x				x		x	x	
Multiplexer	x	x	x	x	x	x		x			x	
Displays and controls	x		x				x	x			x	
10 GUIDANCE AND NAVIGATION												
IMU or TARS	x		x	x	x	x		x		x	x	Mod. Rep. of Suppo't Electronics or Gyro's; Possible Drift Trim Adjustment
Digital computer	x		x	x	x	x	x	x			x	Switching of "Scratch Pad" Memory Presents Near Impossible Problem
Optics (sextant and scanning telescope)	x	x	x	x				x	x	x	x	Optics Repair Demonstrated on Gemini
Displays and controls	x		x				x	x			x	

TABLE 2 (CONT)

Major Subsystem	Type Maintenance				Applicable Maintenance Techniques							Remarks
	Parent S/C		Remote S/C		Auto. Mon. and Sw. of BIR	Auto. Mon. Mnl. Sw. of BIR	Mnl. Mon., Mnl. Sw., of BIR	Mod Replacement	Rlw and Piece Repair	Component Adjustment	No-Maint., PIP or Others	
	IVM	EVM	IVM	EVM								
11 PRIMARY PROPULSION												
Engine(s)		x		x	x			x	x	x	x	Time Critical, Repair Only During Non-
Gimbal and control system (TVC)	x	x	x	x	x			x		x	x	Thrusting Periods
Pressurant storage and distribution		x		x	x	x	x		x	x	x	
Propellant storage and distribution		x		x	x	x	x			x	x	
Displays and controls	x		x				x	x			x	
12 INSTRUMENTATION												
Malfunction detection and warning	x	x	x	x	N/A	N/A	N/A	N/A	N/A	N/A	N/A	Low Duty Cycle and Non-Criticality W/R to Mission Success (In Most Cases) and Crew Safety Precludes Consideration of Redundancy in Maintenance
Bio-medical data	x				N/A	N/A	N/A	N/A	N/A	N/A	N/A	
Engineering data	x	x	x	x	N/A	N/A	N/A	N/A	N/A	N/A	N/A	
Experimental data	x	x	x	x	N/A	N/A	N/A	N/A	N/A	N/A	N/A	
Special and misc. data	x	x	x	x	N/A	N/A	N/A	N/A	N/A	N/A	N/A	
In-flight test system	x		x					x			x	
13 EXPERIMENTATION												
Photographic equipment	x	x	x	x				x	x	x	x	
Special sensors	x	x	x	x	TBD	TBD	TBD	TBD	TBD	TBD	x	
Manipulator and control system	x	x		x				x	x	x	x	
Special test chambers	x	x	x	x	TBD	TBD	TBD	TBD	TBD	TBD	x	
Others	x	x	x	x	TBD	TBD	TBD	TBD	TBD	TBD	x	
Displays and controls	x		x				x	x			x	
14 AUTOMATIC SEQUENCING												
Launch escape sequence	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	Due to g-Loads During Launch and Boost
Boost abort sequence	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	No Maint. Is Possible
Staging and separation sequence	x	x	x	x	x	x	x				x	Time Critical
Earth landing sequence	x	x	x	x	x	x	x				x	
Special device deployment sequence	x	x	x	x	x	x	x	x	x	x	x	
Others	x	x	x	x	TBD	TBD	TBD	TBD	TBD	TBD	TBD	
Displays and controls	x		x				x	x			x	
15 RENDEZVOUS AND DOCKING												
Docking mechanism	x	x		x				x	x	x	x	
Rendezvous radar and antenna		x		x		x	x	x		x	x	
Special displays and controls	x		x					x	x	x	x	
Inter-vehicle tethers		x		x				x	x		x	
16 MISCELLANEOUS												
Stage/space craft separation devices		x		x	x						x	
Sensor/window covers, fairings		x		x					x		x	Repair Consists of Clearing Jammed or Forced Mechanisms or Manual Deployment or Separation
Deployment mechanisms		x		x					x		x	

TABLE 3
MAINTENANCE TECHNIQUES/TASKS TRADESTUDY MATRIX

Task Category	Elemental Activity	Activity No.	Mean Activity Time (Hrs) One-g Shirt-Sleeve	Mean Activity Time (Hrs) Zero-g Pressure Suit	Applicable Maintenance Technique: Auto. Man. and Sw. of BIR	Auto Man. Mnl. Sw. of BIR	Mnl. Man. Mnl. Sw. of BIR	Modular Replacement	Bits and Pieces Repair	Component Adjustment	No. Maint PIP or Other	Remarks
Preparation (A)	Sys. turn-on, warm-up, setting dials, etc.	A1	0.102	0.286	X	X	X	X	X	X	N/A	
	A1 + waiting for component stabilization	A2	0.666	1.863	X	X	X	X	X	X	N/A	
	Open and/or closing hatch	A3	0.236	0.660	N/A	X	X	X	X	X	N/A	Applies to manl. sw. only if sw. is accompanied by valve X'Fer or connector interchange
	Removing and replacing access cover	A4	0.276	0.773	N/A	X	X	X	X	X	N/A	
	Obtaining test equip. and/or procedures	A5*	0.226	0.633	N/A	N/A	X	X	X	X	N/A	
	Checking maintenance data	A6	0.140	0.392	N/A	X	X	X	X	X	N/A	
	Conference w/ground or other crew members	A7	0.046	0.126	N/A	X	X	X	X	X	N/A	
	Obtain components in anticipation of need	A8	0.070	0.196	N/A	N/A	N/A	X	N/A	N/A	N/A	
	Set up test equipment	A9*	0.100	0.280	N/A	N/A	X	X	X	X	N/A	
	Stabilize or orient spacecraft	A10	0.050	0.140	N/A	N/A	X	X	X	X	N/A	
Malfunction Verification (B)	Observation of normal displays	B1	0.105	0.106	N/A	N/A	X	X	X	X	N/A	
	Perform standard tests	B3	0.397	1.112	N/A	N/A	X	X	X	X	N/A	
	Test for pressure leak	B4	0.329	0.921	N/A	N/A						
	Attempt to observe elusive or nonexistent symptoms	B5	1.394	3.903	N/A	N/A	N/A	N/A	N/A	N/A	N/A	Assume unambiguous isolation and verification
	Use standard test equipment	B6*	0.167	0.468	N/A	N/A	X	X	X	X	N/A	
	Use special test equipment designed specifically for given system	B7*	0.250	0.700	N/A	N/A	X	X	X	X	N/A	
	Make visual integrity check	B8*	0.139	0.389	N/A	N/A	X	X	X	X	N/A	Assumed to require locomotion to obtain visual axis
Fault Location (C)	Fault self-evident from symptom observatory	C1	0.010	0.010	N/A	N/A	X	X	N/A	X	N/A	
	Interpret symptoms by mental analysis only (from knowledge/experience)	C2	0.019	1.019	N/A	N/A	X	X	N/A	X	N/A	
	Interpret displays at different control settings	C3	0.333	0.932	N/A	N/A	X	X	X	X	N/A	
	Interpret meter readings	C4	0.141	0.141	N/A	N/A	X	X	X	X	N/A	
	Remove unit(s)/subunit(s) and bench check	C5	0.821	2.299	N/A	N/A	N/A	[illegible]	X	N/A	N/A	
	Switching and/or substituting unit(s)/subunit(s)	C6	0.324	0.907	N/A	N/A	X	X	N/A	N/A	N/A	Applicable to auto. man. techniques only if maint. capability exists in addition to BIR
	Switching and/or substituting part(s)	C7	0.436	1.221	N/A	N/A	N/A	N/A	X	N/A	N/A	
	Removing and checking part(s)	C8	0.181	0.507	N/A	N/A	N/A	N/A	X	N/A	N/A	
	Make visual integrity check	C9	0.139	0.389	N/A	N/A	X	X	X	X	N/A	
	Check voltages, wave forms and/or signal tracing	C10	0.807	2.260	N/A	N/A	X	X	X	X	N/A	
	Consult Tech Mnls.	C11*	0.044	0.123	N/A	N/A	X	X	X	X	N/A	

* Not in Original MIL-HDBK 472 Data

TABLE 3 (CONT)

Task Category	Elemental Activity	Activity No.	Mean Activity Time (Hrs) One-g Shirt-Sleeve	Mean Activity Time (Hrs) Zero-g Pressure Suit	Applicable Maintenance Technique: Auto. Man. and Sw. of BIR	Auto. Man. Mnl. Sw. of BIR	Mn Man. Mn Sw. of BIR	Modular Replacement	Bits and Pieces Repair	Component Adjustment	No. Maint. PIF or Other	Remarks
Fault Location (C)	Confer w/ground or other crew members	C12	0.466	1.306	N/A	N/A	X	X	X	X	N/A	
	Perform standard tests	C13	0.582	1.630	N/A	N/A	X	X	X	N/A	N/A	
	Isolate pressure leak	C14	0.683	1.912	N/A	N/A	N/A	N/A	X	N/A	N/A	
	Decide whether maintenance is necessary	C15	0.015	0.042	N/A	N/A	N/A	X	N/A	N/A	N/A	Applicable to auto. Man. techniques only if maint. capability exists in addition to BIR
	Use special test equipment designed specially for given system	C16	0.524	1.467	N/A	N/A	X	X	X	N/A	N/A	
Part Procurement (D)	Obtain replacement from in-place spares	D1	0.022	0.062	N/A	N/A	N/A	X	X	N/A	N/A	
	Obtain replacement from on-board stock	D2	0.102	0.286	N/A	N/A	N/A	X	X	N/A	N/A	
	Obtain replacement from remote stock	D3	0.316	0.885	N/A	N/A	N/A	X	X	N/A	N/A	
	Obtain replacement from cannibalization	D4	0.315	0.882	N/A	N/A	N/A	X	X	N/A	N/A	
	Attempt to obtain replacement; none available	D6	0.199	0.557	N/A	N/A	N/A	N/A	N/A	N/A	N/A	
Repair (E)	Remove and replace unit(s)/subunit(s)	E1	0.394	1.103	N/A	N/A	N/A	X	X	X	N/A	
	Remove and replace part(s)	E2	0.380	1.064	N/A	N/A	N/A	N/A	X	N/A	N/A	
	Correct improper installation or defective connection	E3	0.066	0.185	N/A	N/A	N/A	X	X	N/A	N/A	
	Make in-place adjustment	E4	0.415	1.162	N/A	N/A	N/A	N/A	X	X	N/A	
	Make on-bench adjustment	E5	0.993	2.780	N/A	N/A	N/A	N/A	X	X	N/A	
	Wait time for curing, settings, etc	E6	1.416	1.416	N/A	N/A	N/A	N/A	X	N/A	N/A	May be done in zero g shirtsleeve
	Precautionary repair (includes so-called fault isolation, part procurement and repair time when symptom not verified)	E7	0.752	2.106	N/A	N/A	N/A	N/A	N/A	N/A	N/A	Assume w/properly designed maint. sys. that precautionary repairs do not occur
	Repair wiring, plumbing or connections	E8	0.810	2.268	N/A	N/A	N/A	N/A	X	N/A	N/A	
	Clean Equipment	E9	0.353	0.988	N/A	N/A	N/A	X	X	N/A	N/A	
Final Checkout (F)	Functional check following repair	F1	0.358	1.002	X	X	X	X	X	X	N/A	Assumed to require locomotion to visual access
	Visual integrity check	F2*	0.139	0.389	X	X	X	X	X	X	N/A	

* Not in Original MIL-HDBK-472 Data

TABLE 4
MAINTENANCE TASK/TOOLS TRADESTUDY MATRIX

MAINTENANCE TOOL \ TASK APPLICATION	Joining	Marking	Positioning	Securing	Fastening	Measuring	Aligning	Forming	Transporting	Hammering	Coupling	Cutting	Drilling	Tightening (loosening)	Lubricating	Crimping	Using Standard Test Equip.	Using Special Test Equip.	Cleaning	Smoothing	Illuminating	Inspecting	Stripping
Stimulus Generator																	x					x	
DC Power Supply																	x					x	
AC Power Supply																	x					x	
VTVM																	x					x	
Oscilloscope																	x					x	
Counter (Digital)																	x					x	
In-Flight Test Set																		x				x	
R.F. Fault isolation Unit																		x				x	
Temperature Test Set																		x				x	
Portable Leak Detector																		x				x	
Screwdriver Set	x			x	x		x							x									
Adjustable Wrench	x			x	x						x			x									
Pliers Set			x				x	x			x	x		x		x							
Hammer								x		x						x				x			
Socket Wrench Set	x			x	x						x			x									
Flashlight																					x	x	
Crimping Tool																x							
C-Clamp	x		x	x																			
Mechanical Finger			x	x				x															
Tweezers			x	x				x															
Wire Stripper																							x
Pocket Knife												x											x
Scriber		x					x																
Punch Set		x						x				x											
Hacksaw								x				x								x			
Chisel Set								x				x											
Lubricant Dispenser															x								
Machinist Ruler						x	x															x	
Shears, Metal Cutting								x				x								x			
Tape Measure						x																	
Allen Head Wrench Set	x			x	x									x									
Torque Wrench	x			x	x						x			x									
Open or Box Wrench Set	x			x	x						x			x									
Tube Bender								x															
Inspection Mirror			x				x															x	
Drill, Electric	x							x				x	x							x			
Soldering Iron	x			x							x												
Rivet Gun	x			x	x			x												x			
Vacuum Cleaner																			x				
Wire Wrap Tool	x																						
Electron Beam Welder	x			x							x	x											
Zero-Reaction Space Power Tool	x			x	x			x			x	x	x										

TABLE 5
MAINTENANCE TOOL CHARACTERISTICS

Tool Category	Tool Description	Weight (lb)	Dimensions (inches) L	W	H	D	Power Watts	Estimated Usage (% of All Repairs)	Remarks
Standard Test Equipment	STIMULUS GENERATORS								
	AF-RF Generator	12.0	10-1/2	6	8		60 0	0.5	Similar to Waveform Model 403C
	FM/AM Generator	17.0	10	13-5/8	7			0.5	Similar to RCA WR-69A
	RF Oscillator	17.0	17	16-5/8	11 3/4			0.5	RCA WR-50B w/crystal control (similar to)
	Microwave Generator	38.0	15-1/2	19	7-1/2		85 0	0.5	Similar to Gen. Radio Type 1360A
	Pulse Generator	10.0	16	19	5-1/4		15.0	2.0	Similar to Wavetex Model 155
	Waveform Generator	8.0	7-1/2	7-3/4	5-1/2		5.0	2.0	Similar to Wavetex Model 106 or 107
	D.C. POWER SUPPLIES								
	Programmable 0 to 600 volts	49.0	17	19	5-1/4		350 0	15.0	Similar to Regatran HV-400-1M
	Fixed, Regulated	47.0	16-5/8	19	5-1/4		490 0	15.0	Similar to NJE SR-150-2
		36.0	16-5/8	19	3 ½		345.0	15.0	Similar to NJESR-24-5
		6.0	9-9/16	4-1/4	4-½		60.0	15.0	Similar to Con Avionics R 250.1
	A.C. POWER SUPPLIES								
	Programmable 1 to 120 volts	15 (1 φ) 48.0	(8-1/2 1 φ) 24.5	(8-3/4 1 φ) 17 1/4	(7 1 φ) 21			15.0	Similar to Elco Model 1078-W
	Programmable 0 to 440							15.0	Unable to identify
	MEASUREMENT EQUIPMENT								
	Vacuum Tube Voltmeter VTVM	10.0	4-1/2	7-1/2	6 3/8		5.0	15.0	Similar to Hickok 470A
	OSCILLOSCOPE	21.0	17-4/6	10	6-3/4		30.0	12.0	Similar to Tektronix Model 422
	COUNTER (DIGITAL)	20.0	17	10-3/4	8-1/2		20.0	2.0	Similar to Altic Model 7A96
Special Test Equipment	AUTOMATIC IN-FLIGHT Requirements similar to semi-automatic system + central processor to control test point selection, stimulus application and measurement evaluation	38.0					82.5	45.0	Based on 300 measurement points % usage based on being only test sys. on-board and no use of standard test equipment
	SEMI-AUTOMATIC IN-FLIGHT TEST SYSTEM DC voltage comparators (300 minimum) Preconditioned input signals anormalized to +1 to +5 vdc +100 mvdc Stimulus generator + dc, - dc, 400, 800 and 3200 Hz ac, ground Operating modes Semi-automatic Manual local } Requirements Manual remote } Similar to VTVM Duty cycle - 10% or greater	33.0					35.0	45.0	Based on 300 measurement points % usage based on being only test system on-board and no usage of standard test equipment

TABLE 5 (CONT)

Tool Category	Tool Description	Weight (lb)	Dimensions (inches) L	W	H	D	Power Watts	Estimated Usage (% of All Repairs	Remarks
Special Test Equipment (Cont)	MANUAL IN-FLIGHT TEST SYSTEM Requirements similar to VTVM + Test point switching Capability (300 test points minimum)	18.0					26.2	45.0	Based on 300 measurement points % usage based on being only test system on-board and no usage of standard test equipment
	RF fault isolation unit	5.0	12	12	6		Battery	30.0	Honeywell RF probe (weight includes battery)
	Temperature test set	2.0	4-3/8	5-5/16	3-1/16			5.0	Simpson type 652 adapter for VTVM (similar to)
	Portable leak detector	15.0	11	7	6		Battery	19.0	Teledyne Mode, 6010 (weight includes battery)
Standard Hand Tools	Tools								
	Screwdriver set, 12 pieces	2.03					N/A	100.0	
	Wrench, open end adjustable, 6 in.	0.21	6-1/4	1-5/8	3/8		N/A	55.0	
	Pliers, set, 3 pieces	0.91					N/A	95.0	
	Hammer, Ball peen, 4 oz.	0.38					N/A	7.0	
	Wrench set, socket, 1/4 in. drive w/rachet, extension, flexible handle, 3/16-7/16 in. hex heads	1.60	6-3/4	2-1/16	1 9/16		N/A	3.0	
	Flashlight, 2 cell, flexible head	0.91						4.0	
	Crimping tool, terminal, wire size 14-22	1.07	10-3/4	3	1		N/A	2.0	
	File set, 14 pieces	3.52					N/A	<1.0	
	Clamp, C, set, 2 in.-6 in., 3 pieces	5.70					N/A	<1.0	
	Finger, mechanical, flexible, 15 in.	0.35	17 5/8			3/8	N/A	<1.0	
	Tweezers, hollow jaw, 5-1/2 in.	0.05	5-1/2	1	3/8		N/A	<1.0	
	Wire stripper, wire size 12-32	0.13	5-1/4	2-1/4	5/16		N/A	<1.0	
	Knife, pocket	0.19	3-1/8	1/2	15/16		N/A	<1.0	
	Scriber, machinist	0.04	8			1/4	N/A	<1.0	
	Punch set, tapered, 4 pieces	0.61	6-1/4			3/4	N/A	<1.0	
	Hacksaw	1.25	14-1/2	4-1/2	13/16		N/A	<1.0	
	Chisel Set, 12 pieces	1.62					N/A	<1.0	
	Lubricant Dispenser, 5 oz.	0.42	4-3/4			1-3/4	N/A	<1.0	
	Rule, machinist, 6 in.	0.02	6		1/2		N/A	<1.0	
	Shears, metal cutting, 4 pieces	3.84					N/A	<1.0	
	Tape measuring, 6 ft	0.27	2-5/16	2	11/16		N/A	<1.0	
	Wrench, Allen head, set	1.00	13.6 mx	1.40	3.38		N/A	55.0	Apollo IFM tool and extensions
	Wrench, torque, 10-150 in.-lb 1/4 in. drive	0.70	8-5/8			15/16	N/A	23.0	
	Wrench set, combination, open and box ends, fixed, 5/16 - 1-1/4 in., graduated 1/16 - 1-1/8 in., 15 pieces	8.70	18-1/2				N/A	55.0	
	Bender, tube, 1/8 to 1/2 in., 6 pieces	8.38					N/A	<1.0	
	Mirror, inspection								
	Drill, electric, 1/4 - 3/8 in. chucks	3.85	7 1/2	5	2-1/2		330.0	<1.0	
	Soldering iron, electric, 225 watts	0.90	11			2	225.0	45.0	
	Rivet gun	2.92	5 3/8	1-7/8	6		300.0	<1.0	
	Vacuum cleaner	6.13	11	7	4-1/2		160.0	<1.0	
	Wire wrap tool	0.75	13		1-1/4			<1.0	
	Electron Beam Welder	45.00	10			3.5	500.0	<1.0	
	Zero-reaction space power tool	7.13	11-1/16	5-1/4	9 9/32		85.0	50 95.0	

TABLE 6

MAINTENANCE TECHNIQUES/TOOLS TRADESTUDY SUMMARY MATRIX

Major Subsystem	Applicable Maintenance Techniques						Applicable Maintenance Tasks						Applicable Tools			
	Auto. Mon. and Sw. of BIR	Auto. Mon., Mnl. Sw. of BIR	Mnl. Mon.; Mnl. Sw., of BIR	Modular Replacement	Bits & Pieces Repair	Component Adjustment	Preparation	Malfunction Verification	Fault Location	Procurement	Repair	Final Checkout	Special Test Equip.	Standard Test Equip.	Standard Hand Tool	Special Tools
1 STRUCTURE																
External																
Primary					x		x	x	x	x	x	x			x	x
Secondary					x		x	x	x	x	x	x			x	x
3 LANDING AND RECOVERY																
Parachute or Paraglider	x	x	x				x					x			x	x
Deployment			x		x		x	x	x	x	x	x		x	x	x
Beacon and Communications RCVR/XMTR			x	x			x	x	x	x	x	x	x	x	x	x
4 CREW SYSTEMS																
Portable Emergency Life Support	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x
Lighting	x	x	x	x	x	x	x	x	x	x	x	x		x	x	x
5 ENVIRONMENTAL CONTROL & LIFE SUPPORT																
Oxygen (Including Storage)	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x
Diluent Gas (Including Storage)	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x
Humidity		x	x	x		x	x	x	x	x	x	x	x		x	x
Temperature		x	x	x	x	x	x	x	x	x	x	x	x	x	x	x
Cabin Pressure	x	x		x		x	x	x	x	x	x	x			x	x
Equipment Cooling	x	x	x	x	x	x	x	x	x	x	x	x	x		x	x
6 ELECTRICAL POWER																
Primary Source(s) (Fuel Cell)	x	x	x			x	x	x	x			x			x	x
Secondary Source(s) Battery	x	x	x	x		x	x	x	x	x	x	x		x	x	x
Cryogenic Storage		x	x			x	x	x	x			x			x	x
Inverters	x	x					x					x			x	x
Power Distribution (ac and dc) Busses	x	x	x				x	x	x			x			x	x
Solar Panels			x	x			x	x	x	x	x	x	x	x	x	x
Orientation Mechanism			x	x		x	x	x	x	x	x	x	x	x	x	x
7 STABILIZATION AND CONTROL																
Horizon Scanners, Sun, Sens.	x	x	x	x		x	x	x	x	x	x	x	x	x	x	x
Control Electronics		x	x	x		x	x	x	x	x	x	x	x	x	x	x
Control Moment Gyros				x		x	x	x	x	x	x	x	x	x	x	x
Inertial Wheels				x		x	x	x	x	x	x	x	x	x	x	x
8 SECONDARY PROPULSION																
Engines	x	x	x	x			x	x	x	x	x	x	x	x	x	x
Prop. Storage and Dist.	x	x	x				x	x	x			x			x	x
Press. Storage and Dist.	x	x	x		x		x	x	x	x	x	x	x		x	x
9 COMMUNICATIONS AND DATA MANAGEMENT																
Antenna(s)	x	x	x	x		x	x	x	x	x	x	x	?	x	x	x
Antenna Orientation and Control	x	x	x	x	?	x	x	x	x	x	x	x	?	x	x	x
Antenna Switching				x		x	x	x	x	x	x	x	x	x	x	x
Transmitter(s)	x	x	x	x		x	x	x	x	x	x	x	x	x	x	x
Receiver(s)	x	x	x	x		x	x	x	x	x	x	x	x	x	x	x
TV Camera	x	x	x	x		x	x	x	x	x	x	x	x	x	x	x
Signal Conditioner				x			x	x	x	x	x	x	x	x	x	x
Multiplexer	x	x		x			x	x	x	x	x	x	x	x	x	x

TABLE 6 (CONT)

	Applicable Maintenance Techniques						Applicable Maintenance Tasks						Applicable Tools			
	Auto. Mon. and Sw. of BIR	Auto. Mon.; Mnl. Sw. of BIR	Mnl. Mon.; Mnl. Sw.; of BIR	Modular Replacement	Bits & Pieces Repair	Component Adjustment	Preparation	Malfunction Verification	Fault Location	Procurement	Repair	Final Checkout	Special Test Equip.	Standard Test Equip.	Standard Hand Tool	Special Tools
10 GUIDANCE AND NAVIGATION																
Optics (Sextant and Scanning Telescope				x	x	x	x	x	x	x	x	x	?	x	x	?
11 PRIMARY PROPULSION																
Engine(s)	x			x	x	x	x	x	x	x	x	x	?	?	x	x
Gimbal and Control Sys. (TVC)	x			x		x	x	x	x	x	x	x	x	x	x	x
Propellant Storage and Distribution	x	x	x				x	x	x			x			x	x
Pressurant Storage and Distribution	x	x	x		x	x	x	x	x		x	x	x	?	x	x
12 INSTRUMENTATION																
Malfunction Detect/Warn	Low Duty Cycle and Non-Criticality with Respect to Crew Safety Precludes Consideration															
Engineering Data																
Experimental Data																
Special and Misc. Data																
13 EXPERIMENTATION																
Photographic Equipment				x	x	x	x	x	x	x	x	x	x	?	x	x
Manipulator and Control Sys.				x	x	x	x	x	x	x	x	x	x	x	x	x
14 AUTOMATIC SEQUENCING																
Staging and Separation	x	x					x					x	N/A	N/A	N/A	N/A
Earth Landing	x	x	x				x	x	x			x			x	x
Special Device Deployment	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x
15 RENDEZVOUS AND DOCKING																
Docking Mechanism				x	x	x	x	x	x	x	x	x	x	x	x	x
Rendezvous Radar and Ant.		x	x	x		x	x	x	x	x	x	x	x	x	x	x
Inter-Vehicle Tethers				x	x		x	x	x	x	x	x	x	x	x	x
16 MISCELLANEOUS																
Stage/Spacecraft Separation Devices	x						x					x	N/A	N/A	N/A	N/A
Sensor/Window Covers, Fairings					x		x	x	x	x	x	x		?	x	x
Deployment Mechanisms					x		x	x	x	x	x	x		?	x	x

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ADVANCED SPACECRAFT SYSTEMS IN-FLIGHT DEPENDABILITY

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SUMMARY: This paper discusses design approaches, philosophies, and data that were developed for long-term space missions of 1 to 5 years. In-flight dependability is defined as having been achieved when the resources, methods, and design provisions are available to restore spacecraft operations after failures, thus extending the spacecraft's reliability life. The design philosophy embodies the concept of providing on-board resources and utilizing man's potential to restore the spacecraft. Emphasis is on methods to restore the spacecraft in flight after failure (unscheduled maintenance), although preventive (scheduled) effort is also necessary. Tradeoffs are conducted on design options available for system reliability. Finally, the design options that make possible in-flight restoration are applied to typical subsystems (electric power, environmental control and life support (EC/LS), etc.) of a spacecraft.

The parameters that constitute the capability to restore the spacecraft in flight are considered in each baseline design. These parameters are diagnosis and fault isolation, accessibility, provisions of spares and repair tools, and training of crew. Mission success is greatly improved with a design capability to restore spacecraft in flight when failures occur.

INTRODUCTION

The magnitude of the reliability problem for spacecraft* is illustrated by examining existing technology in the context of future mission times of 90 days to 2 years (see Figure 1). Short-term missions, the 24-hour Mercury and the 200-hour Apollo Lunar Lander, approach the mission requirement of 0.95.

However, this requirement for mission reliability is not met when mission times are increased for the 90-day Apollo Applications, the Venus 1-year fly by, and the 2-year Mars flyby. The 90-day Apollo Mission has an unacceptable reliability limitation of 0.58, based on existing technology capability. The Mars 2-year flyby has an unacceptable reliability of 0.0006. As the mission time is increased from

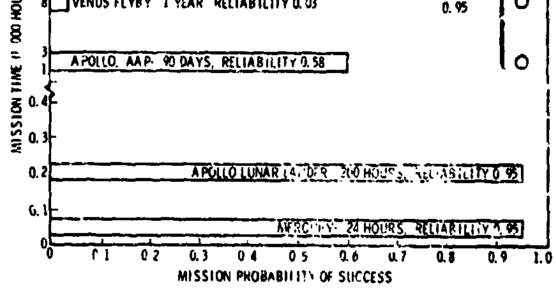

Figure 1. Space-Mission Reliability Requirements and Technology Estimates

*In this paper, spacecraft is the crew ferry or logistics module, which is recoverable and the space station is the orbiting laboratory (not recoverable).

90 days to 2 years, a tremendous disparity occurs between the technology limitations and the mission success requirement of 0.95.

The unacceptable values shown by Figure 1 for each mission take into account the extent of redundancy in the baseline design, e.g., two power supplies (solar cells for primary and batteries for backup) and extensive part redundancy within each subsystem. Therefore, major improvements in design and technology are necessary to increase mission success to acceptable levels.

Design options available to increase system reliability have been investigated. The two most promising approaches are (1) active redundancy with restoration* features and (2) standby redundancy with restoration features. Active redundancy (for example, the electric power source) is essential when the function allows no downtime. Standby redundancy applies when the inherent downtime is adequate to allow the crew to react and switch. However, the above two design options are not always the best approach.

This study was subject to the following constraints: (1) existing or near-term technology, (2) a 1-year space station with a 0.95 mission success, (3) reliability data from such short-term missions as the 200-hour Apollo, the manned orbiting research laboratory, the AAP Workshop, and other programs.

*The terms, "restore" and "repair" are used interchangeably; actually, "repair" denotes physical work on the components of a subsystem, while "restore" encompasses the repair function plus any other means of reinstating the failed operation such as activation of standby or backup modes.

As described in later sections, high levels of success for space missions of 1 to 5 years are indicated for a design that possesses a capability to restore the spacecraft after in-flight failures.

DESIGN CONSIDERATIONS OF A MANNED SPACE MISSION

Major design considerations that are applicable to a manned space mission are listed in Table 1. These considerations become critical design constraints during various phases of the mission. Each subsystem of the spacecraft should be analyzed to check its compatibility with its capability for repair, mission phases, severity and time limits of malfunctions, availability of crew, and other resources.

A timeline analysis should be made of each subsystem for the entire mission profile. A first iteration timeline analysis is necessary early in the design phase to uncover potential problems in sufficient time to make a workable solution possible.

Table 1

DESIGN CONSTRAINTS OF MANNED MISSIONS

1. Identify functions critical to crew safety and mission completion.
2. Determine the time allowed to restore malfunctions.
3. Assess reaction of crew to restore malfunctions.
4. Identify critical maneuvers that inhibit movement of crew (powered maneuvers, rendezous, orbital changes, others).

Design Approaches

The selection of a design approach is influenced by the following considerations:

1. Criticality of the component, assembly, or subsystem to the mission.
2. The inherent downtime to restore the function as permitted by the subsystem.

The first provision demands a high probability of functional success, i.e., a reliability margin for all malfunctions that affect the safety of the crew or the ability to continue the mission. To this end, the study indicated that all subsystems of a long-term space station, i.e., electric power, stability and control, EC/LS, and others, ultimately become flight-critical in various phases of the mission.

The second provision requires that crew-safety items be capable of being immediately restored to operation with either active or standby redundant modes or, in limited cases, with functional backup modes. This also assumes that the space station has a capability for diagnosis and isolation of malfunctions.

Allowable and Estimated Times to Restore

The concern in a space station is for a capability that restores malfunctioning hardware to a satisfactory level within time constraints set by safety and mission requirements. A reliability analysis will provide knowledge of the failures and desired courses of action. This analysis must respond to the following:

1. Whether the function will stop as a result of failure.
2. Whether the failure affects crew, mission, degradation, or experiment functions.
3. The time the function can remain inoperative without jeopardizing crew or mission.
4. Whether protection from failure potential is needed in the form of redundant circuits, backup modes, or spares.

In addition, the reliability analysis must detect failures with potential to cause single, catastrophic loss of crew or mission. This type of failure generally provides little or no response time or may be caused by an uncontrolled reaction (examples are fire, explosions, tank ruptures). In some mission phases, the time to restore a spacecraft function allows little or no downtime, as indicated below under "Illustration." Also, some functions, such as electric power and oxygen supplies, require a capability to maintain operation continuously. The most recent estimates of the time required to restore the subsystems is given by Table 2. Notice that the average time to restore the subsystems is greater than the anticipated time the functions can remain inoperative. Data management, for example, is the least critical; however, in some mission phases the mission objectives for data may not be compatible with a downtime of 401 min. It matters little that the average daily maintenance is very low if, when maintenance action is required, there is not sufficient time available to perform it.

Illustration

The propulsion, navigation, and electrical subsystems illustrate the

Table 2

UNSCHEDULED REPAIR REQUIREMENTS
SPACE STATION SUBSYSTEMS*

Subsystem	Mean Days Between Failures	Average Min. / Task	Average Min. / Day
Life Support	15	141	9.4
Communications	97	95	1.0
Data Management	110	401	3.6
Electric Power	500	284	0.6
Stability and Control	45	129	2.9
(Other Equipment)			4.1
			21.6/day

*From Reference 1.

urgency of these design considerations. Analysis of the electrical power supply indicates little or no downtime because of several critical functions, such as control electronics and emergency lights. Therefore, a secondary source of power with automatic transfer is necessary. Similarly, during powered maneuvers, the propulsion and navigation designs required a backup or redundant mode because the maneuver is critical and the crew is immobile (strapped to couches).

DESIGN OPTIONS FOR IMPROVING EFFECTIVENESS

The design options available to the design technologies to improve the effectiveness of the system are listed in Table 3. The problem for a designer, then, is to select the most efficient option for his design.

The characteristics of these options are described below, as they apply to a manned space mission of

Table 3

DESIGN OPTIONS TO INCREASE SYSTEM EFFECTIVENESS

1. Single-thread design without repair.
2. Single-thread design with repair.
3. Redundancy (one or more) without repair.
4. Redundancy with repair.
5. Majority voting.
6. Special cases: Alternate equipment; functional modes.

long duration. The purpose of these design options is to enhance reliability and assist designers in performing tradeoffs for the best approach, consistent with the mission environments and objectives.

Single Thread Without Repair

With a few exceptions, the single-thread without repair design (Figure 2) is not useful for a space mission, simply because it is too vulnerable to failure. In complex equipment, such as life support (mechanical) or communications (electronic), this approach is subject to an unacceptable failure potential.

Single Thread With Repair

The single-thread with repair design has the advantages of weight, volume, and cost. However, it is incompatible with the downtimes allowed by most functions of the spacecraft. The reaction time allowed by failure of most spacecraft functions is such that the single thread would not yield sufficient time for the crew to react and restore the function. For this reason, this design option is not very useful for the spacecraft. Preliminary analyses also indicate this option is compatible and a good candidate for experiments, such as those that can be suspended when the equipment is undergoing recalibration or repair.

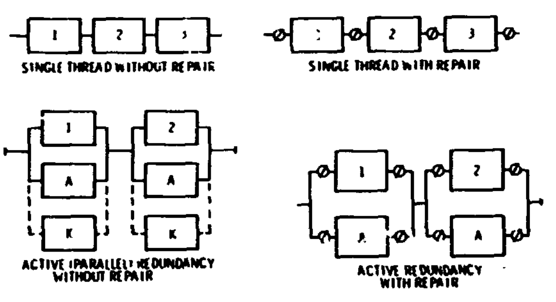

Figure 2. Design Options to Increase System Reliability

Active Redundancy (One or More) Without Repair

The active parallel or redundancy without repair design (Figure 2) has several disadvantages:

1. It uses too many elements to increase the reliability required for a long space mission.
2. In the active phase, the redundant elements are using up their design life. As shown in later sections, the design life of many components is critical, e.g., pumps and control moment gyro assemblies.
3. Electrical or plumbing lines of either liquids or gases use up too much volume and weight, and increase complexity and degree of checkout to support the elements in active redundancy.

Therefore, for a long space mission, the active redundancy without repair is not very useful and efficient.

Active Redundancy With Repair

The active redundancy with repair design (Figure 2) is very useful for many functions in a spacecraft. It is of special interest for functions that allow little or no downtime. For instance, the EC/LS and electrical power subsystems require active redundancy because the function has to be restored almost immediately. Another advantage of the active redundancy with repair is the ability to operate either out of the primary loop, the secondary loop, or perhaps the third loop, while at the same time allowing repair of either of the loops. Therefore, the active redundancy with repair is one of the most promising design approaches for a spacecraft on a long-term mission.

Standby Redundancy Without Repair

The standby redundancy without repair design (Figure 3) has an advantage in that the standby element are not consuming their design life. However, the disadvantage of this design option is the increased loads and complexity from plumbing and electrical lines and switches to support and activate the standby elements as the operating loops begin to fail. This design approach exceeds the volume and weight capacities, even though it conserves the design life of the standby elements.

Standby Redundancy With Repair

Standby redundancy with repair design (Figure 3) is very promising for a long-term space mission. It is compatible with spacecraft functions that allow a nominal amount of downtime: greater than 2 to 5 min. Therefore, it enables the crew to be informed of the failure so they can transfer the function to the standby element. This option conserves the components with limited design life. It also has the flexibility of operating from either the primary or standby loops and at the same time permits repair and checkout of the remaining loops.

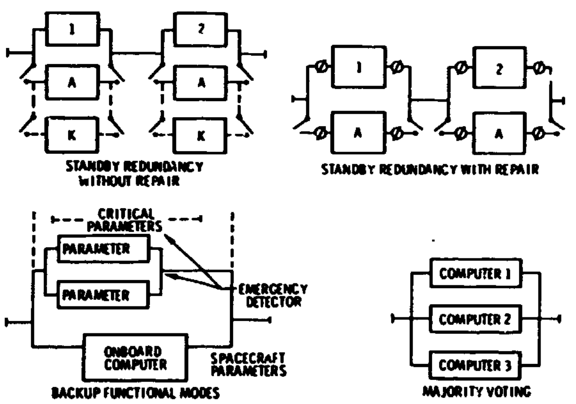

Figure 3. Design Options to Increase System Reliability

Majority Voting

Douglas evaluated the majority-voting approach (Figure 3) for several space and Earthbound functions. This design approach, for example, is useful and efficient for sensors and related circuits of critical parameters (or measurements) of a space mission. Critical parameters are confronted by interacting reliability and safety constraints:

1. A high level of reliability to ensure that the crew is informed of any malfunction.
2. A capability to verify the integrity of the signal or alert condition to prevent a premature abort or loss of mission objectives.
3. A very limited reaction time allowed by a critical parameter for the crew to make a decision.

A majority-voting approach (normally three elements) has the advantage of yielding high-reliability levels and also the inherent capability to isolate erroneous signals.

Alternate or Backup Modes

Alternate equipment or backup modes should be evaluated early in design, such as during failure-mode analysis and systems analysis. This approach is a good means to improve mission success. In this study, an emergency detector continuously monitors critical mission parameters. (See Figure 3.) In addition, the on-board computer can check out all the subsystems, hence, critical parameters, on demand by the crew. In this functional mode, the emergency detector is supported by the computer.

Design Tradeoffs

Each subsystem requires an individual analysis for tradeoffs that consider restoration by the options listed above: redundancy, backup modes, and spares. No single design approach is best or even compatible for all the subsystems of a space station.

The two most flexible design options for a space station are the active redundancy with repair and the standby redundancy with repair. These two options satisfy two recurring demands of a space station design: (1) criticality of the design in meeting crew safety and mission objectives and (2) large failure potential resulting from complexity of design and, therefore, a demand for a capability to restore the function within narrow time limits.

Redundancy and backup modes are not always needed. As an example, the structure subsystem does not have a redundant shell but does have a meteoroid bumper. However, on a recent space-station design effort, a spares kit was added to account for the two expected contingencies that require restoration: meteoroid penetrations and gas pressure leaks. The kit contains wall patches, door seals, extravehicular activity (EVA) exit seals, and docking seals. Airlock doors and EC/LS provisions were also added to convert the structure into three pressurized compartments in an emergency. Design provisions for access and 80 lb of spares kits increase the structure reliability from 0.969 to 0.999 for a 1-year mission.

PARAMETERS OF IN-FLIGHT REPAIR

The capability to restore the spacecraft during flight consists of the four parameters listed in Table 4. These parameters identify the type of resources, such as spares stock, tools, and crew time that will be required. Collectively they respond to the question, "Can the crew restore the spacecraft during flight and how much does it cost?"

Table 4

IN-FLIGHT DEPENDABILITY PARAMETERS

Diagnosis and Fault Isolation

1. Emergency detection unit.
2. On-board computer checkout.
3. Inputs from analyses: safety and abort, failure modes, systems design.

Accessibility

1. To see and inspect.
2. For electrical/fluid test points.
3. Clearance for tools and test equipment.
4. Clearance to remove and replace failed parts.
5. Spacesuit access: EVA, unpressurized areas, pressure cells.

Provision of Spares and Repair Tools

1. Existing deficiencies: depletion of spares, unused stock.
2. Weight constraints.
3. Improvements by reliability data and computer programs.

Training of Crew for Proficiency of Repairs

1. Diagnosis/fault isolation vs level of crew skills.
2. Trend of increasing complexity and sophistication.
3. Skills: nominal, average, critical.

Diagnosis and Fault Isolation

A combination of two approaches is recommended to resolve this parameter adequately. First, an emergency-detector unit is needed to continuously monitor those critical parameters that are vital to crew safety or continuation of the mission. Second, a modification to the on-board computer is required to check out the equipment upon demand by the crew (or automatic sequence). The on-board computer, therefore, will provide a check of readings that could become critical. It will also diagnose and isolate failures for the crew to repair.

Also indicated under this parameter are the types of analysis that will provide design criteria for the emergency-detector unit and the diagnosis and isolation function of the on-board computer. For example, a mission-hazard analysis should be conducted, as indicated by Table 5, for four typical mission hazards. This analysis will identify hazards that may be encountered by the crew throughout the mission. These hazards, therefore, will yield the parameters that must be monitored by the emergency-detector unit to inform the crew of potential hazards and allow them the time required to initiate corrective action. It will also establish other less sensitive parameters, that influence the logic of the computer checkout.

Accessibility

Five access requirements are listed in Table 4. These requirements are necessary for in-flight restoration and will also improve the efficiency of checkout during Earthbound and launch-pad activities. A major problem with existing designs is the inability to restore spacecraft because of obstructions. In most cases, the obstructions can be eliminated by repackaging.

Repairs that require the crew to wear a space suit should be minimized because of the difficulty and expense, in terms of crew resources and equipment. Of course, such repairs must be considered for operations outside the pressurized compartments and for contingencies occurring inside the pressurized compartments. However, if the EC/LS subsystem is accessible with a space suit and has the first priority for repair, other equipment can be restored in a shirtsleeve environment. This is important because repair on complex electronic equipment while wearing a space suit is difficult, if not impossible.

Provision of Spares and Repair and Repair Tools

A review was conducted of spare-part efforts for previous projects. In the past, spare-part efforts have suffered from a common problem: depletion of the needed spare while inundated with tons of unneeded stock. Fortunately, both the reliability and logistics technologies recently made significant improvements in determination of correct spares. Several computer programs and hand analysis techniques have been developed for spares. A hand analysis approach was used to determine the spares, as described below, under "Sparing Logic, Decision Process." A maintenance study[2] of an Earthbound radar system illustrates the magnitude of the spares and maintenance problem (Table 6). The average total time for the radar repair is unacceptable for a space mission.

The improvements in determination of spares have resulted from

Table 5

STATION MODULE MISSION HAZARD ANALYSIS

Hazard and Description	Causes and Effects	Probable Failures [1]	Criticality [2]	Reaction Time [3]	Alert Situation Duration [4]	Remedial Action [5]	Area of Influence [6]	Safe Distance Requirements [7]	Hazard Detection [8]
Pressure Loss [A]	Loss of pressure and life-support gases from space station at a rate greater than could be replenished from internal supplies	• Penetration by meteorite • Failure of internal parts which penetrate exterior • Failure of environmental control system • Seal failure	Minor Slow loss of pressure Catastrophic Explosive decompression	Smaller Holes seconds to hours Explosive Decompression 10 to 12 seconds	Approximately 30 min. if hole or equipment can be repaired	With warning, go into pressure suits and repair equipment or structure	Compartment with leak or entire station in case of explosive decompression	If equipment can be repaired, crew can remain in station in pressure suits	Leakage detection system used with pressure sensing equipment will pinpoint location of leak and rate of pressure loss
Atmosphere Contamination [B]	Contamination of the space-station atmosphere with any toxic gas above maximum allowable concentration	• Fire of any kind • Release of toxic gases from faulty equipment • Failure of environmental control system	Minor to major	Seconds in case of a large fire to hours in case of slow contamination	Emergency will continue until equipment is repaired	• Depressurize and remove cause • Purge with N_2 • Repair equipment • Repressurize	Compartment with faulty equipment	Outside of compartment	Mass spectrometer and other gas analyzers will be used to continuously monitor composition of space-station atmosphere
Temperature and Humidity Out of Limits [C]	• Temperature going out of limits (70° - 80°F) • Humidity not between 40% - 60%	Failure of environmental control system	Minor to major	30 min. to several hours	Emergency will continue until equipment is repaired	• Attempt repairs • If temperature goes out of human tolerance limits (below 0° and above 100°F and 60% relative humidity), abandon station	Compartment with faulty equipment	Outside compartment if temperature goes out of human tolerance limits	Temperature and humidity indicators
Electrical Power Loss [D]	Complete or partial failure of space-station power subsystem	• Several short circuits and failure of circuit breakers • Failure of all batteries because of overloads • Failure of all inverters • Failure of voltage regulators	Major	30 min. to 1 hour	Space station is not habitable if repairs cannot be made	• Attempt repairs • If not repairable, enter LCSM and abandon station	Entire space station	Outside of space station (within 500 yd)	• Power subsystem should be instrumented to measure voltage, current, frequency, and temperatures • Out-of-tolerance fluctuations indicate impending failure

Table 6

AVERAGE REPAIR TIME
RADAR SYSTEM, AN/APS-20 E
(time in hours)

Mean actual repair time	2.5
Mean actual man-hours	4.5
Mean administrative time	43.9
Mean logistics time	75.6

better engineering and reliability data of components and analysis of total systems. Component data, for example, have provided the failure potential of a given component for the total mission based on the failure rates, environments, and operating stresses. Effectiveness of total systems, in turn, has identified those designs or components that are critical to maintain the spacecraft functioning for both crew safety and primary mission objectives. Therefore, the critical components receive an adequate share of the limited resources, (e.g., weight, volume, cost of a spacecraft) and, hence, increase the probability of mission survival.

Some departure from previous trial and error programs should be considered for a space mission. For instance, in Earthbound projects a tendency has existed to balance the number of spares stocked against those used for a given task (mission). However, given a space mission success of 0.95 and an analysis of critical components, the computer or hand analysis technique will stock the components required to support the overall 0.95. The laws of probability dictate that in any one mission, only a portion (10% to 12%)[1, 3] of the on-board spares will be used. However, on the succeeding mission, a different portion of the on-board spares will be used, and so forth for other missions. Therefore, the higher probability or levels of spares increases the availability of spares as uncontrolled or probable failures occur in a given mission.

The cost in spares and tools to support the overall mission success is not prohibitive (see tabular data under "Space-Station System Analysis"). In this study, the spares and tools required for a 1-year space mission are 2,822 lb, which is approximately 6% of the baseline space-station weight of 47,700 lb.

Training of Crew for Proficiency of Repairs

In this parameter, there are interacting design provisions and cost and crew considerations. First, if it is assumed that the crew is well-trained, the amount of equipment for diagnosis and fault isolation can be reduced. However, an opposing interaction is that the trend of spacecraft design is for increasing complexity and sophistication that soon begins to overburden and overtax the capability of the crew. Therefore, the problem is to strike a good balance between the training imposed on the crew and the amount of diagnosis and fault isolation. The objective is to provide an investment of resources that will enable the crew to operate the space station and enable them to cope with the repair problems.

Man's ability to decrease the time for repair is considered in the following discussion in terms of its impact on mission design objectives. Briefly stated, man has the ability to learn and this learning process, commonly referred to as a "learning curve," is a property that applies to all situations involving man. This human ability to learn results in progressive decreases in performance times for iterative tasks[4].

This implies that the crew will decrease the time required to restore the spacecraft after malfunctions, as a function of time and experience with the equipment. The learning process will start during Earthbound checkout and training, and will continue throughout the flight regimes. It is not likely that crew reaction time will be decreased to the point that the design can be reduced to a single-thread design or be vulnerable to "single-point failures." The response to counter critical failures must be so rapid generally that it presents a too-severe requirement even under more benign Earth environment. However, the learning process can be useful to compensate for (1) increasing work that may result from "wear and tear" of equipment, (2) crew resources demands for experiments, and (3) unexpected contingencies.

Estimates[3] of the crew resources required for Earth-orbiting stations are given by Figure 4 and Table 7. The time shown by Table 7, 4.71 man-hours/day, is about equally divided between station operations and scheduled and unscheduled maintenance. For an Earth-orbiting station, approximately 23 min./day of unscheduled (Table 2) and 133 man-min./day scheduled maintenance, for a total of 2.6 man-hours are estimated by the Boeing study[1]. This correlates quite well with the 2.3 man-hours/day estimated by Douglas for scheduled and unscheduled (failure) maintenance.

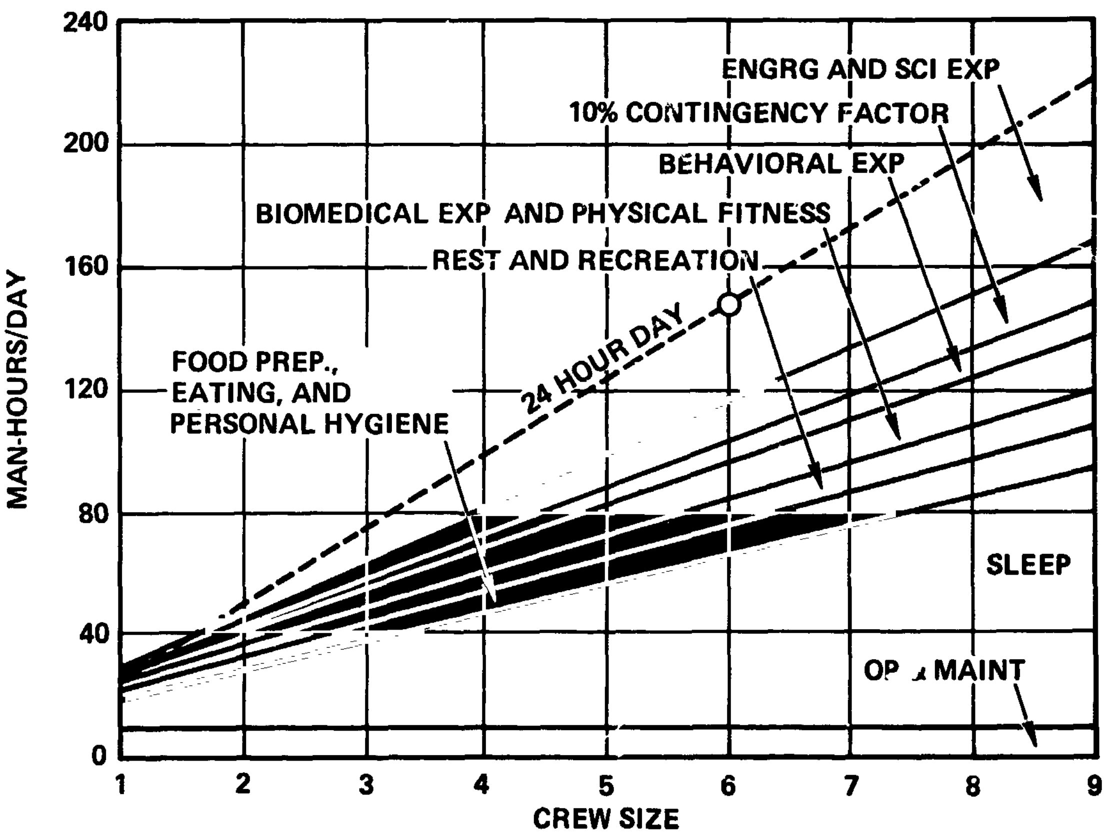

Figure 4. Manned Orbiting Research Laboratory (MORL) IIA – Typical

Table 7

MAINTENANCE AND CHECKOUT TIME MORL-11A SPACE STATION

Subsystem		Average Man-Hours/Day
On-board test and maintenance		0.21
Ferry-resupply craft and cargo module		0.36
Stabilization and control		0.71
Propulsion		0.14
Structural and mechanical		0.86
Communication and telemetry		0.71
Electric power		0.36
EC/LS		1.36
	Total	4.71

The training and skills of the crew should be directed to the following tasks: (1) to assist in developing and validating the diagnosis techniques (2) to test and improve the manual techniques (nonautomatic) for restoration of the spacecraft after failures have occurred, (3) to test the automatic or semi-automatic functions of redundant and backup modes for time-critical failures, and (4) to assist in updating the analytical values for allowable downtime and maximum time to restore spacecraft failures.

SPARING LOGIC (DECISION PROCESS)

Each subsystem must be provided with sufficient spares to reach a nominal level of 0.99 reliability for a 1-year mission. This exacting requirement was necessary to support the 0.95 reliability for the space station.

Reliability and safety margins of design are provided for components whose failure would cause the subsystems to become inoperative. The design requirements are provided by redundant or backup equipment modes. For example, the oxygen and water sources were designed for multiple tanks with independent flow from each tank.

In addition, spares are provided for components that have reliability weaknesses as indicated by limitations of existing technology: failure rates, design life limits, and mission stresses.

The space-station subsystems must fulfill interacting requirements that include maintenance of operation upon failure, repairability, and accessibility. This design approach is compatible with the sparing scheme described below. Two distinct reliability problems must be accounted for in sparing for long-term missions:

1. Spares for constant failure rate or exponential failure.
2. Spares for components with limited design life or "wear-out" failures.

Spares for Constant Rate

Individual assemblies need high levels of reliability to support the nominal 0.99 requirement for the subsystem. For most assemblies or components, a spares stock to support a 0.9999 or greater value was found to be adequate. This value, for each of 100 components or assemblies in a subsystem design will yield the nominal 0.99.

This type of individual spare was determined by hand analysis and with the constraints listed below:

$$R_{O-N} = \frac{1 - (x)^{n+1} e^{-x}}{N!}$$

This approximation* assumes that the sum of the converging series of terms for which spares are not provided is approximately equal to the largest term in the series, i.e., n + 1 spares.

or

$$R = e^{-x} + xe^{-x} + \frac{x^2 e^{-x}}{2!} + \frac{x^3 e^{-x}}{3!} + \ldots \frac{x^n e^{-x}}{n!}$$

where

- n = number of spares of an individual assembly or component.
- R = reliability of an assembly or component required for a 1-year mission, normally 0.9999 or higher.
- x = (failure rate) (mission time).
- e^{-x} = reliability of baseline assembly or component.
- xe^{-x} = first spare (and so forth, for other spares).

Assumptions that have been applied are as follows:

1. Components must be of equal reliability or failure rate (λ).
2. The failure rate of spares in the storage phase is assumed to be zero. Actually, the most recent research into storage and dormant phases indicates a modest failure rate. These phases have a nominal failure rate equal to 1% to 5% of the active phase. However, the storage failure rate is more severe for a few specific classes of components such as hydraulic systems, accumulators, seals subject to an inactive set, or deterioration.

The nominal storage failure rate does not become critical in extending the reliability life of a design. The approach taken to reduce the impact of this problem was:

1. Evaluation of components to ensure that their inherent reliability is not adversely reduced during storage phases.

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*Based on a conversation on pragmatic applications of mathematics to effectiveness with D. J. Davis, Douglas Aircraft Company.

2. Design precautions to ensure that the storage phases plus the mission time do not exceed or approach the design life of components. For this study, the storage phase could be up to 1-year in flight plus additional Earthbound phases.
3. Accounting for storage or dormant rate in the determination of final quantities of spares.

Spares for Wearout

The problem of wearout, not encountered till now because only short-term missions of days or hours were conducted so far, is being met for the first time in 1-year missions. The spares load from wearout was greater than the spares load for the constant failure rate. Therefore, the wearout is the prevailing reliability problem for existing technology of components and assemblies during a 1- to 5-year mission.

The high-reliability demands require that components be replaced by spares before their mean wearout life, at which 50% of the components have failed. To ensure adequate margins, 2.5 to 3 standard deviations from the mean were used to calculate the quantity of spares for typical components. The components that have a wearout life less than their mission expected usage had spares provided to the next highest level of spare. This option provided a margin that protects from wearout deviations and nominal failure rates during storage or dormant phases. The vacuum pumps, for example, of the EC/LS have a design life of 2,500 hr, based on at least two standard deviations. Therefore, three spares plus one for initial baseline provided sufficient spares for 8,720 hours or a 1-year mission.

SUBSYSTEM IMPROVEMENT--LIFE SUPPORT

The water-management subgroup of the life-support subsystem reviews the application of the resources and design approaches described in the above sections. These design approaches are intended to increase the reliability life of a long-term space station.

The nonredundant design had a reliability of 0.420 based on a 1-year mission (Figure 5). Because the entire life support subsystem has a requirement of at least 0.99, the 0.420 for a subgroup was unacceptable. The next iteration was a redundant approach. But with redundancy of the critical loops, the design increases to only 0.550. In complex space-station designs, the yields from redundancy are not as great as might be expected as a result of increased parts and therefore, increased failure potential.

The final iteration was a redundant baseline with spares and a capability to restore any of the failures. The final design is active redundancy, but can also be configured for standby redundancy. It provides operation from either of the two loops. Similarly, repairs can be accomplished from either loop while the other loop maintains the operation. Notice that the reliability increases to 0.999 for 1 year.

Spares were provided for two distinct reliability problems: (1) constant failure rate or exponential failures and (2) components with limited design life or wearout failures. In this design, for example, the filter, pumps, and heaters because of limited life will be

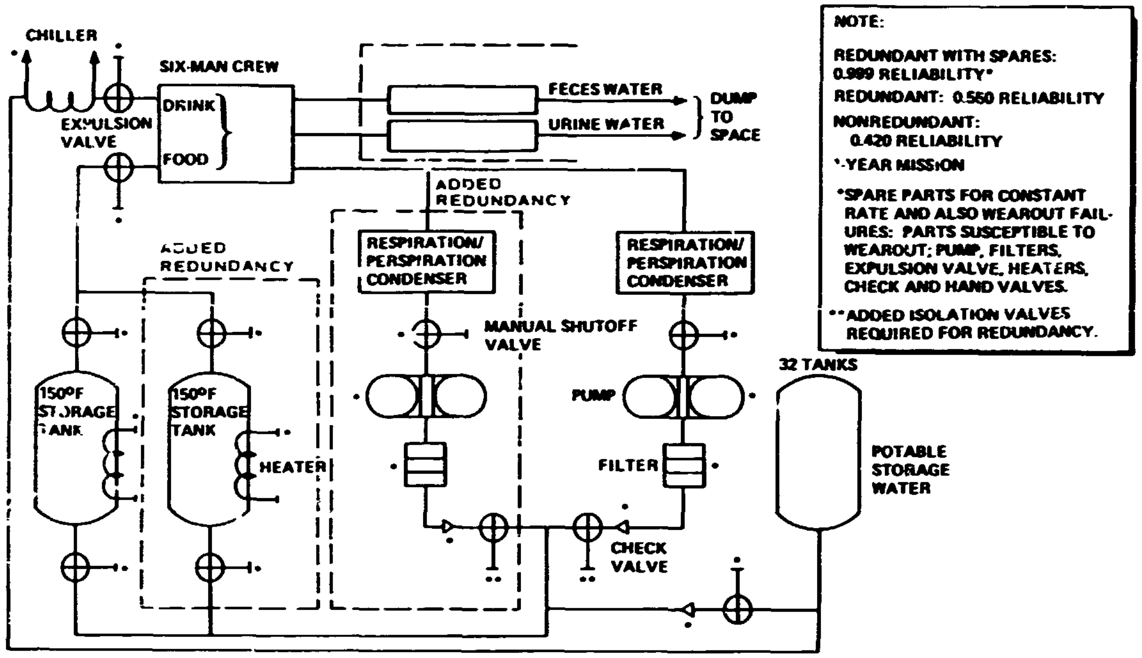

Figure 5. Water Management – EC/LS Subsystem

replaced before the end of the 1-year mission.

SPACE-STATION SYSTEM ANALYSIS

An individual analysis was performed on each subsystem of the space station to select the best design approach as described in the preceding sections. A comparison is given in Table 8 for baseline options versus the increases in their respective reliabilities. The weight and volume penalties are also included.

A summary is presented by Figure 6 of the combined effects of the subsystems upon space-station reliability during a 1-year mission. The single-thread and redundant baseline-design options, are compared with a design and required spares to restore the spacecraft in flight after failures.

A redundant design capability is necessary generally to satisfy crew safety and mission objectives. This design approach complies with the "no single-point failure" criterion of a space mission, which means that no single failure must jeopardize the crew or cause abort of the mission.

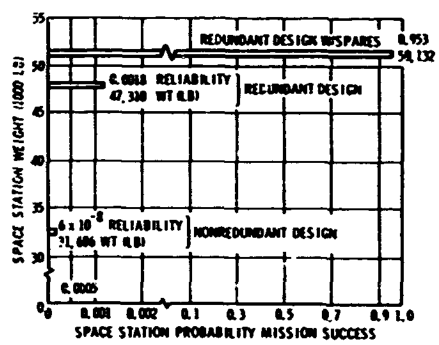

Figure 6. Space Station Reliability – 1-Year Mission

Table 8

SPACE STATION OPERATIONAL EQUIPMENT RELIABILITY PREDICTIONS/SPARES BASED UPON 1-YEAR MISSION

Selected Operational Subsystem	Nonredundant Design			Redundant Design			Redundant With Spares			Crew Skills
	Reliability	Weight (lb)	Volume (ft^3)	Reliability	Weight (lb)	Volume (ft^3)	Reliability	Weight (lb)[a, b]	Volume (ft^3)[a, b]	
Electrical power-solar cells and batteries	0. 358	4, 020	5	0. 941	5, 810	15	0. 994	5, 980	19	Nominal
EC/LS and storage	0. 021	4, 250	100	0. 062	8, 320	170	0. 990	9, 260	195	Average
Stability and control/ electronics	0. 122	1, 575	120	0. 413	2, 985	228	0. 996	3, 067	230	Average
Communications and data management	0. 0002	871	196	0. 082	1, 125	249	0. 989[c]	2, 425	273	Average
Stability and control/ propulsion	0. 414	6, 000	--	0. 986	7, 495	--	0. 986[d]	7, 495[e]	--	Not applicable
Structure	0. 723	11, 770	40	0. 969	17, 915	52	0. 999	17, 995	54	Nominal
Crew systems	0. 926	3, 200	210	0. 926	3, 660	235	0. 999	3, 910[f]	252	Nominal
Summation	6×10^{-8}	31, 686	671	0. 0018	47, 310	958	0. 953[g]	50, 132[h]	1, 023	--

[a] Volumes are only for the displacements of hardware within the pressurized compartments (15, 000 ft^3); weights are total for all areas.

[b] Weights or volumes do not include interconnections, tubing, or wire harness.

[c] Will be 0. 989 for 97% data return; for 100% data return, reliability is 0. 961.

[d] Stability and control/propulsion improvements in control circuitry will raise reliability from 0. 961 to 0. 986.

[e] Includes 5, 603 lb of propellants and pressurants.

[f] Spares only for critical items such as space suits, personnel life-support system (PLSS) unit and EVA equipment.

[g] Improvements possible as noted in (c) and (d) will raise overall reliability from 0. 907 to 0. 953.

[h] Weights and volumes include allowance for maintenance tools and equipment, total spares weight is 2, 822 lb.

The active and standby redundancy with a restoration capability are the most flexible options and were used extensively. However, some exceptions were necessary as follows: (1) structure, as described earlier; (2) propulsion; and (3) stability/control. The propulsion thrustors, tanks, lines, and so forth are redundant but, for safety, are not repairable. The hazards result from use of storable, hypergolic propellants and also from the difficulties resulting from location in unpressurized areas of the station.

The stability and control electronics use a hybrid of design options. The sensor assemblies are not repairable in space and also are limited in useful life to about 2,500 hr. The sensors include attitude and rate gyros, star trackers, horizon sensors, and control-moment gyros. For these reasons, the sensors are installed in a standby redundancy design with two to three sensors in standby, plus one for initial operation. Most other electronics are repairable and, therefore, are installed in a single redundant, active design with a repair capability. Failures of the control electronics are restored with such replaceable modules as circuit boards. Upon malfunction, the design allows repair of either of the two control loops or circuits, while the other sustains the operation. These design provisions require approximately 82 lb of spares and increase the reliability from 0.413 for a design that is only redundant to 0.996 for redundant with spares.

CONCLUSIONS

As a result of the study, the following conclusions were reached:

1. No single design approach is optimum for the diversity of designs and technologies of a manned space station. Therefore, tradeoffs that consider redundancy, backup modes, and spares are necessary on each subsystem.
2. Given existing and projected limits of technology, spares and design provisions to restore the space station in flight after failures have advantages over alternate design approaches.
3. The penalties in crew resources to restore the space station in flight appear to be within practical limits.
4. Single-point failures must be eliminated from subsystem design because of the hazard to the crew and mission success objectives. Similarly, design provisions to restore the spacecraft are required to conserve the margins of crew safety and functional levels of backup on redundant modes for reliability.
5. This paper has shown the improvements in crew safety and mission success that can be accrued by providing a means to restore the space station in flight after failures. However, the parameters and resources that are required for these improvements are marked by uncertainties. The uncertainties result, in part, from insufficient test data and because, at this juncture, a space station has not been built with a design objective to restore the space station in flight.
6. Alternate studies have been undertaken by Douglas to develop criteria to increase

the reliability and design useful life of components. The emphasis is to replace in-flight maintainability with improved designs to fulfill high reliability for space missions of 1 to 5 years. The indications are that the final result will be a hybrid of both.

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THE IMPACT OF MAINTAINABILITY ON EC/LSS DESIGN

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SUMMARY: A novel Environmental Control and Life Support System (EC/LSS) was designed for space station use. High reliability is achieved for earth orbital missions by utilizing the concept of on-board maintainability. Major maintainability elements, such as commonality, packaging and sparing of components are discussed. The CO_2 Reduction Subsystem is singled out to illustrate the complex interrelationship between maintenance level, crew involvement and system weight. Special designs of maintainable valves and regulators are described.

INTRODUCTION

Current spacecraft are designed to achieve reliability by built-in redundancy. A typical 3-man, 14-day mission is supported by a non-regenerative EC/LSS consisting of 150 components, many of which must be redundant in order to attain the 0.993 reliability goal. Assuming that this EC/LSS could be extended to a five-year mission, EC/LSS reliability would drop to a totally unacceptable 0.365, on the basis of a failure rate constant at 21×10^{-6} failures/hr. This failure rate would, most likely, increase as equipments approach their wear-out-periods thus causing a further decrease in reliability. The search for a new approach to high reliability then becomes mandatory.

SPACE STATION EC/LSS RELIABILITY

Space stations are being planned now to take 4-to 12-man crews into earth orbit for future missions ranging from 1 to 5 years. Payload limitations and cost considerations demand that "the loop be closed" by the application of regenerative EC/LSS. This results in a sharp increase in complexity and a decrease in reliability. A 0.950 reliability goal was set for a 4-man crew on a 5-year mission with 90-day resupply. Conventional methods were employed to meet this goal:

- relaxation of the demand for minimum weight and minimum power requirements
- refinements in design
- increased redundancy

These methods were not particularly successful, because a small incremental increase in reliability was accompanied by greater incremental increase in system weight.

SYSTEMS ANALYSIS INDICATES MAINTAINABLE EC/LSS IS REQUIRED FOR LONG-DURATION MISSIONS

A paper by Frumkin and Hodge, Reference 1, states that long-term missions may not be weight-effective with redundant systems, or, to use the paper's term, "non-maintainable systems." Based on a study recently conducted at Grumman, the authors plotted the function "system weight versus mission time" for non-maintainable and maintainable systems. After an initial savings, the non-maintainable system quickly loses out to the maintainable system. From the cross-over point - which may be taken at 90 days, or less, for the mission discussed above - the non-maintainable system increases at an increasing rate whereas the maintainable system increases at a decreasing rate.

In their comparative analysis, the "non-maintainable system" is understood to mean a system in which enough redundant components are provided with automatic switch-over to reach the desired reliability goal. Conversely, the "maintainable system" contains functions and devices that permit diagnostic and remedial action by the crew. As long as enough spares are provided, such a system could be maintained indefinitely.

In electronics the practice has been to use standardized circuitry with built-in, self-checking logic and common components. Also, multi-point connectors of high reliability have been developed to permit the convenient replacement of plug-in modules. When comparing electronic systems to fluid-mechanical systems, it is surprising to see how far the latter is lagging in the application of maintainability.

MAINTAINABLE EC/LSS DESIGN STUDY, OBJECTIVES AND IMPLEMENTATION

The problem of designing a highly reliable EC/LSS for space station use can be resolved only by the full scale adaptation of maintainability to fluid-mechanical systems. Recognizing this challenge, Grumman started an in-house study program during the fall of 1967 that was dedicated to the development of a truly maintainable EC/LSS for long duration missions. Maintainability was established as the major design goal in the definition of a space station EC/LSS which could be installed in a manned altitude chamber for design verification and development testing.

In an attempt to dramatize maintainability, the original mission model was abandoned (5-year mission, 90-day resupply, 0.950 reliability goal) in favor of a 5-year mission, without resupply, but for the same reliability goal. In search of a new approach, five major design objectives were defined:

- o Built-in fault detection and isolation capabilities
- o Compatibility with the on-board checkout system (OCS)

- o Equipment layouts must permit direct access for the replacement of failed modules and/or components

- o Valves, couplings and hoses to permit evacuation and refilling of liquids and toxic gases from failed components

- o Maximum use of commonality and interchangeability (valves, regulators, couplings, gages, meters, etc.) to permit minimization of spares requirements on a total system basis and to simplify the overall maintenance task

These objectives were implemented by concurrent investigations in the following areas:

- o Extensive failure mode and effects analysis (FMEA), at the subsystem level, to identify components critical to the crew's safety

- o Establish maximum allowable downtimes to ascertain subsystems amenable to maintainability by removal and replacement methods (R&R)

- o Categorize component functions and fix ranges of operating parameters, to determine the degree of commonality which can be achieved without compromising system performance

- o Evolution of new maintainability concepts, through packaging studies and from considerations of human factors, that would yield the most practical level of maintenance.

- o Reliability analyses to define spares requirements

- o System and subsystem layouts with major emphasis on man-machine interfaces and subsystem performance.

AiResearch Manufacturing Div. of Garrett Corporation conducted an EC/LSS design study for Grumman that produced a design which is unique in that it is maintainable for five-year missions - with or without resupply.

DESIGN STUDY RESULTS: HIGHLIGHTS OF A MAINTAINABLE EC/LSS

The system designed represents a true prototype using flightworthy concepts, materials and design features (see Reference 2). For the first time, an EC/LSS has been designed which can support 4 men on missions up to 5 years without resupply. This system is regenerative and maintainable; it comprises 589 components without redundancy. Figure 1 shows a schematic of the base-line system. A flight version would weigh 1660 lb dry and 1730 lb wet. Power consumption is rated at 4000 watt.

This EC/LSS consists of 10 subsystems: Water Management, Humidity Control, CO_2 Removal, CO_2 Reduction, Oxygen Generation, Trace Contaminant Removal, Thermal Management, Cabin Temperature Control, Suit Circuit and Waste Management.

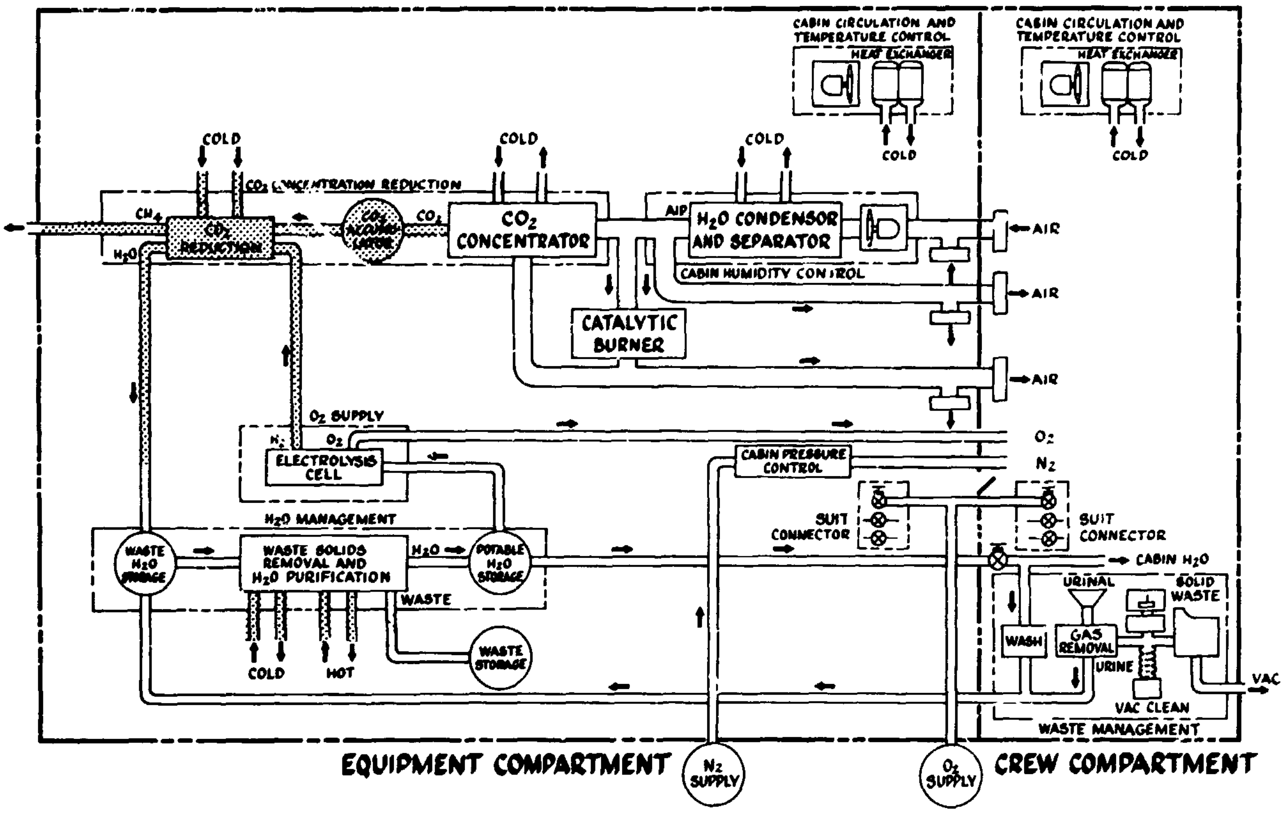


FIGURE 1 BASE LINE SYSTEM FOR SPACE STATION EC/LS

FMEA's revealed weaknesses in subsystem design. Reiterative design steps were therefore taken to reduce downtimes to permit on-board maintenance. For example, the FMEA resulted in a complete redesign of the Water Management Subsystem which now features 7 tanks and 10 control panels. Patch-cords with quick-disconnects permit positive isolation of contaminated waters by means of manual switching.

Table 1 lists the Maximum Allowable Downtime for all 10 subsystems. The Thermal Management and the Cabin Temperature Control Subsystems are critical and have maximum allowable downtimes of only 0.5 hours. All other subsystems have allowable downtimes of the order of hours to days, to provide ample opportunity for maintenance. Additional studies should determine if standby-redundant coolant pumps are needed for the Thermal Management Subsystem.

All subsystems, except for the Thermal Management, were packaged in modular form and mounted in the EC/LSS Console (see Figure 2) for direct access for maintenance, from both front and rear, (see Figure 2).

The basic groundrule was that components in need of service should be accessible without prior removal of adjacent components. The final design in the evolution of packaging techniques featured panel mounted valves and gauges, but the panel itself is not removable. This design was judged optimum, since it provided the widest possible range of applications to component commonality. Maximum component

TABLE 1 MAXIMUM ALLOWABLE DOWNTIME

Subsystem	Downtime, hr	Effect
Thermal management (external failure, complete shutdown)	0.5	Shutdown: Humidity control, CO_2 removal, CO_2 processing, Water recovery, Cabin temperature control
Cabin temperature control	0.5	Cabin temperature increases above tolerable limits (>100°F) if both units failed
	Indefinite	Cabin temperature increases to 80°F with one unit failed
Humidity control	3.5	Cabin dew point increases to 70°F
CO_2 removal	75	Cabin CO_2 partial pressure increases to 30 mm Hg
CO_2 processing	12.5	Loss of CO_2 overboard: 6 lb CO_2 loss assumed.
O_2 generation	12	Cabin pO_2 decreased from 160 mm Hg to 150 mm Hg
Water management		Water supply restricted to essential requirements
• Water recovery	120	
• Water distribution	3	
Cabin pressurization	14	Cabin pressure drops to 5 psia
Trace contaminant removal	Probably not critical	
Suit circuit	Not critical	No suited operation until repair completed

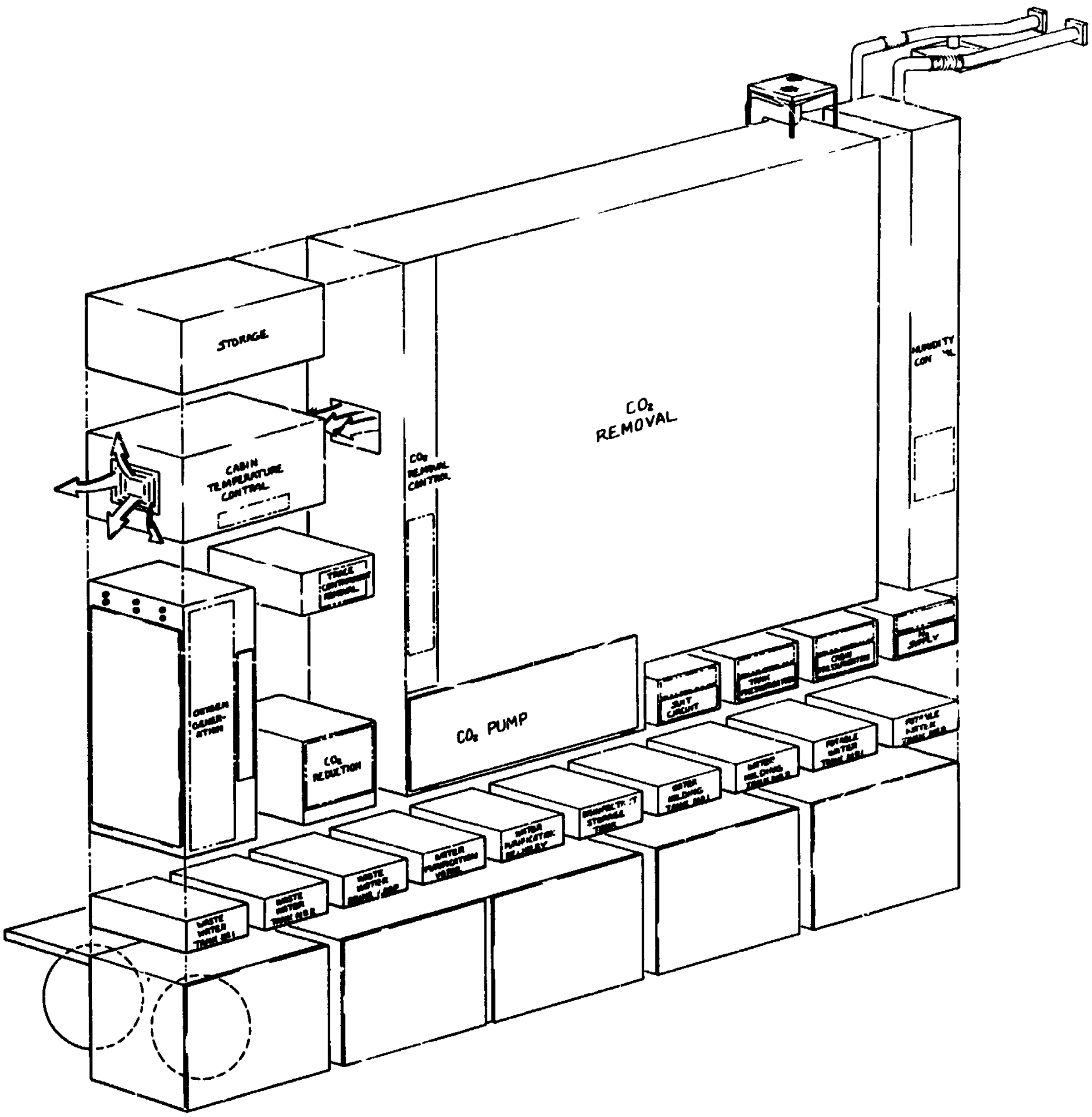


FIGURE 2 INSTALLATION OF MAINTAINABLE EC/LSS FOR GRUMMAN SPACE CHAMBER

commonality produces a lighter EC/LSS, since there are fewer spares. Figure 3 illustrates the spares requirement for the proposed system. 640 lb of spares need to be launched initially for a 5-year mission with 90-day resupply versus 1700 lb for a 5-year mission without resupply. The total systems weights corresponding to these launches are 3890 lb and 18,100 lb, respectively. These weights include all consumables.

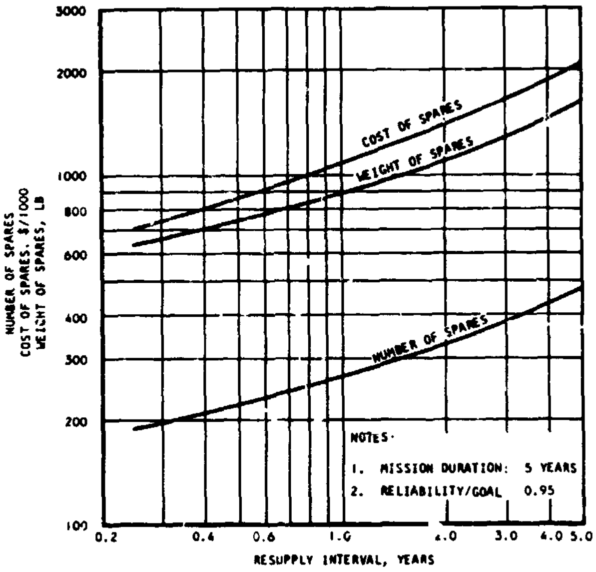


FIGURE 3 SPARES REQUIREMENT FOR MAINTAINABLE EC/LSS

COMMONALITY, A KEY FACTOR IN THE DESIGN FOR MAINTAINABILITY

The degree to which commonality has been stressed is indicated by the fact that the 589 components are based on 127 different component designs. A single component (a manual shutoff valve), specifically conceived for space station use, is deployed at 80 different locations throughout the system. Other components, such as quick-disconnects, instruments and fans are used in as many as 6 to 98 different locations.

COMMON SHUTOFF VALVE

Investigation of the operational requirements for this type valve revealed that a common design should:

- Have a flow passage of approximately 3/8 in.
- Be capable of sealing against pressures in the range from 0 to 100 psia
- Be able to withstand a hard vacuum.

Figure 4 depicts a valve which weighs 1.2 lb and fits into a 4 x 4 x .5 in envelope. This insert-type shutoff valve, proposed for all applications, required a flow passage of up to 1/2 in diameter.

This removable cartridge design speeds replacement of the valve's working parts. The use of monoblock assemblies is also facilitated where many functions are grouped in a single area. The monoblock method utilizes castings or brazed assemblies that eliminate the many mechanical joints used with conventional tubing and fittings.

The receptacle portion of the valve is made common except that the location and size of the inlet and outlet ports are specific to each application. The receptacle is brazed to the

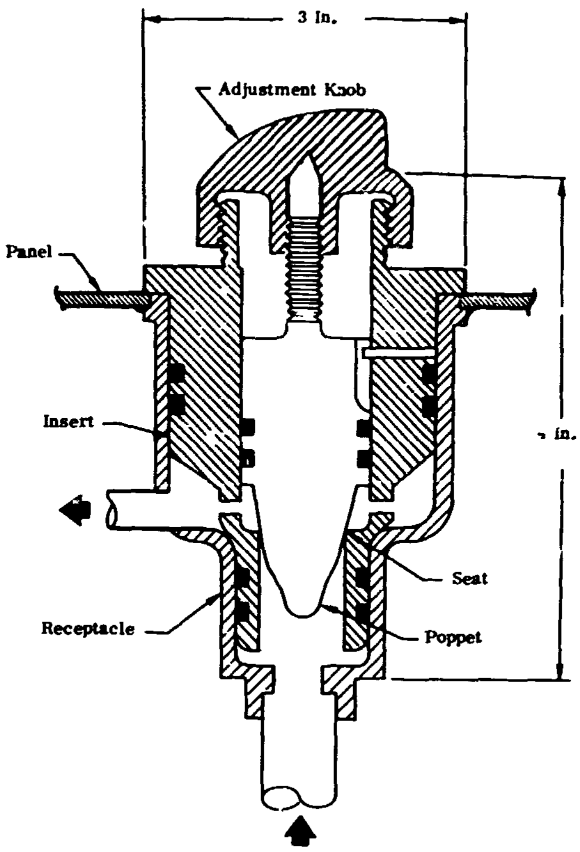


FIGURE 4 COMMON SHUTOFF VALVE

subsystem structure, i.e., the panel, and to its interface connections, thereby making it a highly reliable stationary part. The insert portion is identified as the valve proper and contains all the seals and working parts for flow control or shutoff. The dual O-ring seals contained on the insert valve bear against the inner walls of the receptacle to isolate inlet and outlet ports. The valve's inner diameter is stepped down to provide a flow annulus for the side port and to prevent sliding the O-rings past the side outlet port.

The insert portion provides flow control in accordance with the position of the manual control knob. The taper of the poppet has a variable slope; the first part permits slow opening to provide metering control for low flow applications. The second part is shaped such that a few turns of the knob produces full opening of a $\frac{1}{2}$ in. flow passage. A pin in the valve body protrudes into the machined groove in the side of the poppet to prevent it from rotating when the valve is opened or closed; this reduces wear on the O-ring seals and greatly prolongs their life.

An inner and outer thread is included in the knob to accommodate

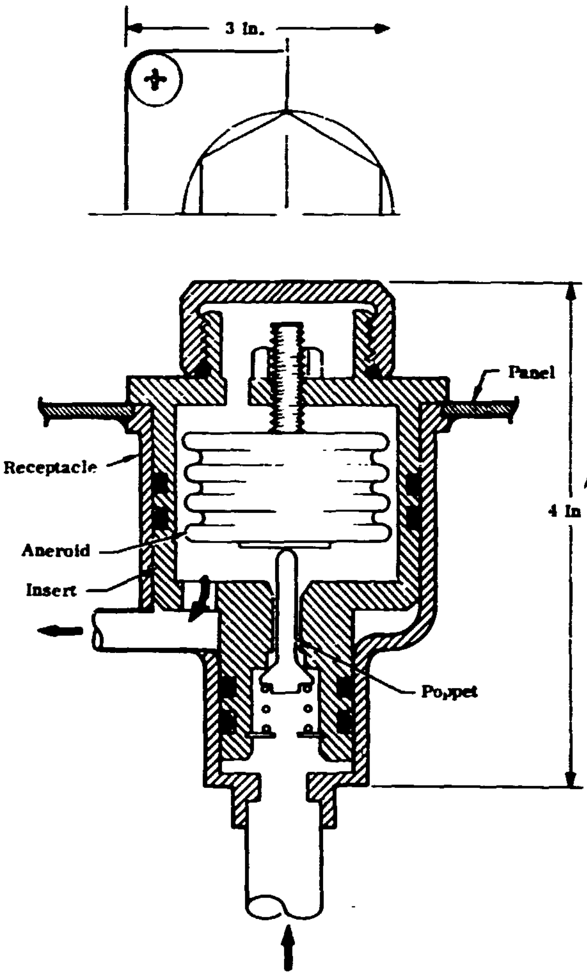


FIGURE 5 COMMON PRESSURE REGULATOR

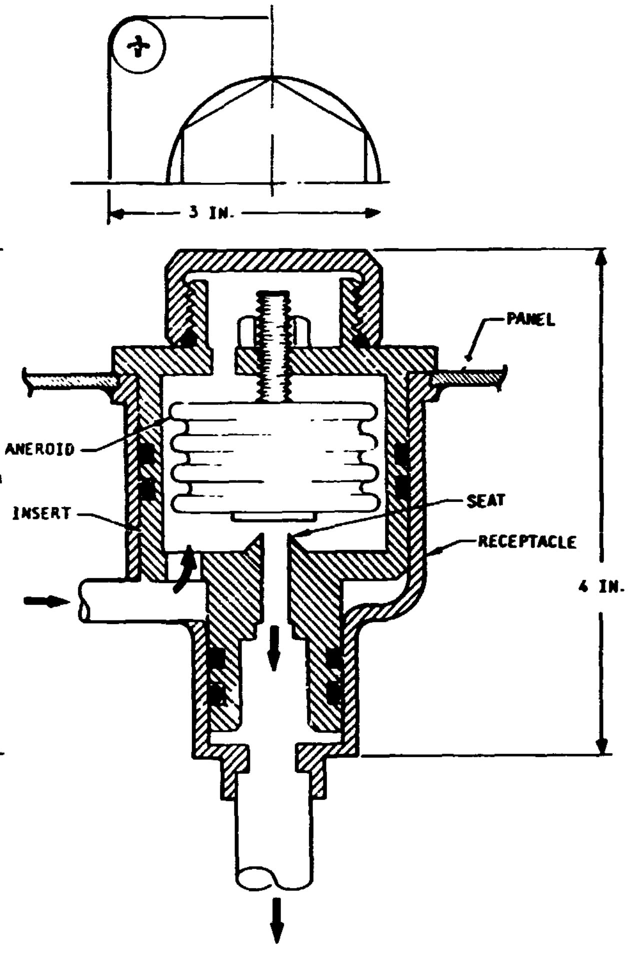


FIGURE 6 COMMON PRESSURE RELIEF VALVE

the non-rotating poppet. The selection of the number of threads per inch on the inner and outer thread will depend on the displacement per turn to satisfy the metering requirements. Position indication of the valve can be incorporated by attaching a device to the knob to monitor the relative displacement between the knob and the threaded portion of the poppet.

COMMON PRESSURE REGULATOR

Four pressure regulators are required in the EC/LSS to produce outlet pressures of 7, 10, 20 and 30 psia. A single regulator was designed to accept all performance and media requirements, except that individual calibration for each of the four applications is required.

Figure 5 shows this regulator which, in many aspects, is identical to the shutoff valve described above. Weight, size, porting and sealing of the receptacle are identical. The regulator insert is shaped differently to house a poppet valve with an integrally molded elastomeric seal that is spring-loaded against the seating surface in the valve portion of the insert. The small diameter poppet is made from pinion gear stock with the outer diameter of the gear accurately guided by the bore in the regulator. When the poppet is depressed from its seating surface, the flow is directed through the gear teeth to the outlet port via the aneroid chamber. The aneroid senses outlet pressure and contracts or expands depending on the outlet pressure. The aneroid position is calibrated (on a test bench) such that it pushes the poppet off its seal at a preselected pressure. The aneroid adjustment is secured by the jam-nut during calibration.

A cap is installed over the aneroid adjustment to provide a seal and to protect against inadvertent changes to the adjustment.

COMMON PRESSURE RELIEF VALVES

Eight different relief valves are used in the EC/LSS. The required relief pressures range from 0.35 to 100 psia. The relief valve designed for these applications is presented in Figure 6. This valve can be used for all of the eight applications, except for calibration and/or aneroid change. Many of its features are identical to those of the valve and the regulator previously discussed.

Common cartridge design for shutoff, relief and pressure regulating functions offers the advantages and the flexibility of interchangeability of components. Use of common parts such as bodies, O-rings, aneroids and adjustment caps, can reduce spare stocks if bench level maintenance is contemplated. Use of the cartridge design concept also provides valuable backup features. If spares are not available, a manual valve insert could be used in an emergency in place of a failed aneroid. Commonality and interchangeability at both component and parts levels make it possible to continue subsystem operation in a degraded mode. A test bench is required for recalibrations and/or aneroid changes of the pressure regulators and relief valves. The basic equipment required in the test bench is schematically shown in Figure 7. While special equipment such as test benches adds weight to the system, this addition is considered minor when compared to the reductions achieved by extending the

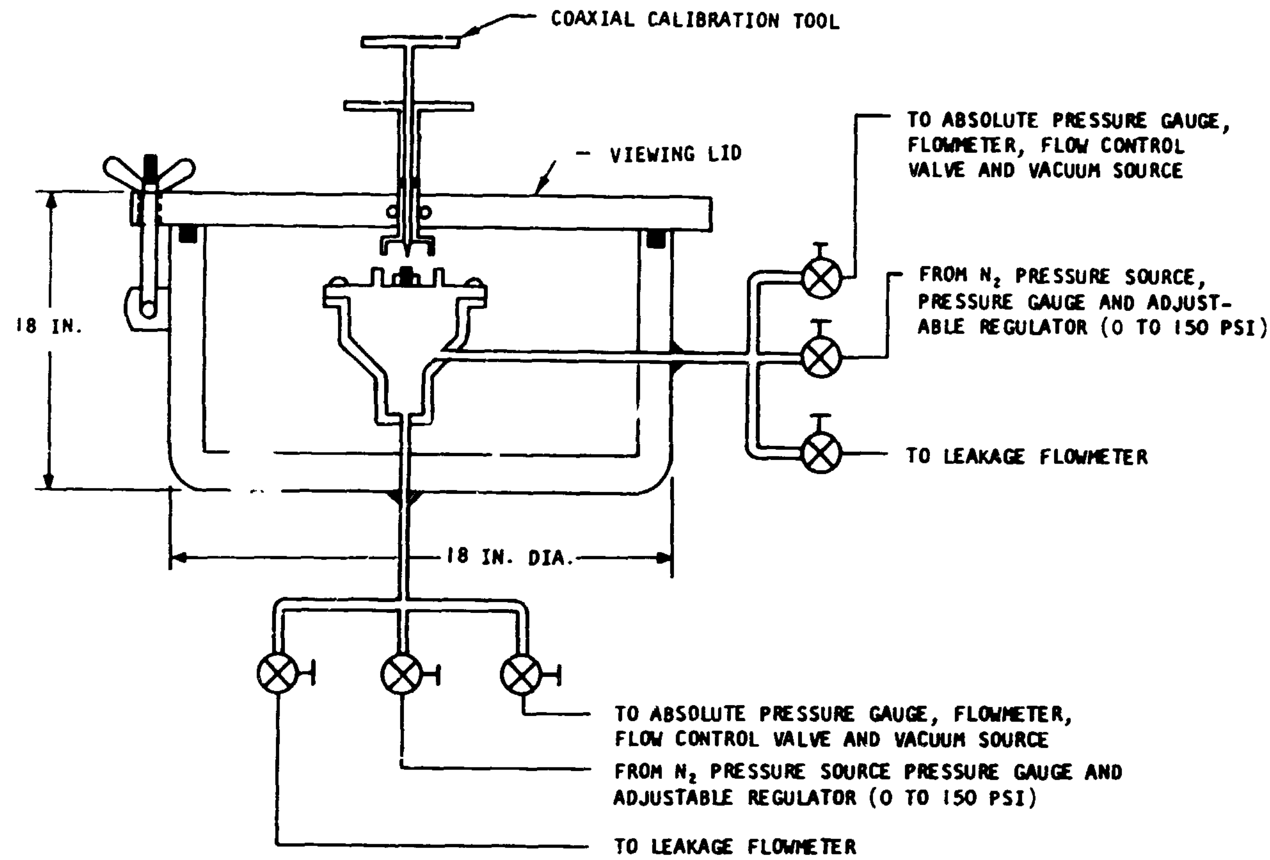


FIGURE 7 TEST BENCH FOR REGULATORS AND RELIEF VALVES

application of commonality and interchangeability from the component level to the parts level.

The case of the maintainable EC/LSS can be further illustrated by comparing its design use ratio and its redundancy percentage with those of current and past spacecraft programs. A synopsis of this comparison is contained in Table 2. The last column entitled "Space Station Simulator EC/LSS" clearly shows the progress made during the design study.

RELIABILITY CONSIDERATIONS

Following initiation of the FMEA, all subsystems were documented with detailed schematics for the performance of reliability analyses. Subsystem goals of 0.995 were established by predicating an overall reliability of 0.950 for the 5-year mission, and by apportioning the overall reliability goal equally among all

TABLE 2. EC/LSS DESIGN EFFICIENCY

Program	Past Spacecraft EC/LSS	Current Spacecraft EC/LSS	Space Station Simulator EC/LSS
Crew Size	2	3	4
Functional Components	78	121	325
Component Designs	47	62	73
DUR	1.65	1.95	4.45
Redundant Components	23	30	4
RP	30	25	1.2
Sensors	33	24	114
IR	0.44	0.20	0.35

$$\text{Design use ratio DUR} = \frac{\text{Number of Functional Components}}{\text{Number of Component Designs}}$$

$$\text{Redundancy Percentage RP} = \frac{\text{Number of Redundant Components}}{\text{Number of Functional Components}} \times 100\%$$

$$\text{Instrumentation Ratio IR} = \frac{\text{Number of Sensors}}{\text{Number of Functional Components}}$$

10 subsystems. Component failure rates were enumerated with the aid of two sets of data. One set consists of current spacecraft data, the majority of which is analytically predicted by the examination of component designs, and the minority of which is the result of life testing. The second set of data is based on operational experience with jet aircraft that have similar components. Generally, the stress environment of jet aircraft is considered to be ten times more severe than that of an earth orbiting space vehicle. For example, aircraft experience indicates a mean time between replacement (MTBR) of 10,000 hours for an ECS-type fan as compared to a meantime between

failure (MTBF) of 200,000 hours estimated for the current spacecraft. For the purpose of the Grumman study, values of 100,000 hours (MTBF) were used, i.e., values 10 times larger than the aircraft MTBR.

In order to keep subsystem failure rates low enough to meet reliability goals, it became necessary to provide even the simplest component with a backup unit to serve as a replacement. Consequently, every component had to be replaceable. Statistically, even the simplest component, such as a heat exchanger, may require replacement in the course of a 5-year mission (Reference 2).

WHAT LEVEL OF MAINTENANCE IS DESIRABLE?

In general, R&R of failed equipment was considered as the only realistic type of maintenance. Repair work, such as the mending of broken or leaky equipment was not intended, because this would have required technology not presently available. The adjustment of aneroids represents a borderline case, because the adjustment may be used to ready a general spare for installation in a specific application, as to readjust a malfunctioning regulator for renewed use.

Equipment maintenance was contemplated at four different levels: subsystem, module, component and component parts. What level of maintenance is most desirable? There are two extremes, replacement at the subsystems level and replacement at the parts level. Replacement on an entire subsystem requires a minimum of time, skill, tools and instrumentation. The penalty is an extremely high system weight. Maintenance on the parts level, on the other hand, demands a maximum of time and skill on part of the crew, and special tools and fixtures. Maintenance on the parts level may result in the lowest possible system weight, if commonality and interchangeability are diligently applied throughout the entire EC/LSS design and sparing is accomplished on the parts level (one size O-ring for the entire EC/LSS). What is the best location for maintenance: on-line in the EC/LSS console, or at the EC/LSS bench in the onboard workshop?

Since on-line R&R and parts level R&R would be a combination of extremes, it was rejected. The issue of single action versus dual action maintenance was reckoned with. In single action maintenance, failed equipment is removed from the console and replaced. In dual action maintenance, failed equipment is taken to the workshop where additional maintenance is performed at the next lower level. The subject of the most practical maintenance level and action must be discussed further in conjunction with packaging, since R&R is not possible without easy access and removal.

METHOD OF ISOLATION

When the concept of R&R at the component level was adopted, each subsystem was studied to determine the best methods of isolation, system drainage, and the replacement of components. There were two basic approaches: individual component isolation and group or subsystem isolation. Component isolation was rejected because:

o Special provisions would be required for fluid removal and replacement from each individual component, which would impose extremely high penalties on the system.

o The subsystem must be shut down, in any case, for component replacement.

Subsystem isolation was therefore selected. Isolation valves, selector valves and bypass loops were provided for subsystem shutdown. Quick-disconnects allow liquids or toxic gases to be emptied through a quick-disconnect hose assembly and into an overboard vacuum connection. Leak checking, evacuation, sterilization, refilling, etc., can be performed via the same connection. Figure 8 shows the remove and replace methods.

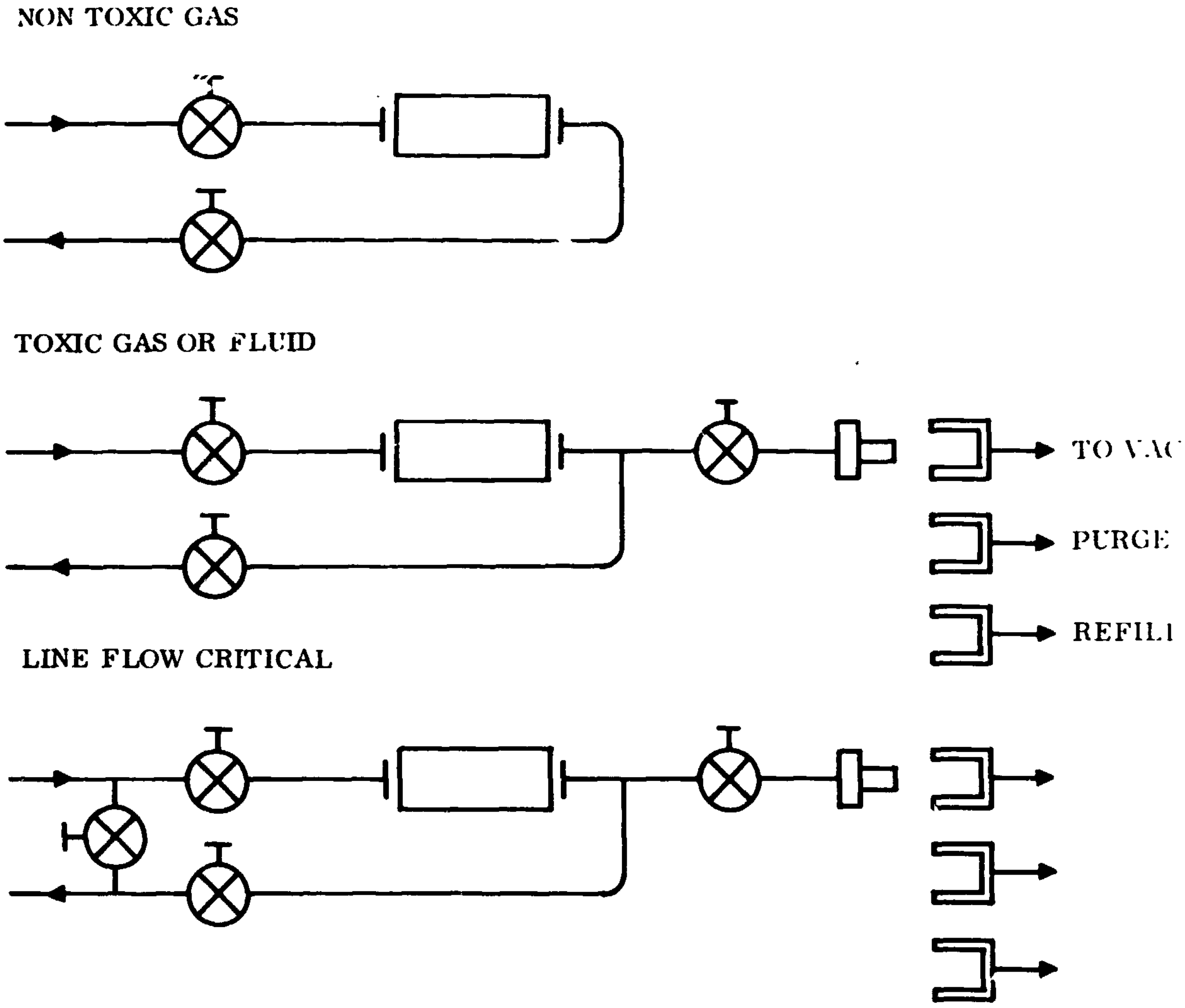


FIGURE 8 REMOVE AND REPLACE METHODS

PACKAGING CONCEPTS

Investigations of various approaches to maintainable package design were guided by the following criteria:

- o Minimum subsystem downtime
- o Unlimited time for component R&R
- o Ease of component replacement
- o Small package size
- o Low weight by minimizing spares requirements
- o Reasonable cost by realistic appraisal of design and development programs

The idea of modular packaging comes intuitively to the design engineer, but the level and type of maintenance action should be decided, after rigorous analysis, in accordance with the criteria outlined above. In typical dual action maintenance, after subsystem isolation and evacuation, the module comprising a failed component is removed and immediately replaced by a spare module. Subsequently, the failed component is replaced on the used module which then becomes the spare. Since spare modules and additional spare components intended for this type of maintenance must be carried onboard, there is a weight penalty. When all spare components (of one kind) have been used, a common component could be removed from spare modules on standby for other subsystems. However, this defeats the quick module replacement concept. Furthermore, module replacement depends on tight packaging which involves long design and development programs that result in high cost. For these reasons, dual action maintenance at the module level was rejected, despite the advantages of small package size and minimum maintenance time.

The approach indicated earlier in this paper, dual action maintenance at the component level, is more rewarding from a cost effectiveness point of view. When subsystem isolation and evacuation are accomplished, the failed components are removed from their stationary panels for replacement and/or repair. A spare component is installed if the subsystem downtime is critical, or the crew is occupied with other tasks. The subsystem is then refilled and returned to service. The failed component is repaired at a later time and put back into the spares bin. Even though it results in longer "off stream" maintenance times, this approach maximizes the advantages conferred by component and parts commonality. This effects a lower number of spares and a reduction in cost.

EVOLUTION OF THE MAINTAINABLE CO_2 REDUCTION SUBSYSTEM

The CO_2 Reduction Subsystem serves best to illustrate this evolution of packaging concepts. It also shows clearly the complex interrelationship of reliability, maintainability, commonality, sparing and packaging. In brief, it elucidates the impact of maintainability on EC/LSS design.

Figure 9 pictures the controls of the CO_2 Reduction Subsystem packaged as an integrated module for dual action maintenance. Common components, such as manual shutoff valves can be directly removed from the front panel. If the integrated module must be removed, fluid connections are automatically separated by releasing the structural support of the chassis.

Figure 10 illustrates the CO_2 Reduction Subsystem controls packaged for single action maintenance through R&R at the component level. Components are directly accessible either from the front, in the panel, or from the rear, in the chassis.

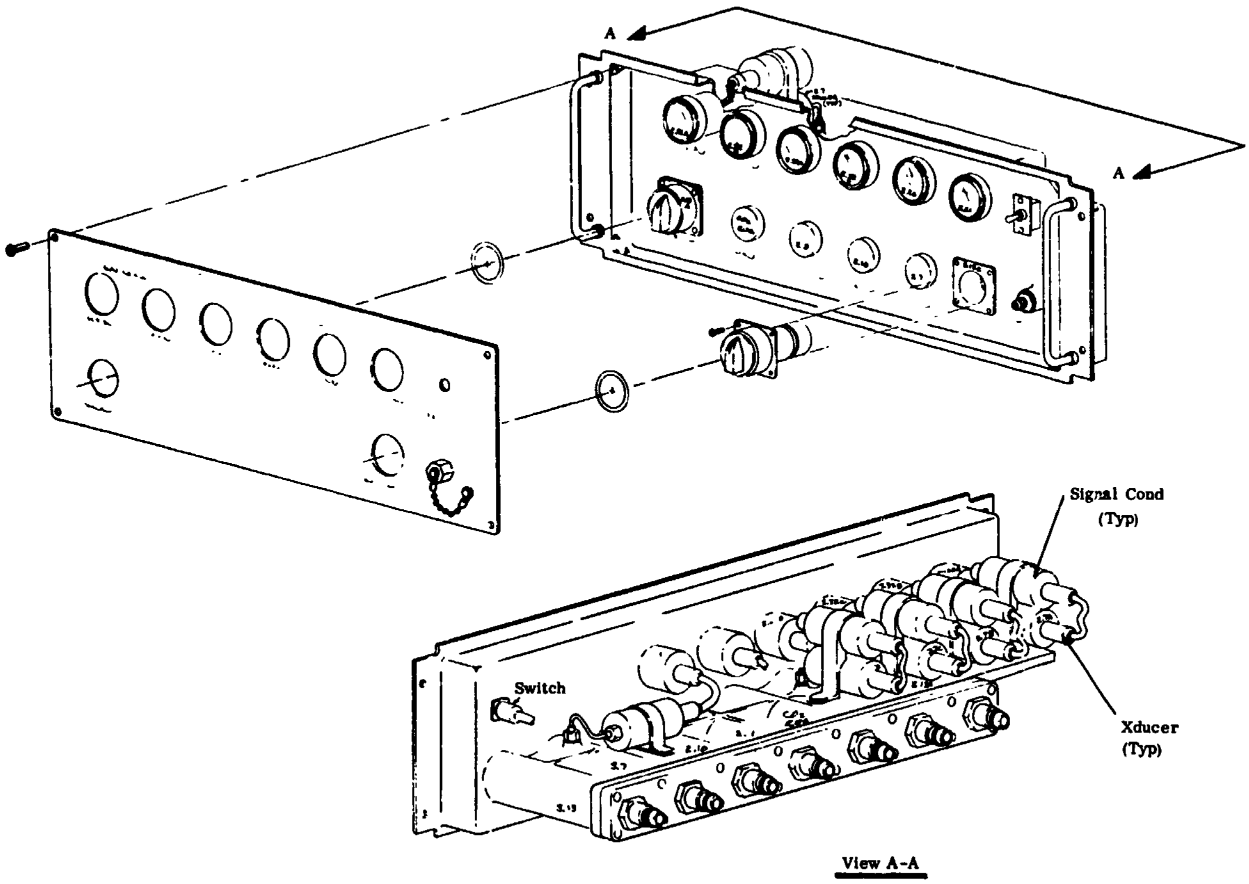


FIGURE 9 CO_2 REDUCTION SUBSYSTEM CONTROLS PACKAGED FOR DUAL ACTION MAINTENANCE

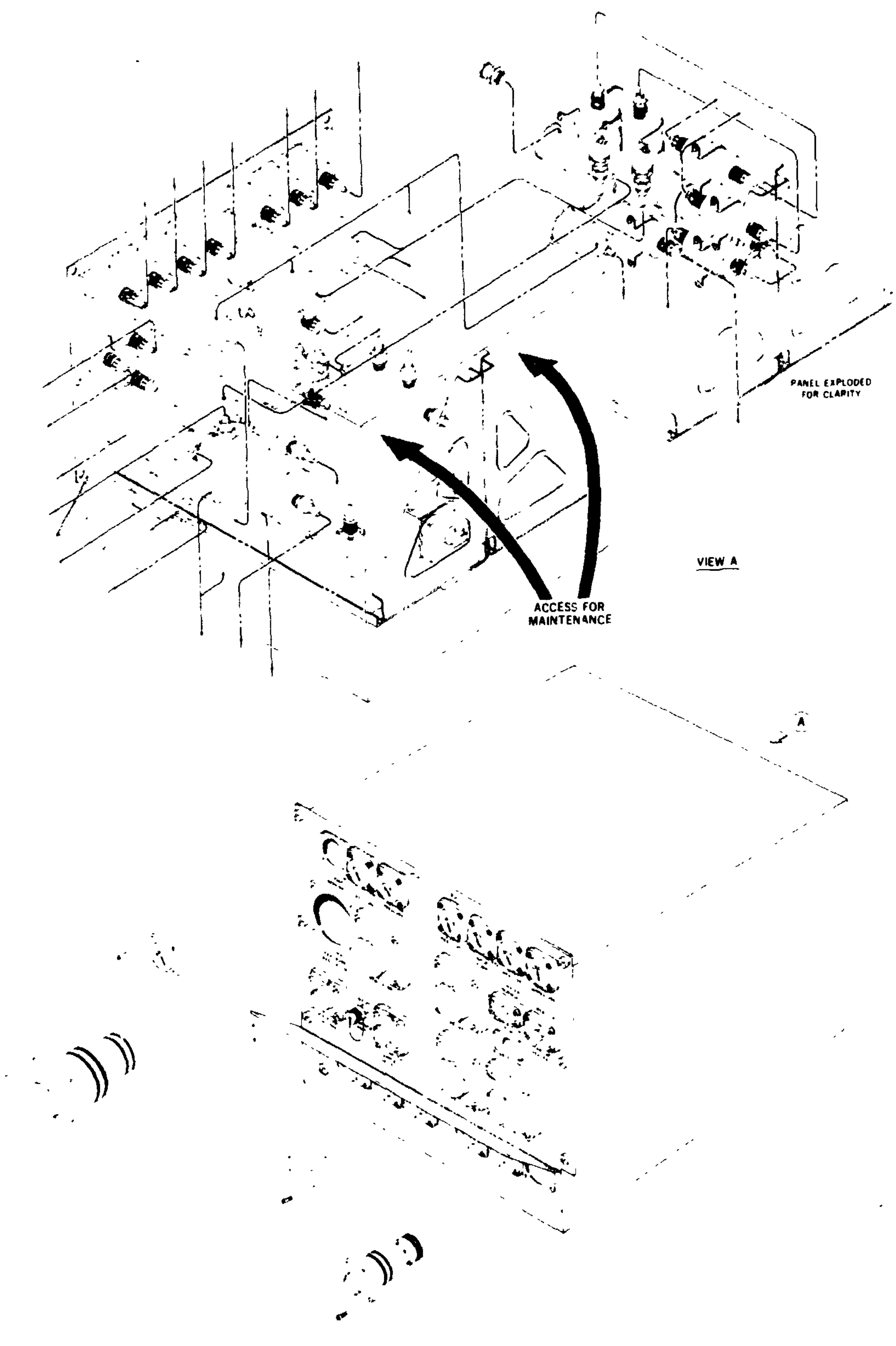


FIGURE 10 CO_2 REDUCTION SUBSYSTEM CONTROLS PACKAGED FOR SINGLE ACTION MAINTENANCE

For the discussion below, a basic understanding of the function of the CO_2 Reduction Subsystem in a Space Station EC/LSS is required. The core of the subsystem is the Sabatier reactor (Figure 11), which uses hydrogen as fuel with which to convert CO_2, expired by the crew, to water and methane. Temperatures average 600°F. With a number of potentially dangerous gases present, a number of controls and instruments are necessary for the safe operation of the reactor, all of whose components have relatively high failure rates.

A detailed reliability analysis was performed to back these components sufficiently with either redundant or spare components in order to achieve the required subsystem reliability goal of 0.995. These components were arranged in five different configurations corresponding to different system weights and various degrees of crew involvement. The Space Station Spares Selector Nomograph, Reference 3, was used as the principal tool to calculate spares requirements, and to perform

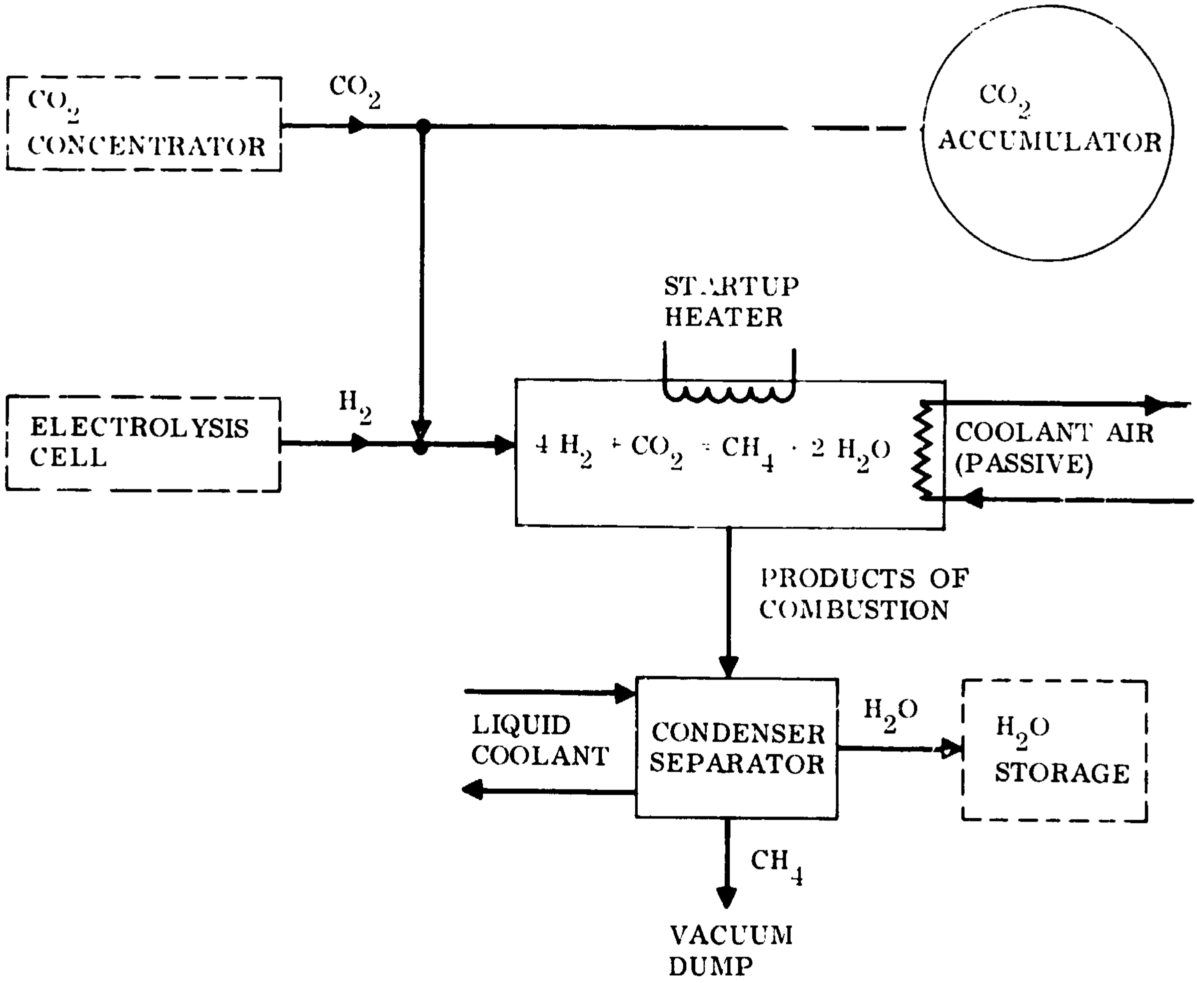


FIGURE 11 CO_2 REDUCTION SUBSYSTEM SIMPLIFIED SCHEMATIC tradeoff analyses for the alternative configurations on an orbital weight basis.

Configuration I of Table 3 comprises standby redundancy with manual switchover. This was used as a baseline (Figure 12). Automatic switchover was not considered because of the additional complexities introduced by the time dependencies of the detectors and switches required for automation.

Configuration IIA concentrates instrumentation and controls in a plug-in module for single action maintenance. Figure 13 explains this schematically. While this approach reduces downtime and crew skill requirements, it ranks highest in terms of orbital weight for the 5-year mission.

Configuration IIB obtains weight savings by adapting the module to dual action maintenance and by introducing commonality for valves, regulators and sensors.

Configuration IIIA is geared for single action maintenance at the component level, as can be

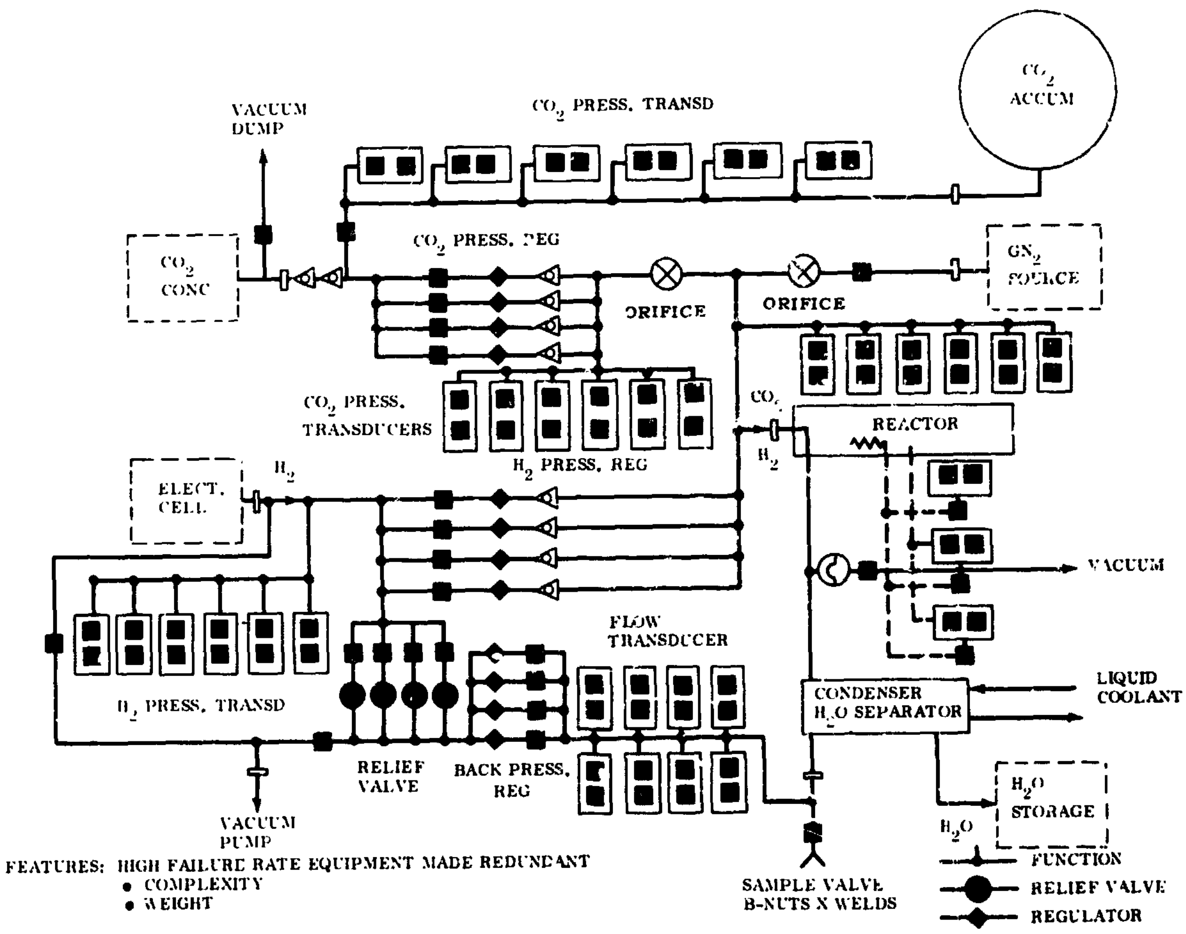
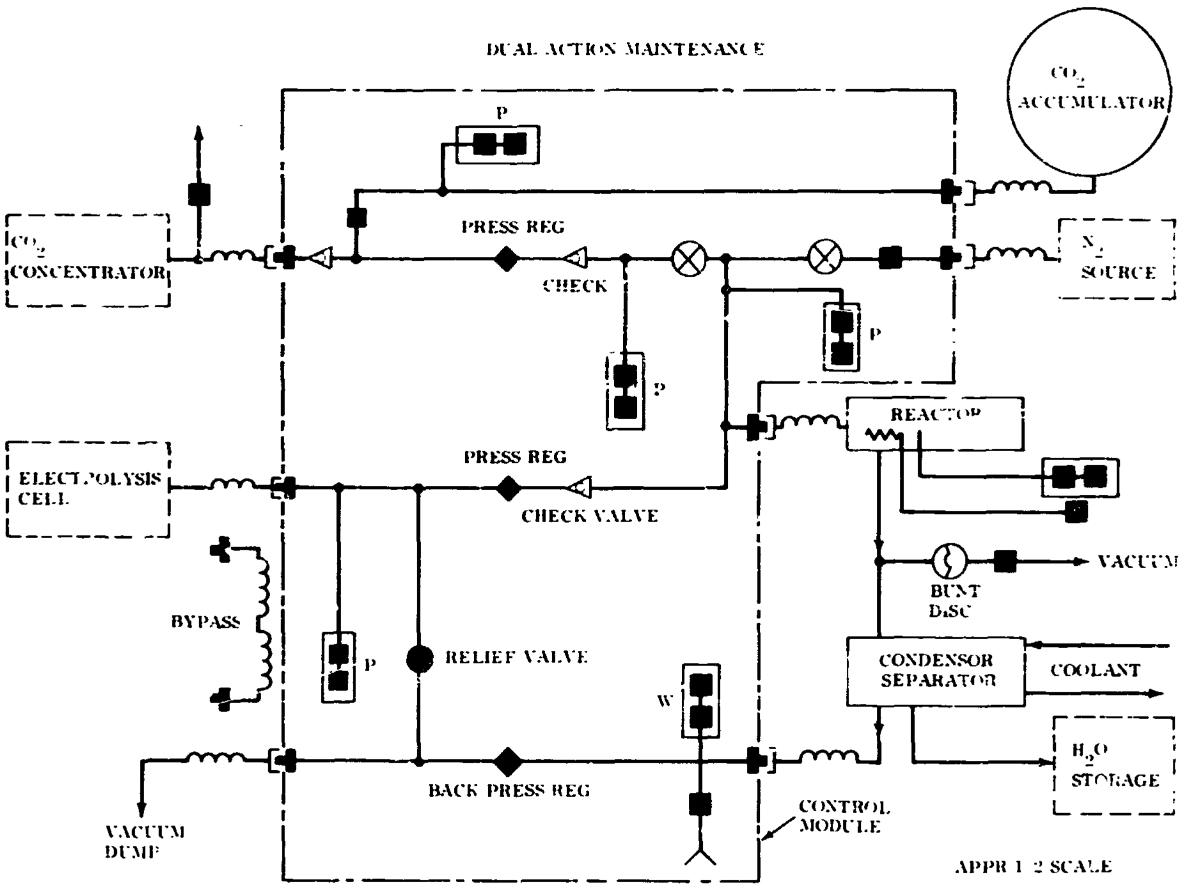


FIGURE 12 NON-MAINTAINABLE CO_2 REDUCTION SUBSYSTEM

seen from the schematic in Figure 14. This approach resulted in a sharp drop in the weight needed to be put into orbit. Specialized components are replaced with spares carried on-board and resupplied as needed.

Components in Configuration IIIB are made common wherever possible and spares are no longer earmarked for particular applications. Instead they are carried as part of a common spares mix for the CO_2 Reduction Subsystem; this permitted another reduction in weight.

Configuration IV points to further savings in spares weight for both on-board and resupply which can be realized by carrying the commonality concept to other subsystems and determining a spares mix on an integrated system basis. For example, the shutoff valves and regulators mounted in



FEATURES. MODULAR APPROACH • FAULT DETECT MODULE INTERFACES • SELF SEALING QUICK DISCONNECTS USED FOR EASE OF FAULT ISOLATION & COMPONENT REPLACEMENT • SIMPLICITY

FIGURE 13 MAINTAINABLE CO_2 REDUCTION SUBSYSTEM

the CO_2 Reduction Subsystem and spared on a subsystem basis, can be applied to the Cabin Pressure Control Subsystem. These can then be spared on a system-wide basis and an additional savings in weight can be allowed to the CO_2 Reduction Subsystem. This would be accomplished by comparing the different subsystems involved on a failure-weight basis. The ultimate in weight reduction can be achieved by taking commonality in design and maintenance right down to the component-part level. Configuration V calls for dual action maintenance by first replacing the failed component, and secondly, by replacing the defective part within the component. In case of the previously discussed common valves and regulators, this may be the replacement of a worn O-ring seal. A numerical illustration of this configuration was not performed.

Table 3 delineates the demands made on the crew in maintaining the different configurations. Table 4 summarizes the orbital weights required for the different configurations.

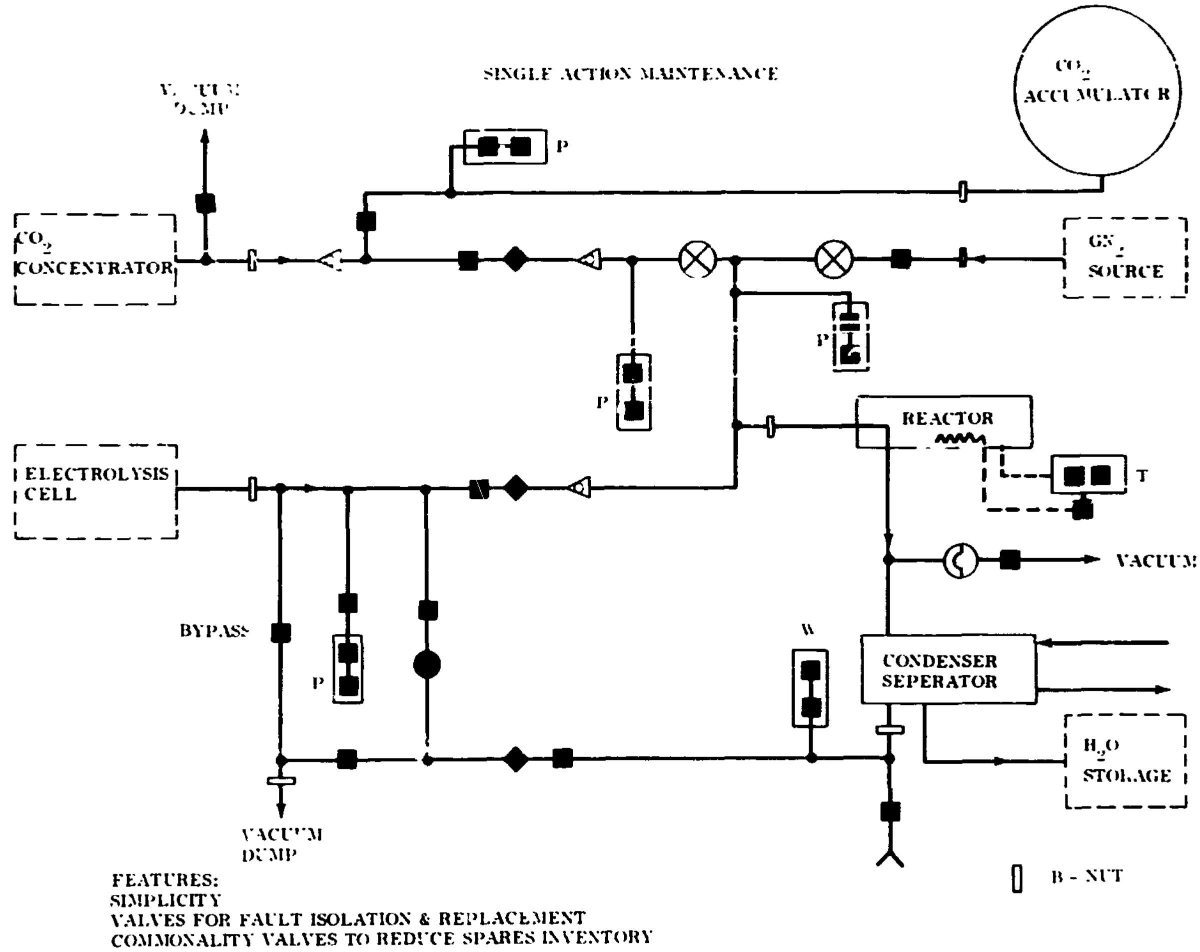


FIGURE 14 MAINTAINABLE CO_2 REDUCTION SUBSYSTEM

TABLE 3. CREW PARTICIPATION IN EC/LSS MAINTENANCE

Configuration	Type of Package	Maintenance Action	Maintenance Level	Crew Tasks	Tool Requirements	Crew Requirements
I	Conventional, chassis	None	None	Identify failed component, perform switchover	None	No special skill, minimum time
IIA	Plug-in module with sealed components	Single	Module	Identify failed module, replace module, order new module	None	Low skill, little time
IIB	Plug-in module with replaceable components of common designs	Dual	Module and Component	Identify failed component, make decision to replace module or component, repair module, adjust aneroid, order new module or component	Simple tools for component removal, test bench for adjusting aneroids	Moderate skill, increased time
IIIA	Panel with replaceable components of individual designs	Single	Component	Identify failed component, replace component, order new component	Simple tools for component removal	As above
IIIB	Panel with replaceable components of common designs	Single	Component	Identify failed component, replace component, adjust aneroid, order new component	Simple tools for component removal, test bench for adjusting aneroids	As above
IV	Panel with replaceable components of common designs, spared on an integrated EC/LSS basis	Single	Component	As above	As above	As above
V	As above, but sparing on component-part basis	Dual	Component and component-part	Identify failed component or component part, make decision, replace or repair, order replacements as req'd	Special tools, fixtures, test bench for adjustment and checkout	High skill, considerable time

TABLE 4. ORBITAL WEIGHT OF CO_2 REDUCTION SUBSYSTEM CONTROLS - STUDY RESULTS

	Configuration Description	Equipment Installation			Onboard Spare Supply			Resupplied Spares Mix, Weight	Total Weight To Orbit
		Equipment Types	Number of Applications	Weight lbs.	Equipment Types	Number of Applications	Weight lbs.	Lbs.	Lbs
I	Standby redundancy with manual switch-over	15	98	34	12	111	78	0	112
IIA	Plug-in module with single action maintenance	17	38	23	1	3	65	114	205
IIB	Plug-in module with dual action maintenance	14	38	23	2	6	57	76	156
IIIA	Insitu repair at component level special designs	16	51	26	13	24	17	3	48
IIIB	Insitu repair at component level common designs	12	51	28	10	19	11	3	42
IV	Common designs throughout EC/LSS to minimize spares	12	51	28	10	>19	<11	<3	<42

Constraints:

4 - man crew for space station simulator

5 - year mission, or 43,800 hrs.

90 - day resupply (except for configuration I which has no resupply), or 2,190 hrs.

0.950 reliability goal for entire EC/LSS

10 subsystems with uniform subsystem reliability goals of 0.997 (0.995 for Configuration 1)

Control section is estimated to represent 1/3 of the CO_2 Reduction Subsystem with a section reliability goal of 0.99992 (0.998 for Configuration 1)

CONCLUSION

While a strong case has been built for the maintainable EC/LSS with dual action maintenance, due caution must be exercised in the absence of actual operating experience with such a system. At present, only limited data are available with which to determine the potential for man-caused damage in operating and maintaining regenerative type EC/LSS for a long duration mission. Consequently, no failure rates could be assigned to the crew that performs the maintenance tasks discussed above.

What is the probability of error on the part of the crew? What is the quantification of these errors? EC/LSS designers will have to look towards the testing of representative maintainable designs in manned chambers. Manned tests of 90-day duration, or more, will be required where, just as for on-board space stations, the crew will be called upon to offset in-flight failures of vital equipment. Tests in the past have been designed to test the crew's ability to endure confinement and measure bio-medical parameters. These new tests will have to include the introduction of artificial random failures by cognizant subsystem engineers located outside the chamber. These chamber experiments need to be augmented by underwater experiments, and experiments on-board AAP and MOL flights, to verify equipment and procedures under conditions of reduced gravity.

It is conceivable that weight and cost considerations will give way to relief of the crew in stress situations. The realities of permissible down times and available crew skills may call for single action maintenance at the module level. For some subsystems, it may even be necessary to institute redundancy with automatic switchover.

The stipulation of a 5-year mission, without resupply, may have been an extreme design goal, but it was most helpful in developing the great potential of maintainability for EC/LSS. The significance of these studies extends beyond EC/LSS design. It is hoped that the new maintainability concepts being developed will become trendsetters for a number of ancillary fluid-mechanical systems on-board space stations. These systems include cryogenic fluid and propellant transfer, reaction control, auxiliary power, etc. Proliferation of these concepts should greatly enhance overall space station reliability and pave the way to i rplanetary travel.

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A SUMMARY OF HAND-HELD MANEUVERING UNIT DEVELOPMENT

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SUMMARY: This paper is a discussion of the background and human engineering requirements that have led to the present Hand-Held Maneuvering Unit (HHMU) configuration, and of the design and performance of an advanced binary hydrazine monopropellant-fueled HHMU that provides improved propulsive performance.

INTRODUCTION

During the Gemini IV Mission, Pilot Ed White accomplished the first propulsive extravehicular-activity (EVA) maneuvering in history with a Hand-Held Maneuvering Unit (HHMU). Following the flight, the pilot narrated films of the EVA and made the following comments on the HHMU evaluation.

"What I tried to do is actually fly with the gun, or maneuver with the gun, right out of the spacecraft, and when I departed . . . , there was no push off whatsoever from the spacecraft. The gun actually provided the impulse for me to leave the spacecraft. . . . I maneuvered approximately down the centerline of the spacecraft, perhaps favoring a little on the right. The gun is actually providing the impulse for my maneuvers. I started a yaw around to the left with the gun. At this time, I knew we had something with the gun - because, it was actually providing me with the opportunity to control myself where I wanted to go up there. The control was actually what we were trying to demonstrate on our EVA operation. We knew a little about the tether dynamics but we wanted to actually find out how well a man outside a spacecraft with a maneuvering unit could control himself and, in later parts, we wanted to find out just how well a man could control himself with a tether. . . . I came back towards the spacecraft, up over the hatch, turned around above the spacecraft out of the view of the cameras and at this time, I told Jim I was coming back again out in front of the spacecraft to see if we couldn't be sure we could record it on the cameras. . . . I went out in front of the spacecraft at this time and actually made a yaw in either direction. I found that control with the gun to the right and to the left was what I felt quite adequate and the pitch was quite adequate. I only had 6 ft/sec (total available velocity increment) in the gun, which is a very limited amount of air, so I tried to use it very sparingly. I just used it enough to satisfy myself and to make maneuvers so that I felt in my own mind that I could control myself in pitch, yaw, and translation. This is the type of control that you need to move from point A to point B in space. . . . I wasn't trying to control myself in roll because we don't really care about the roll as long as the pointing direction is accurate."

Following the flight of Gemini IV, the HHMU was scheduled for use on Gemini Missions VIII, X, and XI. Because of problems with other systems, the planned HHMU evaluations were not completed. The HHMU was used successfully by the Gemini X pilot Mike Collins prior to the scheduled evaluation, when he "fell off the Agena" and used the gun to translate back to the spacecraft, and was used again on his return trip to the Agena.

The limited flight use of the HHMU has indicated its usefulness and has justified continued development of the units. Proposed plans include an HHMU evaluation during the Apollo Applications Program (AAP) M-509 Maneuvering Unit Experiment. In addition, a hydrazine/water-fueled HHMU is being perfected so that it will be available for future EVA missions.

The following summary of HHMU development may be of interest to people engaged in the design of equipment that has a pressure-suited human interface and to people interested in operational requirements and the technology being introduced by the development of the Hydrazine Hand-Held Maneuvering Unit (HHHMU or $H^3\mu$).

HUMAN ENGINEERING

Human engineering in relation to pressure-suit and glove mobility has had an important effect on the development of HHMU configurations. To appreciate fully the problems encountered in designing equipment that will be used by a pressure-suited subject, a brief description of a pressure suit is provided.

Pressure suits are usually categorized as soft suits or hard suits. A soft suit is basically a flexible bladder, usually of rubber-coated fabric, which when inflated, expands until it is restrained by an outer or restraint layer. A hard suit is a rigid shell and must be jointed to provide mobility. A soft suit with no joints can be flexed, but only with some effort to overcome the neutral point of the restraint layer.

The Gemini suits were designed with the neutral point in a sitting position. The arms of the suit were positioned for optimum access to the Gemini flight controls. Whenever a crew member moved within the pressurized suit, he had to overcome the forces tending to return the suit to its neutral position.

Apollo space suits are designed to give greater freedom of movement, since the wearer must be able to walk and perform other motions required for exploring the lunar surface. The Apollo space suit uses bladder and restraint-layer sections connected at the joints by flexible convoluted sections. Extravehicular-activity suits have a cover layer to provide thermal and micrometeoroid protection which tends to increase bulkiness and decrease mobility.

It became evident with early HHMU models (Figs. 1 and 2) that a pistol-type grip works fine on a gun fired at eye

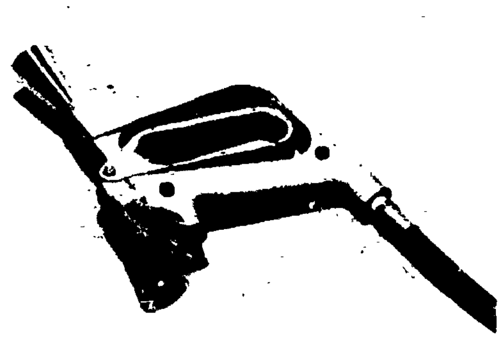

Fig. 1. Developmental Model With Push-Pull Trigger Configuration

Fig. 2. Development Model With Thumb-Actuated Direction Control Valve

Fig. 3. Developmental Model with Adjustable Nozzle Angles

level, but, if fired from the hip or through the center of gravity of the body, the wrist is bent into an unneutral Gemini-space-suit position. A model was made that incorporated adjustable nozzle angles (Fig. 3). The angular relationships of nozzle to grip centerline required to place the thrust through the center of gravity with the right hand in a neutral suit position were determined in experiments with the adjustable-nozzle-angle model. With the nozzles positioned as shown in Figure 4, the grip centerline is located by pitching up 30°, yawing left 15°, and rolling clockwise 10°. This angular relationship has been maintained on Hand Held Maneuvering Units to permit interface with Apollo space suits, even though the suits have improved wrist joints, because the angular relationship allows equal displacement for control motions from the neutral position of the wrist.

Experiments with the adjustable-nozzle-angle model (Fig. 3) also demonstrated that parallel tractor nozzles placed far apart produced much lower thrust losses from gas impingement than nozzles placed side by side and canted outward produced. The proper stance when translating with the HHMU is for the body to be turned to the direction of translation at an angle that will present the least frontal area to gas impingement.

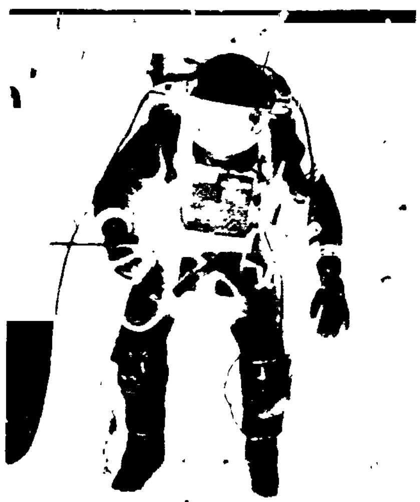

Fig. 4. Gemini VIII EVA Configuration

Current versions of the HHMU are shorter than Gemini models, and in order to minimize stowage envelope, it was necessary to compromise by shortening the nozzle arms and canting the nozzles outward slightly (Fig. 5).

Pressure-suit gloves present another challenge in the design of interfacing equipment. Gloves for the Gemini mission were made with a neutral position that provided ease in handling flight controls. Holding the hands in other positions for any period of time brought about varying degrees of fatigue. The Gemini EVA gloves were designed with a neutral position that would allow easier manipulation of the HHMU and other EVA equipment. Extravehicular-activity gloves have cover layers to provide thermal and micrometeoroid protection and to reduce conductive heat transfer from the spacecraft or equipment surfaces. The cover layer adds bulk and reduced mobility. Both Gemini and current Apollo gloves have hinged metallic members in the palms, which allow cupping of the palm and prevent the gloves from ballooning in the palm area.

Fig. 5. HHMU Being Developed for AAP M-509 Experiment

Several designs were evaluated in early HHMU models to find a control-valve layout that could be manipulated easily with EVA gloves. The first design (Fig. 1) provided tractor thrust when the hand was closed from the glove neutral position and pusher thrust when the hand was opened from the glove neutral position. This design required that hands be bare because the thumb was used to grip the handle and oppose the force required in order to push the trigger forward for pusher thrust. With pressurized gloves, it was fatiguing to hold the thumb in the position to grip the handle. Strapping the gun to the wrist of the suit allowed adequate control, but this measure had obvious disadvantages.

In another early HHMU design (Fig. 2) a thumb-operated lever was used to switch from a tractor to pusher mode, but this action was also too difficult to allow the use of pressurized gloves.

Since finger and thumb dexterity was limited with pressurized gloves, it was decided that gross motions of the hand would be relied upon to initiate tractor and pusher modes. This approach worked well enough and was used on all Hand-Held Maneuvering Units developed for the Gemini Program. Tractor and pusher valves were actuated by sliding the glove forward or aft and depressing the trigger that was hinged at the center (Figs. 6 and 7). In Gemini X and XI Hand-Held Maneuvering Units, the trigger was redesigned to provide two triggers pivoted at the ends (Fig. 8). This change permitted tractor and pusher valve actuation with less movement of the hand.

Post-Gemini development has resulted in a more acceptable control-valve layout. This configuration (Fig. 5) provides a single flow control valve, operated by depressing a button with the index finger. For tractor mode, only the trigger is actuated. For pusher mode, both the trigger and the button are actuated. This design is being used in the AAP M-509 experiment HHMU and in the HHHMU.

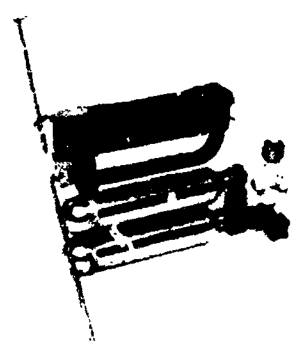

Fig. 6. Gemini IV HHMU With Self-Contained Propellant Supply'

The size and shape of HHMU grips varies with the size and strength of the hand and with the personal preference of the user. Gemini HHMU grips were modified several times to accommodate the astronauts using them. One universal feature incorporated in the grips was to make relief grooves in the grip to mate with the palm restraint wires in the glove in order to eliminate pressure points on the hand. Current Hand-Held Maneuvering Units have a removable grip that will allow custom fitting, if necessary.

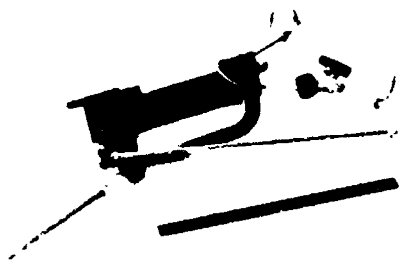

Fig. 7. Gemini VIII HHMU and Backpack Supply Line Coupling and Shut-off Valve

Fig. 8. Gemini X and XI HHMU With Umbilical Coupling and Shut-off Valve

Two coupling designs were used on Gemini Hand-Held Maneuvering Units. A simple unvalved screw-on connection was used on the Gemini IV unit and was made in the pressurized cabin. The same connection was used on the Gemini VIII backpack HHMU, since the connection was made prior to launch. This configuration incorporated an emergency shut-off valve upstream of the connection, which could be used to terminate flow to the gun in case of a control-valve failure. The coupling was keyed to keep the emergency shut-off valve within reach of the astronaut's left hand.

On missions for which the HHMU would be coupled to its gas supply line during EVA, a push-to-connect valve coupling was used. This coupling was also keyed to prevent emergency-shut-off valve rotation. The coupling section with the least connection force in the pressurized condition was placed on the supply line, since any residual pressure in the HHMU could easily be dumped.

OPERATIONAL REQUIREMENTS

Probably the least challenged advantage of the HHMU from an operational standpoint is its compactness. This compactness was especially important on the Gemini spacecraft, on which stowage room was at a premium. With the tractor arms folded, the envelope of the Gemini Hand Held Maneuvering Units, including coupling and shut-off valve, was approximately 16.5 by 6 by 2 inches. The AAP M-509 HHMU measures 11.5 by 5.5 by 2 inches.

The Gemini IV HHMU was used with a self-contained propellant supply. Oxygen was used as the propellant gas to prevent dilution of the cabin atmosphere in case of propellant leakage. Two 4000-psig bottles were connected to a manifold that ported the gas to a shut-off valve. A regulator downstream of the shutoff valve delivered oxygen at a pressure of 110 psia.

and at a rate of 2 lb/min. This assembly was stowed with, but disconnected from, the HHMU in a stowage box measuring 18 by 6 by 4.5 inches.

On the Gemini VIII Mission, the HHMU propellant was supplied from a backpack. The pack contained two tanks. One tank supplied oxygen to the astronaut, and the other tank supplied Freon-14 to the HHMU.

On the Gemini X and XI Missions, nitrogen was supplied to the HHMU through umbilicals. The umbilicals also supplied oxygen to the astronaut and provided communication and biomedical instrumentation lines. The nitrogen was routed through aluminum tubing from a tank in the spacecraft adapter section to a recessed panel behind the hatch. The tubing was clamped to the spacecraft at numerous points to provide heat shorts into the structure for warming the gas, which was cooled during use as a result of adiabatic expansion. A quick-disconnect coupling and a shutoff valve were provided in the recessed panel for connecting the nitrogen line in the umbilical to the nitrogen supply.

A summary of configuration and performance information is given in Table I. As shown in Table I, the cold-gas Hand-Held Maneuvering Units have a variable thrust of 0 to 2 lb. Airbearing training and flight experience to date have indicated that a 2-lb thrust level is adequate for HHMU maneuvering. The control valves in the Gemini Hand-Held Maneuvering Units were designed to be partially pressure balanced. A slight differential area being acted on by gas pressure tended to cause the valve to be closed. Additional closing force was provided by a spring, and an increase in triggering force was provided by the spring as the valve was opened. This arrangement allowed control of thrust level, as well as of duration, and proved to be useful for slow but precise maneuvering. This metering capability has been improved in the AAP M-509 HHMU by reshaping the control-valve poppet.

Freon-14 was used as a propellant for the Gemini VIII HHMU to increase total astronaut change-in-velocity capability. The specific impulse of Freon-14 is lower than the specific impulse of nitrogen but the greater density of Freon-14 provided a net gain in total impulse with a relatively small percentage gain in total system weight. The expansion of the Freon-14 from 5000 to 110 psi resulted in temperatures as low as -150°F in the HHMU. Teflon cryogenic seals were used in the HHMU control valves and emergency shut-off valve to permit operation at the low temperatures.

Nitrogen was used as a propellant for the Gemini X and XI Hand-Held Maneuvering Units. Adequate total impulse was available from the storage system mounted in the adapter section of the spacecraft, and the temperature resulting from expansion was high enough to allow the use of conventional fluid-system components.

Like the Gemini IV HHMU, the AAP M-509 HHMU will use oxygen to prevent dilution of the cabin atmosphere.

The use of cold gases for HHMU propellant during the Gemini EVA Program permitted rapid development of flight-ready hardware and rapid changes of configuration to meet new requirements as they evolved, but did not represent a high degree of efficiency because of the inherent low specific impulses available with cold gases.

In order to develop a more efficient HHMU for the future, Rocket Research Corporation, under contract to the NASA Manned Spacecraft Center, is developing a binary hydrazine monopropellant-fueled HHMU.

HYDRAZINE HAND-HELD MANEUVERING UNIT

The Hydrazine Hand-Held Maneuvering Unit (HHHMU or $H^3\mu$) is directed at improving the propulsion efficiency of the

HHMU PERFORMANCE AND CONFIGURATION SUMMARY

TABLE I

Performance Characteristic or Configuration	Gemini IV	Gemini VIII	Gemini X	Gemini XI	AAF M-509 Experiment
Propellant	O_2	Freon-14	N_2	N_2	O_2
Specific Impulse, sec	59	33.4	63	63	59
Propellant Weight, lb	0.68	18	10.75	10.75	NA*
Total Impulse, lb-sec	40	600	677	677	NA*
Thrust, lb	0 to 2	0 to 2	0 to 2	0 to 2	0 to 2
Total Available Velocity Increment, ft/sec	6	54	34	84	NA*
Trigger Preload, lb	15	15	5	5	5
Trigger Force at Maximum Thrust, lb	20	20	8	8	8
Storage-Tank Pressure, psi	4000	5000	5000	5000	900 (cryo supplied)
Regulated Pressure, psi	120	110	125	125	120
Nozzle-Area Ratio	50:1	50:1	50:1	50:1	4:1
HHMU Weight, lb	3.0	3.0	3.0	3.0	2.1
Propellant Supply System Weight, lb	3.8	NA*	NA*	NA*	NA*
Umbilical Length, ft	---	---	50	30	35

*NA - Not applicable

HHMU through incorporation of higher specific impulse and higher density liquid monopropellant fuel. Under NASA Manned Spacecraft Center Contract NAS 9-5617, Rocket Research Corporation performed analytical and experimental studies of catalytic decomposition reactor assemblies utilizing 51 percent hydrazine/49 percent water. This program, the first of two (2) contracts, was directed at development of a hydrazine-based fuel/thrust chamber assembly (TCA) which satisfied the following criteria: A unit (a) capable of producing a maximum allowable exhaust gas stagnation temperature of 500°F; and (b) capable of a system storage temperature of -60°F. This program had three (3) major objectives, as follows: (a) to identify and fully characterize a suitable propellant blend; (b) to develop a catalytic reactor utilizing Shell 405 catalyst to decompose the selected mixture; and (c) to demonstrate safety margins which would permit the TCA/propellant mixture to be considered for manned applications.

To supplement and substantiate safety characteristics and physical properties of the hydrazine/water propellant, extensive safety and characterization testing of the 51 percent N_2H_4/49 percent H_2O propellant was undertaken under Contract NAS 9-5617.

The propellant mix was subjected to standard CPIA liquid propellant card gap, drop weight, and thermal stability tests, as well as to ASTM standard flash and fire point tests. The propellant mix demonstrated stability and safety characteristics slightly superior to anhydrous hydrazine and hydrogen peroxide monopropellant, as depicted in Table II. Another index of propellant safety is storage stability. In the case of a monopropellant which can be catalytically decomposed, this can be a serious limitation. For this reason, extensive storage stability tests of up to 60 days were conducted at temperatures ranging from approximately 70°F to 160°F. These tests have shown that 6AL 4V titanium, 300 series stainless steels, teflon, and Inconel X, promote essentially no decomposition (incompatibility) with the propellant mix at temperatures up to 160°F. This excellent storage thermal stability is further exemplified when compared to hydrogen peroxide which undergoes decomposition with essentially all materials of construction, even at temperatures as low as 70°F. Currently, storage tests are still in progress at ambient temperatures, and after 40 months there is no sign of decomposition. An additional safety feature of the 51/49 mix noted during the course of the extensive firing and propellant handling program is the fact that a propellant spill will not result in a fire or any apparent combustion hazard. In summary, based on tests conducted during Contract NAS 9-5617 and extensive background data on hydrazine and hydrazine-based propellants, it is concluded that the 51 percent N_2H_4/49 percent H_2O propellant mix is an extremely safe monopropellant, exhibiting stability, handling, toxicity, and storage properties equal to hydrogen peroxide in all respects. Additionally, the mix has superior freezing-point and low exhaust-gas-temperature characteristics.

Thrust chamber assemblies at a nominal one (1) pound and two (2) pound thrust level were designed, fabricated, and evaluated to arrive at a baseline configuration for the 51 percent N_2H_4/49 percent H_2O propellant. Figure 9 illustrates the design which includes a 25-micron absolute multiple disc filter integrated into an in-line relief valve body and a catalytic reactor. The in-line relief valve cracks at a fluid pressure of 15 ± 0.75 psi and reseals (zero leakage) at 10 psi to provide positive propellant shut-off at the injector inlet. This valve is actuated by depressing the HHMU trigger, which increases the fluid pressure at the valve, causing the valve to open. This in-line relief valve serves as a means of improving start and tailoff response by providing a mechanical shutoff near the reactor, thereby eliminating prolonged tailoff and fill times, such as would be experienced if total shutoff occurred in the HHMU trigger assembly.

SELECTED PROPERTIES - MONOPROPELLANTS

TABLE II

	H_2O_2 (90%)	H_2O_2 (98%)	N_2H_4	51% N_2H_4/49% H_2O
Specific Impulse (Theo. at P_c = 300 psia and P_c = 14.7 psia), lbf-sec/lbm	132*	140*	188**	103***
Decomposition Temperature (T_c) at P_c = 300 psia, °F	1,392*	1,825*	1,820**	500***
Vapor Pressure at 68°F, psia	0.006	0.04	0.205	0.195
Density, gm/cc at 68°F	1.15	1.44	1.01	1.03
Freezing Point, °F	+11	+31	+34	-77
Viscosity at 70°F, centipoise	1.15	1.15	0.96	1.95
Shock Sensitivity, Number of Cards	0	0	0	0
Impact Sensitivity, Kg-cm	>120	>120	>120	>120
Thermal Stability (temperature at onset of rapid decomposition), °F	-	Ca 230	Ca 400	Ca 450
Flash Point/Fire Point, °F/°F	-	-	126/126	208/221
Toxicity (threshold limit), ppm	7	1	1	1

* H_2O_2 Feed Temperature = 65°F
** Ammonia Dissociation Ca 50%
*** Ammonia Dissociation Ca 0%

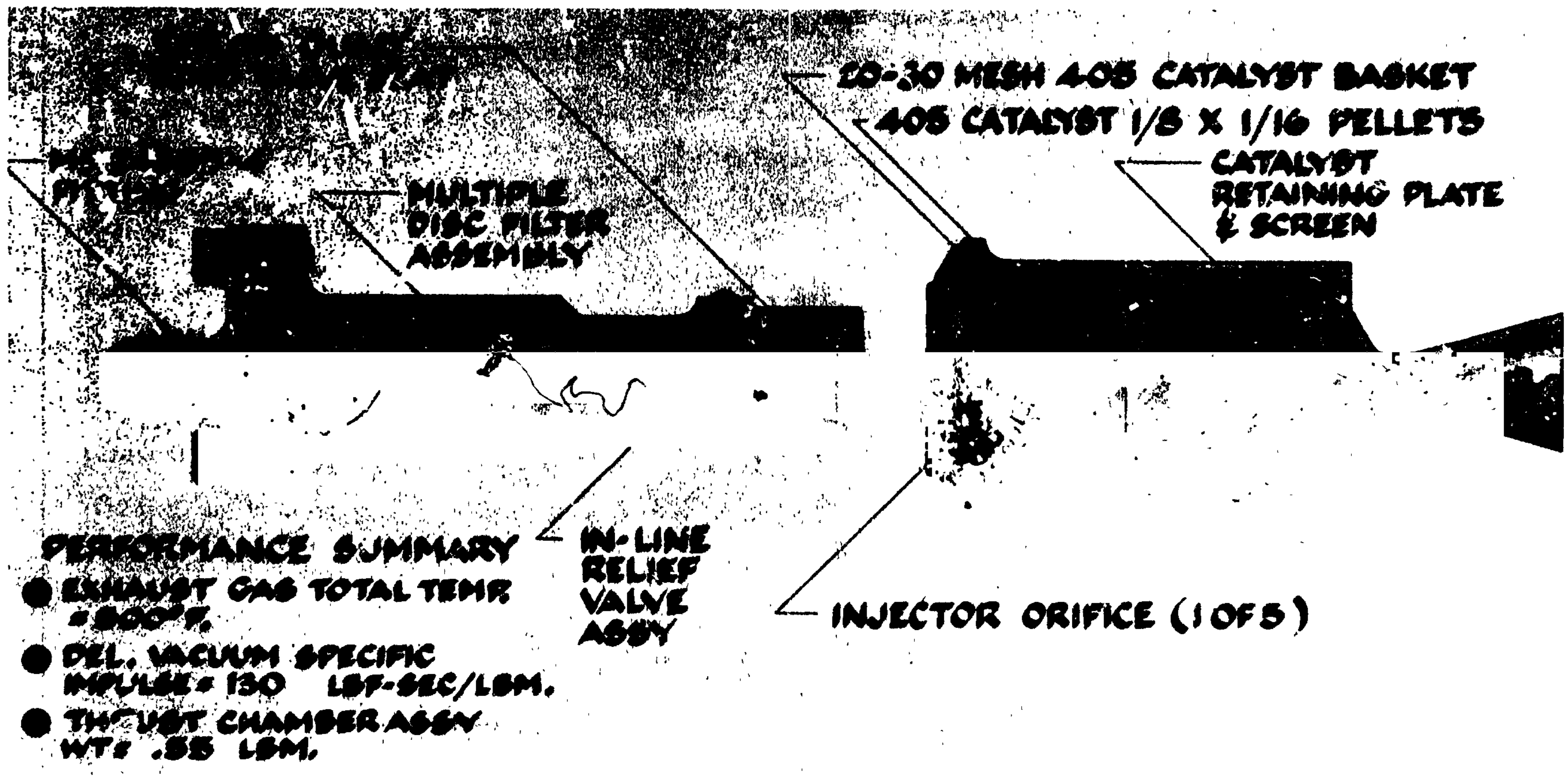

Fig. 9. 1 lbf Thrust Chamber Assembly

As a result of the low stagnation gas temperature, 321 stainless steel is utilized throughout the unit for materials of construction. The reactor employs an upper bed of 0.3 inch of 20-30 mesh Shell 405 catalyst and a 1.6-inch 1/8 by 1/16 inch pelletized lower bed. A bedloading of 0.009 lb_m/in^2-sec at a 70-psia chamber pressure produced a characteristic exhaust velocity of 2260 ft/sec and an exhaust gas gemperature of 487°F at a 70°F propellant inlet temperature. Decomposition occurs at zero percent ammonia dissociation, resulting in nitrogen, ammonia, and superheated water vapor exhaust products.

The injector is a five (5) element showerhead with 325 mesh stainless screens upstream of the injection orifice inlets to preclude migration of catalyst fines into the feed passages during the launch vibration period. The TCA was not expected to operate at expansion ratios beyond 5:1 because of nozzle recondensation phenomena. However, altitude tests at ϵ = 30:1 produced no evidence of homogeneous condensation, and vacuum specific impulses greater than 130 seconds were measured at altitudes in excess of 100000 feet.

The TCA surpassed its design requirements by an ample margin and performed satisfactorily in a series of overstress tests to evaluate safety aspects. Table III summarizes these tests. During the course of Contract NAS 9-5617, 62000 seconds of burntime and 12000 reactor on/off cycles were accrued with the 51 percent N_2H_4/49 percent H_2O propellant mix. Overstress testing verified design integrity in areas of thermal design, low temperature ignition capability, and high/low chamber pressure tests.

Based upon the results of Contract NAS 9-5617, the second phase of the HHHMU program (Contract NAS 9-6909) was initiated, wherein the 51 percent N_2H_4/49 percent H_2O propellant, the 1 lb_f and 2 lb_f TCA's design criteria, the results from the material compatibility study, and $H^3\mu$ system level sizing analyses were integrated with the NASA Manned Spacecraft Center AAP M-509 HHMU configuration to evolve a Handle/Thruster Module H/TM) design. A modularized quick-change Propellant Tank Module (PTM) was configured to mate with the H/TM to form the HHHMU, shown in mock-up form in Figure 10.

Fig. 10. Hydrazine Hand-Held Maneuvering Unit Mockup

While incorporating many of the design features of the HHMU, the $H^3\mu$ is a new unit designed to incorporate the 51 percent N_2H_4/49 percent H_2O liquid monopropellant. In addition to utilizing new materials of construction, two (2) 1.4 lb_f thrust tractor and one (1) 2.8 lb_f pusher thrust chamber assemblies, utilizing Shell 405 spontaneous catalyst, have been incorporated into the handle assembly to permit decomposition of the low freezing point (-77°F)/low exhaust gas temperature (500°F) propellant. As illustrated in the system schematic in Figure 11, liquid propellant flow control is accomplished in the handle with a manual on-off throttle valve and a manual shuttle valve, which directs the propellant flow from the tractor engines (primary engines) to the pusher engine. As discussed previously, an in-line relief valve is incorporated in each thrust chamber assembly to provide positive propellant shutoff near the reactor.

Propellant is supplied to the handle and thrusters (H/TM) from a quick-change PTM. Each PTM contains a nominal 250 lb_f-sec of propellant, which is fed through a

PROPELLANT/THRUST CHAMBER ASSEMBLY PERFORMANCE SUMMARY

TABLE III

Objective	Achievement
500°F Exhaust Gas Temperature	500°F Exhaust Gas Temperature
System Storage Temperature, -60°F	51% N_2H_4/49% H_2O; Freezing Point, -77°F
3000 lb_f-sec TCA Impulse Capability	8000 lb_f-sec Demonstrated
1000 Pulses per TCA	3000 Pulses Demonstrated
Thrust Response to 90% - 100 milliseconds	150 milliseconds (Warm Bed Conditions)
Random Vibration, 8.8 g rms; 20-2000 cps Profile	Vibration Resistance Excellent
117 seconds Vacuum Specific Impulse	130 seconds Demonstrated
Propellant Safety and Characterization	Propellant Characterized and Safety Verified
TCA Safety	62000 seconds and 12000 Reactor On/Off Cycles
Overstress Tests	
Verify TCA Performance at 0.3 Nominal Chamber Pressure ~ 20 psia	Excellent Performance; 1044 seconds, 1060 Pulses
Verify TCA Performance at 1.5 Nominal Chamber Pressure ~ 90 psia	Excellent Performance; 1044 seconds, 1060 Pulses
Verify TCA Thermal Margin with 1150°F Decomposition Propellant Blend	Excellent Performance; 1044 seconds, 1060 Pulses
Low Temperature Ignition	Demonstrated Safe Ignition with Reactor/Propellant at -60°F

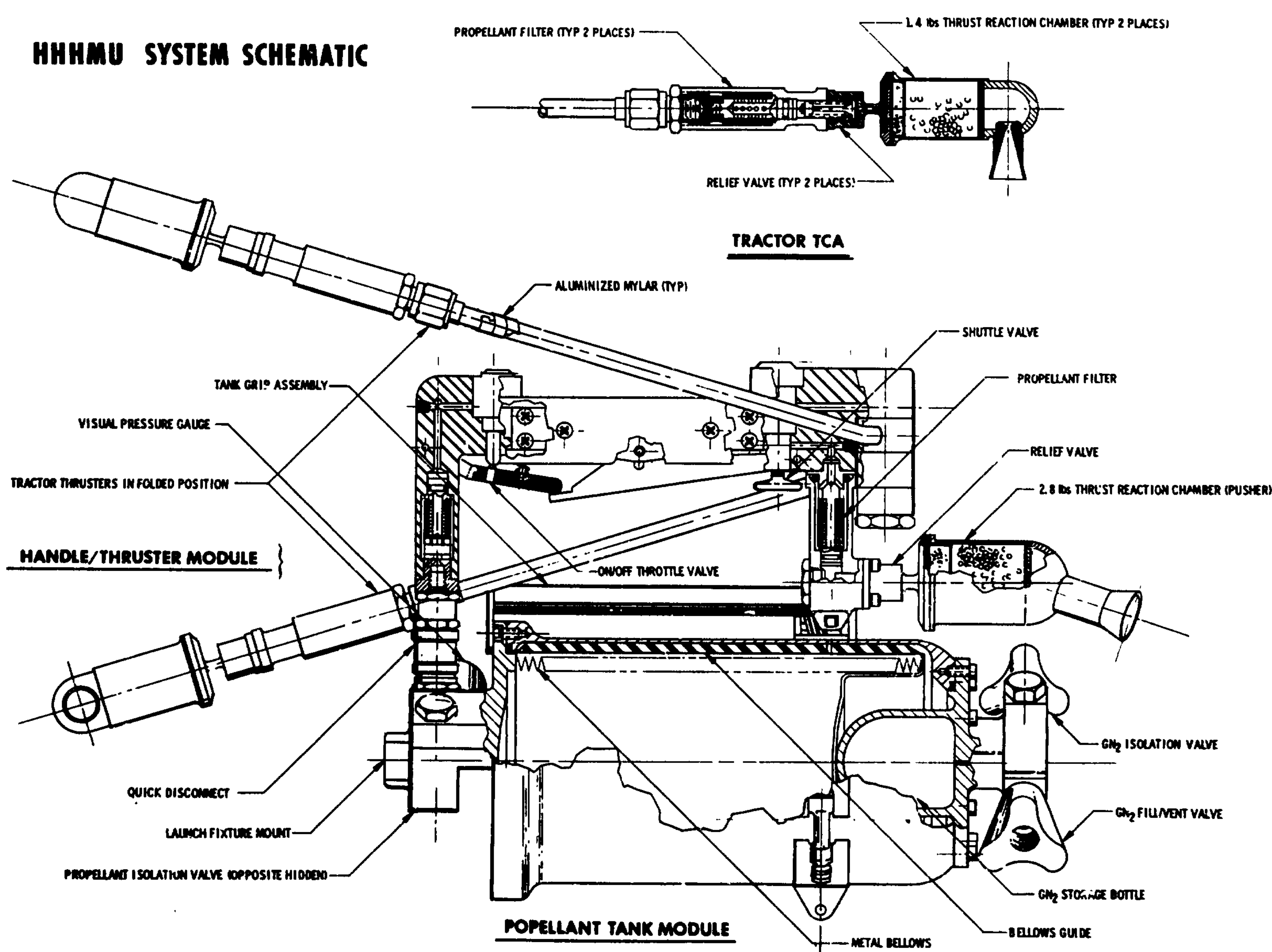

Fig. 11. HHMU System Schematic

quick-disconnect assembly to the H/TM by a 347 welded stainless steel bellows positive expulsion device. Each H/TM is designed to consume ten (10) 250 lbf-sec propellant loads. The $H^3\mu$ is operated in 3 to 1 "blowdown" pressurization mode, with an initial tank pressure of 165 psia. During storage periods, the gaseous nitrogen pressurant is stored in a small cartridge within the PTM at 2000 psia. Pressurant fill/vent, isolation of the pressurizing gas from the propellant tank, and isolation of the liquid propellant from the H/TM is accomplished by three (3) color-coded manual needle valves, which are integral with the PTM.

The modular design approach has been utilized to improve packaging and to permit PTM interchangeability which manifests itself in extended mission capabilities. Each 250 lbf-sec module has a 165-second nominal burntime. Human engineering of the basic flow control handle, needle valve handles, the tank grip assembly, the PTM thumb release lever, and definition of the launch mount interfaces have been provided by NASA-MSC engineering personnel.

The Handle/Thruster Module consists of a 321 stainless steel body structure which contains a 321 stainless steel on-off throttle valve (1.4:1 flow control) and a two (2) position shuttle valve. Seals are ethylene propylene O-rings, and valve return is provided by 302 stainless steel springs. Inlet filtration is provided by a 25-micron absolute Vacco multi-segmented filter which mates with a low spray volume Snap-Tite stainless steel quick-disconnect nipple. The tractor arm and cam assembly is constructed of 321 stainless steel and contains friction washers and O-ring seals to permit locking of the arms in the extended position while providing leak tight joints. The tractor TCA assembly is the same basic construction as the Contract NAS 9-5617 TCA, except that the relief valve flange has been removed and the 30:1 nozzle section has been rotated 90° to permit improved packaging. Chamber pressure has been boosted to 100 psia (nominal) and the bedloading increased to 0.015 lb_m/in^2-sec, corresponding to the 1.5 nominal chamber pressure level demonstrated during Contract NAS 9-5617. Each tractor unit produces a nominal 1.4-lbf thrust initially and blows down to 0.5 lbf at mission end. Throttling over a 1.4:1 range can be accomplished anywhere within the blowdown range. Similarly, the 2.8-lbf thruster pusher unit is basically the same as the earlier TCA, except that the nozzle has been rotated to provide proper pitch and yaw control and the filter/in-line relief valve package redesigned to conform to a 90° packaging requirement. This unit produces 2.8 lbf initially and blows down to 1.0 lbf. Bedloading and chamber pressure are essentially equivalent to the tractor TCA. Figure 12 presents a photograph of the tractor and pusher TCA with test pressure taps in place. A plastic grip covers the handle skeletal structure and can be removed and replaced with custom contoured grips, as required. The triggers are clear anodized 6061-T6 aluminum. Thermal analyses predicated upon assumed mission profiles have resulted in low-emissivity surfaces on the reactors, handle grip, handle body, and the tractor arms and in high-emissivity surfaces on the tractor in-line relief valve bodies (ϵ = 0.80 Sicon R paint). All of the low-emissivity surfaces have a value of 0.20, with the exception of the tractor arms, which are wrapped with a pressure sensitive aluminized mylar tape which has an emissivity of 0.05. One of the PTM to H/TM interfaces is through the forward thumb release latch bracket mount. This mount interface is utilized to dump thermal energy to the PTM and, hence, results in a low-emissivity surface (ϵ = 0.20) on the pusher TCA in-line relief valve body.

The Propellant Tank Module consists of a 347 stainless steel aft closure assembly joined by welding to an all-welded 347 stainless steel bellows assembly. Integrated into the aft closure is a stainless steel quick-disconnect coupler which mates with the H/TM. During storage, a dust cover is attached to the coupler (as well as the nipple). While the quick-disconnect

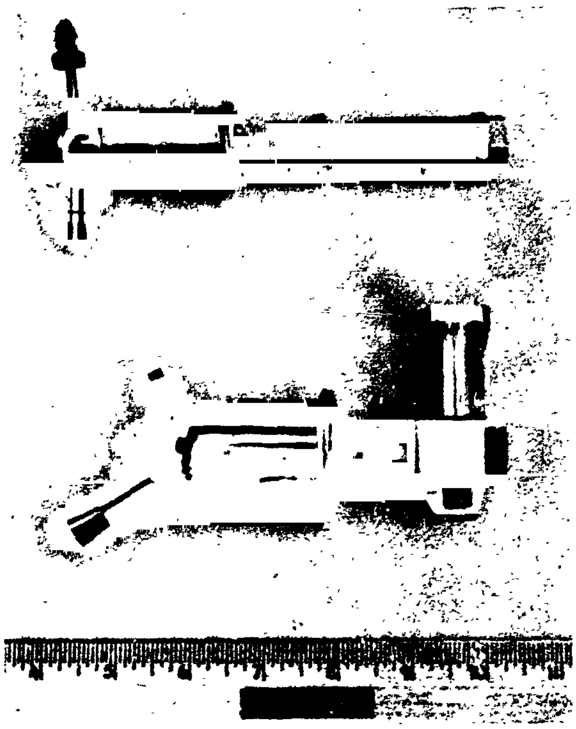

Fig. 12. Tractor and Pusher TCA

coupler provides a redundant seal feature, primary propellant sealing is accomplished with a manual stainless steel needle valve which isolates the monopropellant in the PTM. A visual pressure gauge is also integrated in the aft closure and serves, upon actuation of the pressurant isolation valve, as a go-no-go indicator by providing an analog reading of the initial tank pressure. During EVA periods, the gauge also serves as a propellant depletion indicator and has color bands indicating percentage of propellant remaining. The bellows is contained in a 6061-T6 aluminum shell with a TFE teflon bellows guide to preclude damage to the bellows weld bead during dynamic testing. The bellows can be loaded with 2.28 pounds of propellant and has exhibited expulsion efficiencies in excess of 96 percent following random vibration at 110 percent of qualification levels. The forward closure is constructed of 321 stainless steel and includes a manual-fill/vent valve and a pressurant isolation valve. Three-lobed color-coded cast aluminum handles are utilized to permit actuation of the manual valves. A thin coating of clear anodize is applied to the highly polished aluminum tank shell to provide protection of the shell during humidity exposure. A surface emissivity of 0.2 is attained. To facilitate H/TM to PTM couple/decouple operations, a stainless steel tank grip has been integrated into the PTM to provide a convenient grip point. A stainless steel latch bracket assembly secures the PTM to the H/TM. Table IV presents the environmental design requirements for the HHHMU; Table V summarizes the performance characteristics; and Table VI summarizes salient operating loads which the astronaut must apply to operate the unit.

The following table summarizes the weight and envelope of the $H^3\mu$:

<u>Dry Weight</u>	
PTM, lb_m	8.09
H/TM, lb_m	5.60
$H^3\mu$	13.69
<u>Serviced Weight</u>	15.97
<u>Stowed Envelope</u>	
(inches)	
PTM	12.2 by 6.8 dia.
H/TM	19.5 by 10.2 by 4.0
<u>$H^3\mu$ Envelope</u>	
(inches)	
Deployed	13.5 by 11.5 by 31

Contract NAS 9-6909 requires development, formal qualification, and delivery of end-item hardware to NASA-MSC. All development activities have been completed, and the qualification $H^3\mu$'s are undergoing acceptance testing prior to initiation of module level environmental qualification. Following module qualification, the $H^3\mu$ will be subjected to a 21-mission vacuum firing test program at temperatures of +10°F, ambient, and +120°F. Qualification testing will be culminated with a sea-level manned firing test of a unit which has completed ten (10)

ENVIRONMENTAL REQUIREMENTS

TABLE IV

Operating Temperature	
Nominal Propellant	+ 60°F
Minimum Thruster and/or Propellant	+ 10°F
Maximum Propellant Temperature	+ 120°F
Nominal System Temperature	
Unserviced:	
Maximum	+ 160°F
Minimum	- 60°F
Serviced:	
Maximum	+ 120°F
Minimum	- 60°F
Space Storage and Operation	
Duration	1 year
Pressure	15.5×10^{-6} mm Hg
Relative Humidity	85% with temperature cycling of 70°F to 100°F
Acceleration	1 to 7.25 g linearly with time over 326 seconds - each axis
Shock	30 g half sine wave, 11 ± 1 msec, two (2) shocks each axis
Vibration	Random, 8.8 g rms (profile not included herein)

PERFORMANCE CHARACTERISTICS

TABLE V

	Preliminary baseline NAS 9-6909
Initial Nominal Vacuum Thrust*	2.8 lb_f ± 10%
Initial Minimum Vacuum Thrust**	2.0 lb_f
Final Nominal Vacuum Thrust*	1.0 lb_f
Final Minimum Vacuum Thrust**	0.7 lb_f
Delivered Total Impulse	250 lb_f-sec/load
Delivered Specific Impulse	130 lb_f-sec/lb_m
Propellant	51% N_2H_4/49% H_2O
Stagnation Gas Temperature at 70°F Propellant Feed Temperature	500°F maximum
Maximum Number of Starts (Single TCA)	1000
Maximum TCA Burn Time	1500 seconds
Average Pulse Width	0.25 to 7.0 seconds
Duty Cycle	5 to 50%
Response (In-Line Relief Valve Open to 90%)***	150 msec
Propellant Tank Initial Pressure at 70°F	165 psia
Propellant Tank Maximum Initial Pressure at 120°F	185 psia
Propellant Tank Final Pressure at 70°F	60 psia
Propellant Tank Design Burst	750 psia
Pressurant Cartridge Initial Charge Pressure at 70°F	2000 psia
Pressurant Cartridge Maximum Working Pressure at 120°F	2200 psia
Pressurant Cartridge Design Burst	8800 psia
External Leakage, scc/hr	0 GN_2
Internal Leakage, scc/hr	0 liquid; 2 cc GN_2 in H/TM

* Total forward (two (2) tractor thrusters) or aft direction (one (1) pusher thruster); handle in full open position.

** Total forward (two (2) tractor thrusters) or aft direction (one (1) pusher thruster); handle in minimum flow position.

*** Warm bed conditions.

OPERATING LOADS
TABLE VI

	Required Trigger Force - lbf*		Required Button Force - lbf		Other Loads	
	180 psia**	100 psia	180 psia	100 psia	Maximum	Minimum
On-Off Throttle Valve						
• Crack Open Load	16.0	9.9				
• Hold Open at Minimum Stroke	2.4	2.4				
• Hold Open at Maximum Stroke	8.3	8.3				
Shuttle Valve						
• Crack Open Load			13.2	10.1		
• Hold in Pusher Position			3.5	6.6		
Manual Needle Valve						
• Open					15 in-lbf	6 in-lbf
• Close					10 in-lbf	6 in-lbf
Quick Disconnect						
• Couple/Decouple					Average 15 lbf	
• Decouple at 185 psia Fluid Pressure Differential (Malfunction Condition)					28 lbf	-
PTM Thumb Latch Release						
• Open					5 lbf Nominal	

* Based upon a concentrated load applied to the trigger at 4.00 inches from the forward pivot

** Fluid pressure within Handle/Thruster Module.

250 lb_f-sec missions. During preliminary qualification, the $H^3\mu$ demonstrated its ability to perform satisfactorily in overstress conditions with 250 lb_f-sec vacuum firings at $H^3\mu$ conditioned to temperatures of 0°F and +130°F. Both the PTM and H/TM satisfactorily passed random vibration tests at 110 percent of qualification levels. The needle valves, quick disconnect, bellows, visual pressure gauge, in-line relief valves, and the on-off throttle valve were subjected to overstress cycle testing and all passed satisfactorily.

While Contract NAS 9-6909 is directed at demonstration and qualification of a 250 lb_f-sec baseline $H^3\mu$, the inherent growth potential of the design suggests performance improvements in the area of reduced weight, improved throttleability, increased total impulse and increased specific impulse. For example, the judicious utilization of 6AL 4V titanium as the primary material of construction in lieu of stainless steel will result in a 31 percent dry weight reduction in the H/TM and a 25 percent dry weight reduction in the PTM.

As previously discussed, the reactors, which are the heart of the $H^3\mu$ system, have demonstrated an 8000 lb_f-sec (2 lb_f thrust TCA/4000 seconds burn time) capability with the 51 percent/49 percent propellant blend without indication of performance degradation. Direct application of this single TCA endurance demonstration results in a minimum total impulse capability of approximately 8000 lb_f-sec for the H/TM tractor engines. This represents a 3 fold increase over the baseline qualification level of 2500 lb_f-sec, i.e., ten (10) 250 lb_f-sec loads. In practice the minimum deliverable total impulse would be greater than 8000 lb_f-sec as the pusher engine (2.8 lb_f initial thrust) would be utilized some fraction of the time thereby adding additional impulse capability to the H/TM. Secondly, a life limit value has not been defined for the reactors and it is anticipated that the on-set of performance degradation is well beyond the 4000 second demonstrated value.

This increased total impulse capability could be realized by simply providing additional propellant to the H/TM. Additional propellant could be supplied by 1) carrying more PTM's, 2) developing an in-orbit PTM refueling capability, 3) developing larger capacity PTM's which could be configured as part of a backpack and 4) by providing propellant to the H/TM through an umbilical from a storage tank aboard the spacecraft.

Additionally the throttleability of the reactors (over 10 to 1) represents a means for providing increased flexibility to the current throttled/pulse modulated operational mode.

Performance of the reactors can be increased by the direct substitution of the 1150°F exhaust gas temperature, 72.5 percent N_2H_4/27.5 percent H_2O, propellant (-40°F freezing point) utilized for off-limit tests during Contract NAS 9-5617. This upgraded mix will produce a vacuum specific impulse of 158 lb_f-sec/lbm at ϵ = 30:1 and increase the total impulse of the baseline $H^3\mu$ from 250 lb_f-sec to 304 lb_f-sec. For the same launch weight as the baseline system, a ten (10) tank PTM mission would deliver an additional 540 lb_f-sec, or approximately 178 seconds of burntime. Expansion ratios in excess of 30:1 could probably be entertained with the 72.5 percent N_2H_4/27.5 percent H_2O mix in view of the higher water vapor superheat (1150°F versus 500°F), thereby increasing the specific impulse by an additional 3 to 4 percent.

Should additional performance be required, a ternary hydrazine monopropellant with a -20°F freezing point could be utilized with the $H^3\mu$, although the reactor would require redesign to withstand the increased thermal soakback to the injector under pulse mode operation. At ϵ = 50:1, a 226-second vacuum specific impulse is produced with a 1900°F exhaust gas temperature. Reactor tests at the 5-lb_f thrust level have verified performance and reactor design for this ternary blend. Materials of construction of the baseline system are compatible with this ternary blend.

Uprating of the reactors to produce higher specific impulse is contingent, however, upon the ability of astronaut suit materials to withstand the increased temperatures of the exhaust gas plume. Fundamentally, the $H^3\mu$ can be uprated with minimal impact to the current baseline design configuration.

The development of the $H^3\mu$ has not only provided a more sophisticated, higher performance maneuvering unit but has also provided technology and hardware that is adaptable to other HHMU systems and propulsion systems as well. Cold-gas Hand-Held Maneuvering Units, though less efficient might still be useful for maneuvering around objects like camera or telescope lens systems that could be contaminated with combustion products from liquid fueled devices. The compactness, lightweight, operational versability and uncomplicated astronaut interface of the HHMU and $H^3\mu$ favor their utilization in the future either as primary EVA maneuvering devices or as supplements or backups to their more sophisticated integrally stabilized cousins.

MECHANICAL DEVICES FOR ZERO GRAVITY SIMULATION

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NASA-MARSHALL

SUMMARY

Mechanical simulator development conducted in support of the Apollo Applications Program and in anticipation of future programs is summarized in this review. Mechanical simulators that have been developed are illustrated and their characteristics described. Results are reported on investigations of air bearing systems, air bearing cart thrust systems, and devices for supporting hand tools and the serpentuator. Applications of mechanical simulators are also illustrated.

INTRODUCTION

The Manufacturing Engineering Laboratory divides earth orbital weightless and lunar gravity simulation devices into two categories: mechanical and neutral buoyancy. Mechanical simulators include all those devices that do not use a liquid such as water to support the subject, workpiece, or tool.

Some of the objectives of mechanical simulation are evaluation of design concepts, evaluation of hardware, and determining the subject's capability for performing tasks.

Mechanical simulation offers certain advantages over neutral buoyancy simulation in that much less preparation time is required. It can be performed by a minimum of two people (a test subject and technician to balance him and operate the suit), and the simulators can be moved to the work site. The advantages of neutral buoyancy simulation over mechanical simulation are that complete tasks requiring vertical clearance or large changes in vertical elevation can be performed in one operation as opposed to breaking the task up into several part tasks so that it can be performed in a mechanical simulator.

The research and development efforts in the mechanical simulation category are discussed in this review under the divisions of mechanical simulators, air bearing systems, air bearing cart thrust system development, devices for supporting hand tools and serpentuator, and the use of mechanical simulators.

MECHANICAL SIMULATORS

FIVE-DEGREES-OF-FREEDOM SIMULATOR

1. General Information

The five-degrees-of-freedom simulator is an aluminum framework mounted on air-bearing pads. Yaw and horizontal translation in two directions are obtained by moving the entire simulator on its air-bearing pads. Pitch and roll motions are obtained through gimbals mounted with anti-friction bearings. Pitch is the only motion that is limited. This simulator was designed to be used with the Lunar Gravity and Earth Orbital Simulator that is described in this paper.

2. Technical Information

The five-degrees-of-freedom simulator consists of three major assemblies: a cradle or seat, roll yoke, and base (Fig. 1). The cradle, or seat, supports the subject in an erect position and contains provisions for adjusting the position of the subject relative to the roll and pitch axes of the simulator. The roll yoke supports the cradle at the pitch axis and permits 108 degrees of rotation in pitch between the cradle and the yoke. The yoke also contains a system for supplying ventilation, breathing, and pressurization gases to a subject in a space suit.

The "U"-shaped yoke is supported at the base of the "U" roll bearings that permit unlimited rotation around the roll axis. The base structure supports the roll bearings and distributes the total load of the simulator and subject to three air-bearing pads equally spaced around the nominal yaw axis.

The simulator has an onboard air supply which requires a 115 Vac, 5 A, 60 Hz (60 cps) power input.

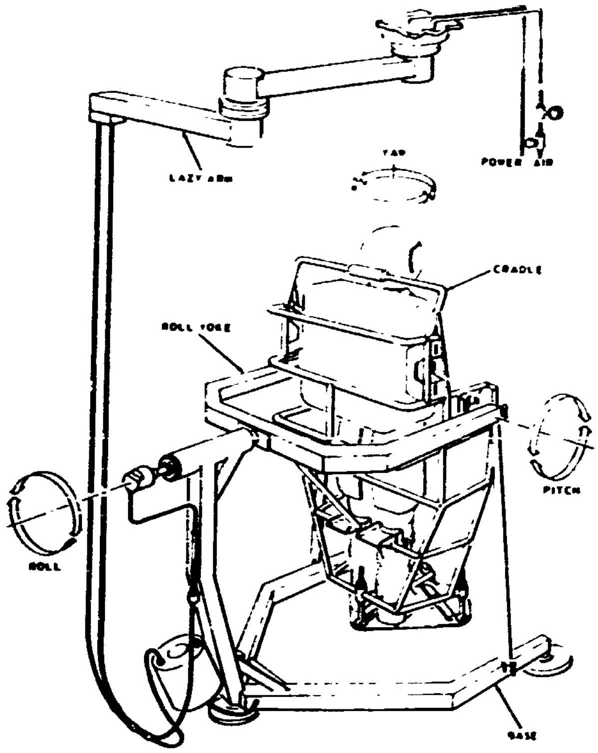


FIGURE 1. FIVE-DEGREES-OF-FREEDOM SIMULATOR

In addition, there are provisions for supplying 0.283 m^3/min (10 scfm) of breathing and suit pressurization air at 0.31 MN/m^2 (45 psig) through a hose to the subject.

A lazy arm may be used to minimize the hose and power cable drag by positioning the hose in an essentially constant vertical position. Air is fed through rotating unions so that each of the two sections of the lazy arm is capable of unlimited rotation.

Cradle Assembly. The cradle assembly consists of the supporting structure for subject and back pack; the subject's restraint system, consisting of the torso corset, leg supports, and restraining straps; and the vertical and fore and aft balancing adjustments and pitch axis ball bearings (Fig. 2).

The supporting structure is welded tubular aluminum with an aluminum foot plate supported by three adjustment screws. The adjustment screws permit raising or lowering of the subject's center of gravity (c.g.) over a 0.152 m (6 in.) range to place the

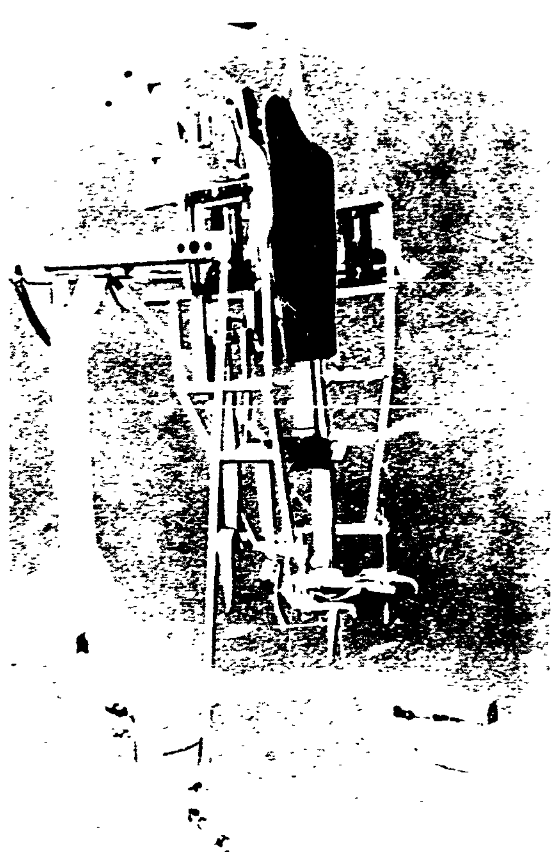


FIGURE 2. FIVE-DEGREES-OF-FREEDOM SIMULATOR WITH CRADLE ASSEMBLY ATTACHED

c.g. within the range of the vertical balancing adjustment. The back pack supports are attached with machine screws fitted in slotted holes to permit vertical adjustment of the back pack position over a range of 0.076 m (3 in.). Right and left adjustment of the back pack position is accomplished by selective tightening or loosening of the back pack attachment screws. An angle on each side of the cradle is provided for attachment of the torso corset. These angles are located in the plane of the subject's back, and they establish the fore and aft positioning of the subject. The torso corset is attached to each angle at 4 points with 8 machine screws. Threaded screw holes are provided on 0.013 m (0.50 in.) centers over a range of 0.229 m (9 in.) and slotted holes in the corset attachment fitting permit locating the corset at any point within the extreme limits. A 0.203 m (8 in.)

section of tube is welded to the back of the structure, perpendicular to the vertical axis of this cradle. It is used for attachment of counterweights, if required, to bring the cradle and subject's c.g. within the range of the balancing mechanism.

The subject's restraint system confines his torso and legs while his head and arms remain free. The torso restraint consists of a two-piece fiberglass corset. The two pieces of the corset are supported by two 0.025 m (1 in.) diameter aluminum tubes bolted to the angles described above. The two pieces of the corset can be adjusted horizontally to move the subject right or left and to accommodate different torso widths. The sections are clamped into place on the tubes by the integral split ring clamps and bolts accessible from the front. Supplementing the corset are restraining straps located at the following positions: shoulders, pelvis, knees, and feet. The two shoulder straps are fastened to the corset at the approximate location of the shoulder blades. Each strap is brought over the shoulder, across the upper chest, and back under the opposite arm to attach to the cradle structure at waist level. These straps consist of two parts that are connected together with aircraft-type, quick-release, adjustable buckles located in the upper chest area. The pelvic strap is a one piece strap around the corset and subject at his hips. It is attached to one side of the corset to prevent slipping and has the same type buckle as the shoulder straps.

The legs are restrained at the knees by individual, sponge-rubber padded, contoured supports attached to the cradle structure. The supports may be adjusted side to side and fore and aft. Each support has a continuous strap held in place to prevent slipping and fastened on the outside of the knee with the adjustable quick-release buckle.

The foot plate is covered with corrugated rubber tread and is provided with semi-circular heel retainers and a foot strap. The heel retainers prevent the heels from slipping backward and the strap restrains the feet from forward or vertical movement. The strap is attached at both ends with adjustable strap restrainers and, normally, enough slack is left to permit insertion of both feet under the strap. After both feet are in place, a locking bar, pivoted at the back between the heels, is dropped into place between the feet and is secured by engaging a "J" hook with front edge of the foot plate. A screw handle permits tightening the "J" hook to apply tension to the foot strap.

The balancing adjustments (Fig. 3) are built into the right and left sides of the cradle to permit movement of the subject's c.g. 0.038 m (1.5 in.) vertically and fore or aft from the nominal c.g. position with

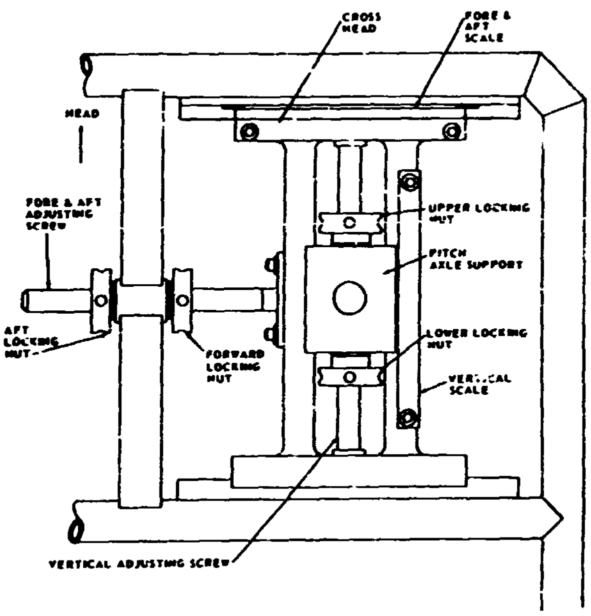


FIGURE 3. BALANCING ADJUSTMENT

respect to the pitch axis ball bearings. Each adjustment consists of a crosshead and adjustment screw that provides fore and aft motion of the cradle. The adjustment screw slides through a bushing and is locked in place by two adjusting nuts, one on each side of the bushing. The crosshead carries the pitch axle support and vertical adjusting screw. The pitch axle support slides on the crosshead, the adjusting screw passing through it. Two adjusting nuts, one on the top and one on the bottom, permit vertical adjustment and locking. The crosshead has two reference scales, one for vertical adjustment and one for horizontal. The cradle has a pointer for the horizontal scale and there is an index mark on the pitch axle support for vertical reference. The scales permit equal adjustment of both right and left sides. When a subject has been balanced previously, the balancing adjustments can be preset, thereby reducing the time required to accomplish this task.

The pitch axle fits into the bore of a self-aligning ball bearing mounted by two bolts to the roll yoke.

The axle has a threaded hold accessible from the outer edge. This can be used for attachment of instrumentation to obtain a readout on the pitch movement.

Seat Assembly. The seat assembly consists of the supporting structure for the subject; the subject restraint system consisting of the bicycle seat, back support, and restraint straps; and the cradle vertical and fore and aft balancing adjustments with pitch axis ball bearings (Fig. 4).

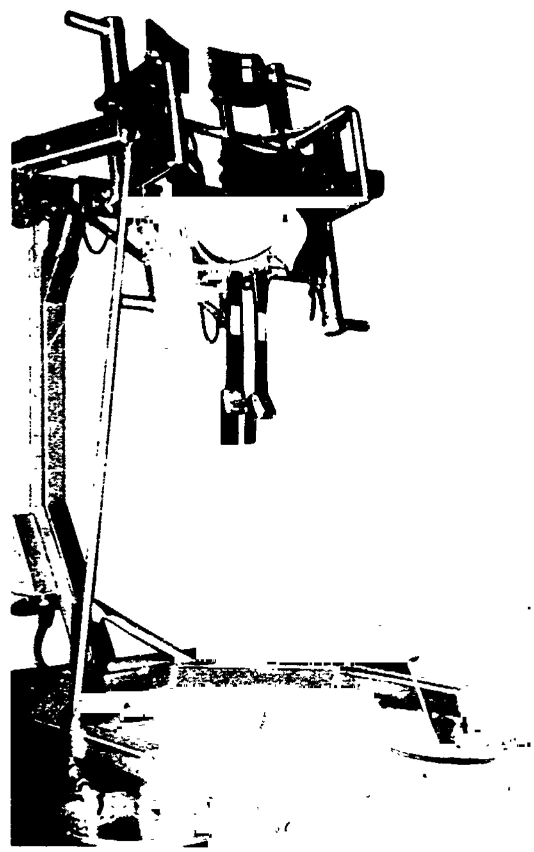


FIGURE 4. FIVE-DEGREES-OF-FREEDOM SIMULATOR WITH BICYCLE SEAT ASSEMBLY ATTACHED

The supporting structure is made of aluminum tubes welded together and a bicycle seat that is fastened in place with a set screw. The adjustment of the seat height permits raising and lowering the subject's c.g. over a 0.152 m (6 in.) range to place the c.g. within the range of the vertical balancing adjustment. The back supports are attached with machine screws fitted in slotted holes to permit horizontal adjustment. A 0.203 m (8 in.) section of tube is welded to the back of the structure, perpendicular to the vertical axis of seat. It is used for attachment of counterweights, if required, to bring the seat and subject's c.g. within the range of the balancing mechanism.

The subject's restraint system confines his torso while his head, arms, and legs remain free. The torso restraint consists of four curved-aluminum, sponge-rubber lined, back supports. The four back supports are supported by two 0.0254 m (1 in.) square aluminum tubes welded to the structure. The four back supports can be adjusted horizontally to move the subject right or left and to accommodate different torso widths. Supplementing the back supports are restraining straps for the torso and pelvis. The two straps are fastened to the upper back supports at the approximate location of the shoulder blades. Each strap is brought over the shoulder, across the upper chest and back under the opposite arm to another strap attached to the lower back support at waist level. These straps consist of two parts which are connected together with an aircraft-type quick-release adjustable buckle located in the upper chest area. The two parts of the pelvic strap are attached to the lower back supports. It has the same type of adjustable buckle as the shoulder straps.

The subject is held on the bicycle seat by two strap assemblies, one of which goes over each leg. The ends of each strap are attached to the frame below and behind the crotch and the lower back supports.

The seat uses the same balancing adjustment mechanism as the cradle.

Roll Yoke. The roll yoke supports the pitch bearings in a "U" shaped structure of welded 0.076 m (3 in.) square aluminum tubing attached to a hardened steel shaft. The yoke permits a pitch movement of 108 degrees with approximately equal pitch-up and pitch-down motion. The cradle has adjustable stops that strike rubber pads on the yoke to limit the cradle motion so that the subject does not strike the base during extreme motion in both pitch and roll.

The yoke also has two small tabs, one welded to the front of the right arm of the yoke and the other to the bottom at the rear center. These tabs are attachment points for locking bars. Locking the pitch movement is accomplished by attaching a 0.025 m (1 in.) diameter aluminum tube to the back tab and to the rear of the cradle at about the knees. A notch in each end of the bar slips over the tabs on the cradle and the yoke, and each end is held secure to the tab by a ball lock pin. Locking the roll movement is accomplished by attaching the roll locking bar between a tab on the base and the tab on the yoke arm. Locking of one axis does not restrict the motion of the other axis.

The roll axle fits through the roll yoke and the bearings. It is attached to the roll yoke from the front (inside the U) by two machine screws. The axle has a 0.013 m (0.50 in.) diameter passage which connects through standard pipe fittings to a Hansen 5000 series quick-disconnect fitting at the front end of the left arm. This is used to provide breathing and pressurization air to the subject in a space suit. The aft end of the axle has a Deublin model 20-8 rotating union to provide rotating freedom and a continuous air supply. The shaft of the rotating union extends completely through the union and may be drilled and tapped to provide a mounting for instrumenting the roll axis, if desired.

The vernier roll balance adjustment is located between the arms of the yoke to the rear of the cradle. It has a 1.36 kg (0.0933 slugs) lead mass mounted to slide freely on a threaded rod. Nuts on each side provide locking for the lead mass. The hole in the lead is positioned slightly off center so that it will hang at about a 50 degree angle from the vertical to clear the cradle in the extreme pitch-up position. The upper corner of the lead has been beveled and padded with rubber to minimize damage to the back pack in the event that contact does occur.

Base Assembly. The base is an aluminum square tube structure that provides support for the roll axle and the three air bearing pads. The vertical member of the base supports the roll axis approximately 1.59 m (62.5 in.) off the floor. It is designed to provide clearance for the cradle to permit unrestricted roll in any pitch attitude with the cradle foot plate in the lowest position.

The bottom tubular structure provides support for the air bearing pads that are equally spaced on a 0.635 m (25 in.) radius circle about the nominal yaw axis. One pad is directly under the vertical base member and the other two are forward and to the side to give stability.

The air bearing pads on 0.019 m (0.75 in.) threaded rods are screwed into fittings welded to the base.

The tubular base structure serves as a plenum chamber to equalize air flow to the pads and minimize line surges. The input to the base plenum is located on the right rear side.

Lazy Arm Assembly. The lazy arm provides for positioning the upper end of both the power cable and breathing and suit pressurization air hose at any point within a 3.66 m (12 ft) diameter circle (Fig. 1). The lazy arm consists of two 0.915 m (3 ft) sections that rotate on thrust bearings to provide minimal friction forces. The lazy arm mounting plate is bolted to the supporting structure and leveled. Suspended from the mounting plate is the inner arm, which rotates about the center on thrust bearings that are concentrically mounted around the rotating unions. The two unions are tandem-mounted on the rotating axis and are capable of unlimited rotation. There's a similar rotating joint between the inner and outer arms. The outer arm terminates in fittings for the attachment of the power cable and flexible hose leading to the simulator. Although the nominal restriction in the rotating union is 0.0063 m (0.25 in.) ID, larger diameter hoses have been provided to minimize functional losses in supply lines. Air is provided through a 0.0095 m (0.375 in.) ID hose.

A force of approximately 0.556 N (0.125 lbf) applied at the hose fittings is required to move the lazy arm when it is fully extended.

Basic Data. Figures 5 and 6 show detailed dimensions and operating clearances for the five-degrees-of-freedom simulator. The total mass of the

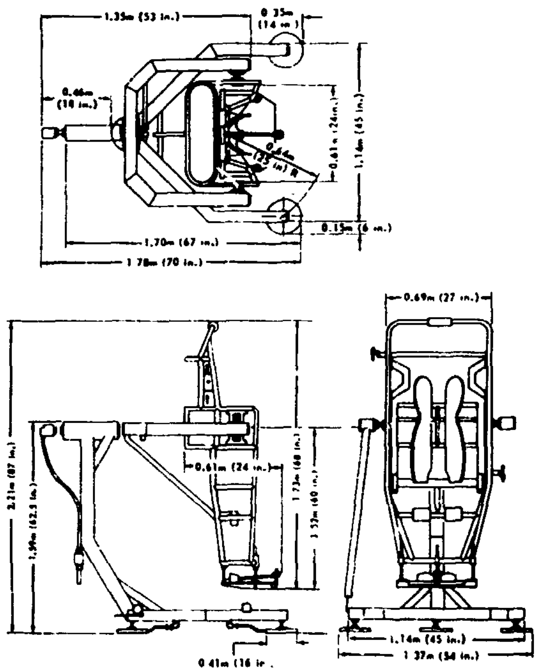


FIGURE 5. BASIC DATA

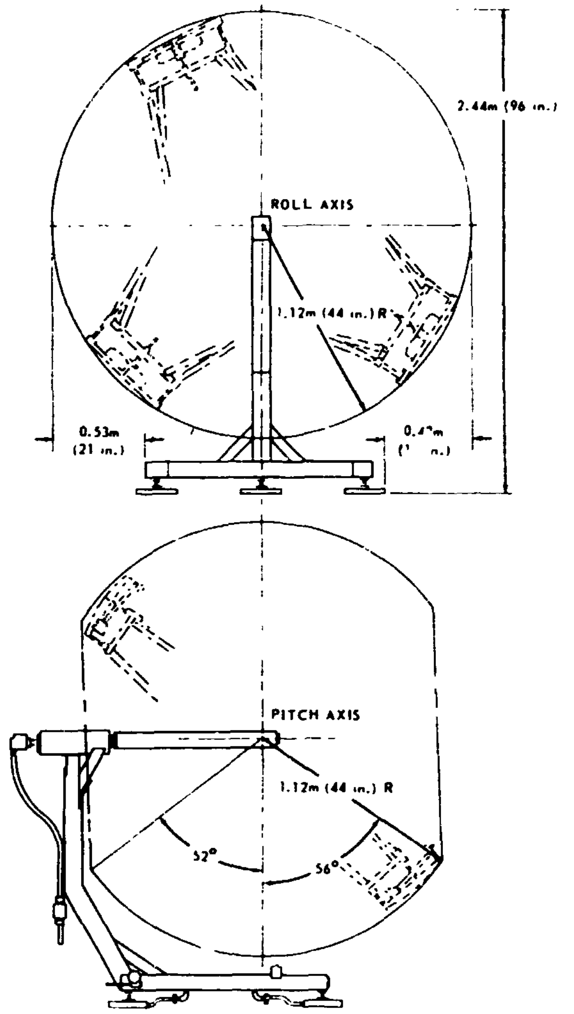


FIGURE 6. CLEARANCES REQUIRED FOR OPERATIONS

simulator with the cradle is 90 kg (6.15 slugs). The cradle has a mass of 28.6 kg (1.96 slugs) and the base and yoke 61.4 kg (4.20 slugs).

The torques required to overcome the static friction of the simulator are the following: pitch, 0.284 J (40 in. oz); roll, 0.494 J (70 in. oz); and yaw, 0.007 J (1 in. oz).

ACTION-REACTION FREE-FALL SIMULATOR

1. General Information

The action-reaction free-fall simulator or six-degrees-of-freedom simulator is a mechanical apparatus that allows the subject to react to any force as he would in space, except where the subject is required to translate over great distances (Fig. 7). This is accomplished by designing the experiment so that the desired test data are obtained.

Assume the following for the purpose of demonstrating that the subject can produce a translation acceleration just as he would in space by properly designing the experiment:

F_a = force applied to the object by the subject, assume 13.3 N (3 lbf)

F_s = force required to bend the pressure suit, given as 44.4 N (10 lbf) for 2.41×10^4 N/m^2 (3.5 psia) pressurization.

$F_c = F_a + F_s$

F_μ = force required to produce translation of the moving parts of the simulator, given as 0.58 N (0.13 lbf)

$F_T = F_a + F_s + F_\mu$

m_m = mass of the moving parts of the simulator, 114 kg (7 78 slugs) [a weight of 1110 N (250 lbf) at 1 g]

m_s = mass of subject and space suit, 91 kg (6.21 slugs) [a weight of 889.6 N (250 lbf) at 1 g]

g = acceleration, standard free fall, 9.80 m/sec^2 (32.16 ft/sec^2)

From Newton's Second Law of Motion, "The change of motion is proportional to the motive force impressed; and is made in the direction of the straight line in which force is impressed," we can determine the subject's acceleration in space.

$$F_c - F_s = ma$$

$$a = \frac{F_c - F_s}{m}$$

$$a = \frac{57.7 - 44.4}{91} = 0.146 \text{ m/sec}^2 \ (0.51 \text{ ft/sec}^2)$$

To produce the same acceleration in the simulator, it is necessary to change a parameter(s) that will produce the correct applied force, F_a, and not degrade the test data. Solving for the total force, F_T, we obtain

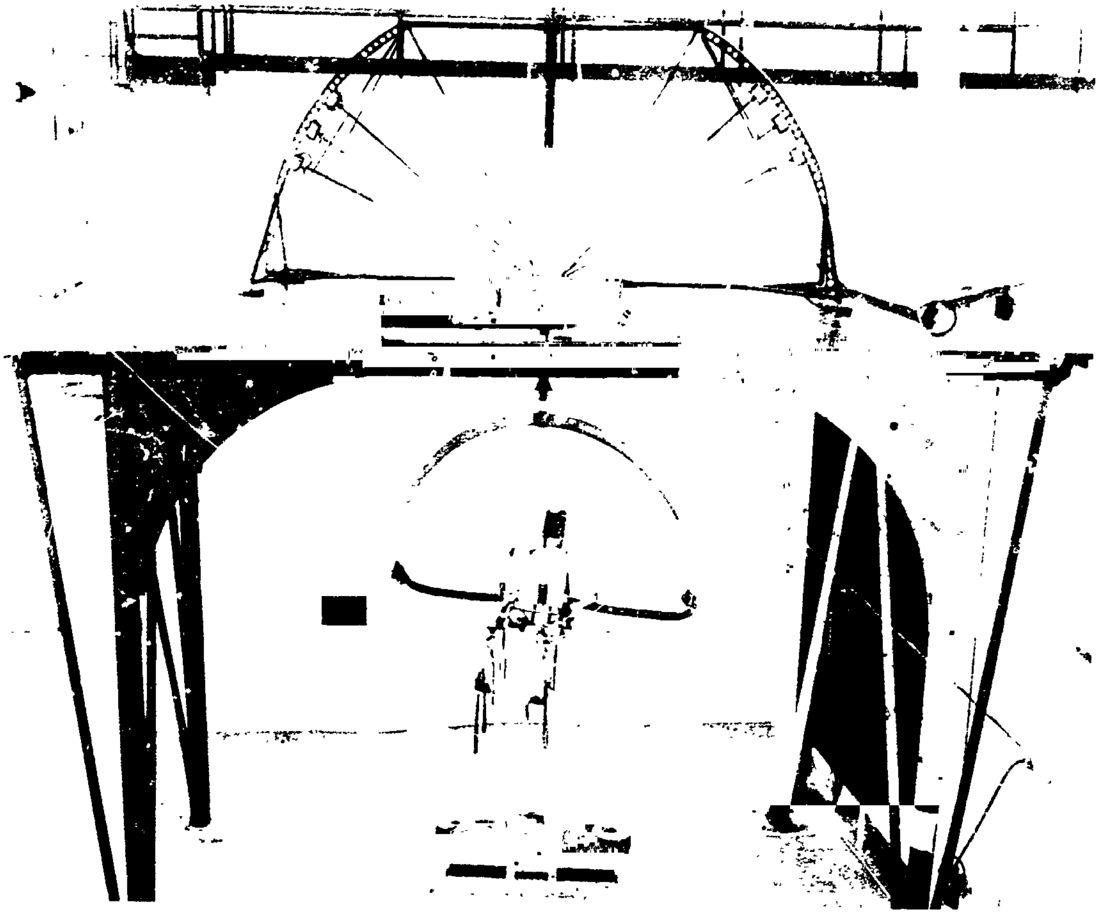


FIGURE 7. ACTION-REACTION FREE-FALL SIMULATOR

$$F_T = ma = (m_m + m_s)a$$

$$F_T = (114 + 91) \times 0.146 = 30 \text{ N } (6.75 \text{ lbf})$$

$$F_T = F_a + F_s + F_\mu$$

Since F_μ is an inherent characteristic of the simulator, it cannot be changed. Assume that the F_a is to be the same as it would be in space. This leaves only F_s which can be altered.

$$F_s = F_T - F_a - F_\mu$$

$$F_s = 30 - 13.3 - 0.58 = 16.12 \text{ N } (3.63 \text{ lbf})$$

A force of 16.12 N (3.63 lbf) to bend the suit can be obtained by reducing the suit pressure to 1.19×10^4 N/m^2 (1.74 psig).

The same analysis can be done for the other degrees-of-freedom for the action-reaction free-fall simulator.

2. Technical Information

The subject is strapped into a fiberglass harness that both positions him properly in the gimbal axis and adjusts to fit anyone between 1.65 and 1.86 m (65 and 73 in.) tall. The test subject is supported in such a way that he can rotate freely about any axis. He is capable of moving to either his left or right side, forward and backward, or up and down. (Fig. 8.) In other words this is a six-degrees-of-freedom simulator.

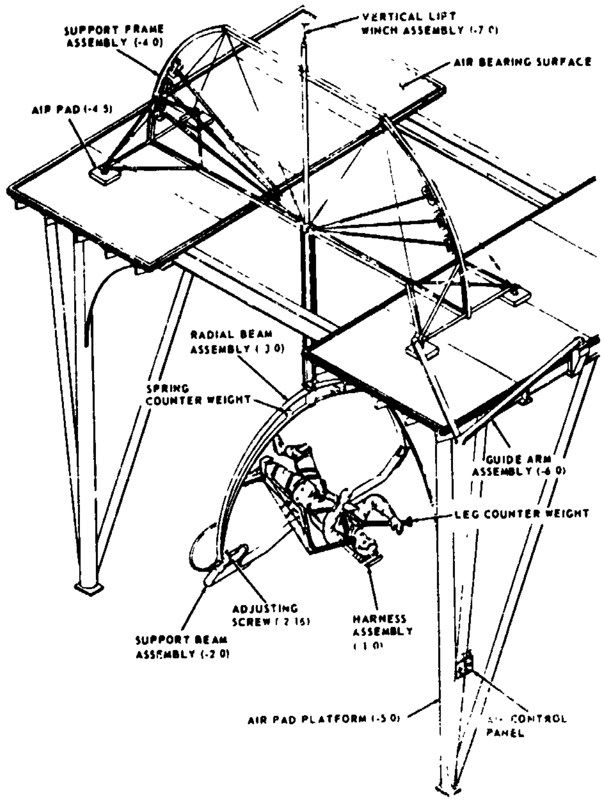


FIGURE 8. ACTION-REACTION, FREE-FALL SIMULATOR

The harness allows the astronaut's leg movements (Fig. 9), while counter-weighting them in any position, and keeping his center of gravity in the same place. (Fig. 10). The gimbal axis and harness have a built-in air line for breathing and space suit pressurization and cooling. This allows the subject to train for weightlessness either with or without his suit.

To allow the astronaut to move freely horizontally, the simulator is mounted on four almost frictionless air bearings. This, combined with the small mass of the simulator, less than 114 kg (9.33 slugs) gives almost no resistance to movement through its 1.83 × 3.66 m (6 × 12 ft) horizontal working envelope. Negator springs, which resemble a belt wound on a spool, are attached to the gimbal axis. By mounting sets of springs as shown in Figure 11, the springs can exert any constant force over their six feet of vertical travel.

Adjustments of the simulator can also be made to lift five-sixths or two-thirds of the test subject's mass. In this way, Lunar, Mars, and Venus gravity can be simulated.

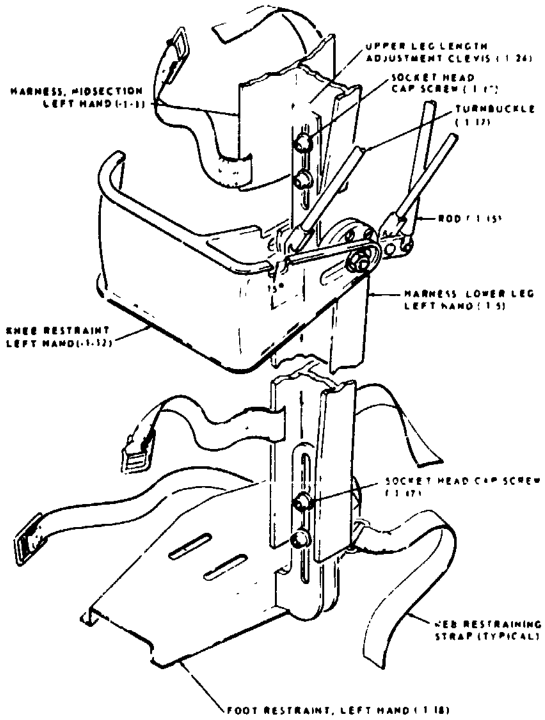


FIGURE 9. ADJUSTING HARNESS ASSEMBLY

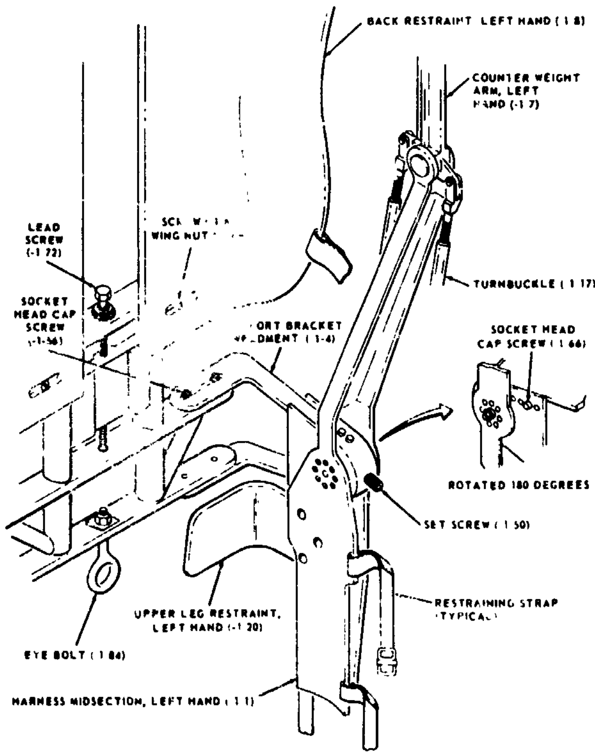


FIGURE 10. ADJUSTING HARNESS ASSEMBLY

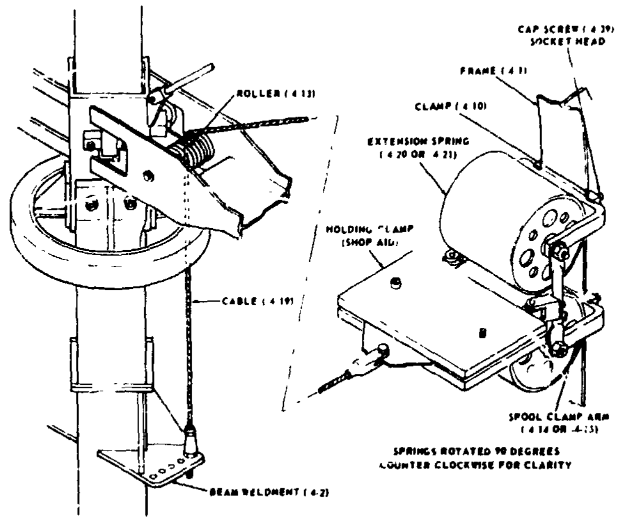


FIGURE 11 INSTALLATION OF EXTENSION SPRINGS

LUNAR GRAVITY AND EARTH ORBITAL SIMULATOR

1. General Information

This simulator was designed to be used with the five-degrees-of-freedom simulator (Fig. 12). The purpose of this simulator is to support the object the subject is working on. It provides four degrees-of-freedom for the workpiece. Vertical translation is produced by moving the work panel up and down. The remaining three degrees-of-freedom are in the horizontal plane. The simulator is supported by three air bearing pads which offer a minimum of resistance to horizontal translation.

For simulating lunar gravity, the subject stands on the semicircular platform (Fig. 12). A mass equal to one-sixth of the subject's mass is placed in the counterweight box. This added mass produces an upward force on the subject's feet which gives him the sensation of walking on the lunar surface.

In the earth orbital (zero gravity) mode, the semicircular platform is removed and sufficient mass is added to the weight box to bring it and the work panel into equilibrium.

2. Technical Information

The lunar gravity and earth orbital simulator consists of the following components: base and support structure, parallelogram arms, work panel assembly, platform assembly, counterweight assembly, and air bearing system (Fig. 13).

Base and Support Structure. The base structure consists of a welded triangular, tubular aluminum base that holds the air bearing pads and serves as a plenum for air supply. The support structure is a rectangular frame made of aluminum channel and welded to the base structure. The support structure supports the parallelogram and transmits all loads to the base.

Parallelogram Arms. The parallelogram arms are two welded aluminum channel assemblies that are supported near their centers on the support structure. Each arm is a lever connecting the work panel and platform assembly to the counterweight assembly. Each arm contains six self-aligning ball-bearing joints for vertical motion.

Work Panel Assembly. The work panel assembly, attached to the front of the parallelogram arms, provides a vertical mounting surface for work objects.

Platform Assembly. The detachable platform is provided for a walking or foot placement surface for experiments other than those performed in zero gravity. An aluminum lip around the edge of the platform is provided to permit gravel or simulated "moon dust" to be placed on the platform for more realistic testing. Folding legs on the platform can be extended for stability when setting up the experiments. These are folded out of the way during operation.

Air Bearing System. The air bearing system consists of a blower and variable transformer control, an air distribution system, and three air bearing pads. The base structure serves as a plenum chamber to prevent air pulses and consequent bearing instability. The front two air bearing lines are provided with bleed-off air valves. The rear pad uses a restriction type valve. The three valves and the variable transformer can be adjusted to provide an even lifting force to elevate the entire structure.

AIR BEARING CAPTURE AND TRANSFER SIMULATOR

1. General Information

This simulator was designed to provide the capability to investigate the following: docking, grappling mechanisms for capture and attachment to work surfaces, manipulators for handling and maneuvering of masses, and boom dynamics associated with mechanisms for capture, despin, respin, and insertion of cooperative and noncooperative objects in earth orbit.

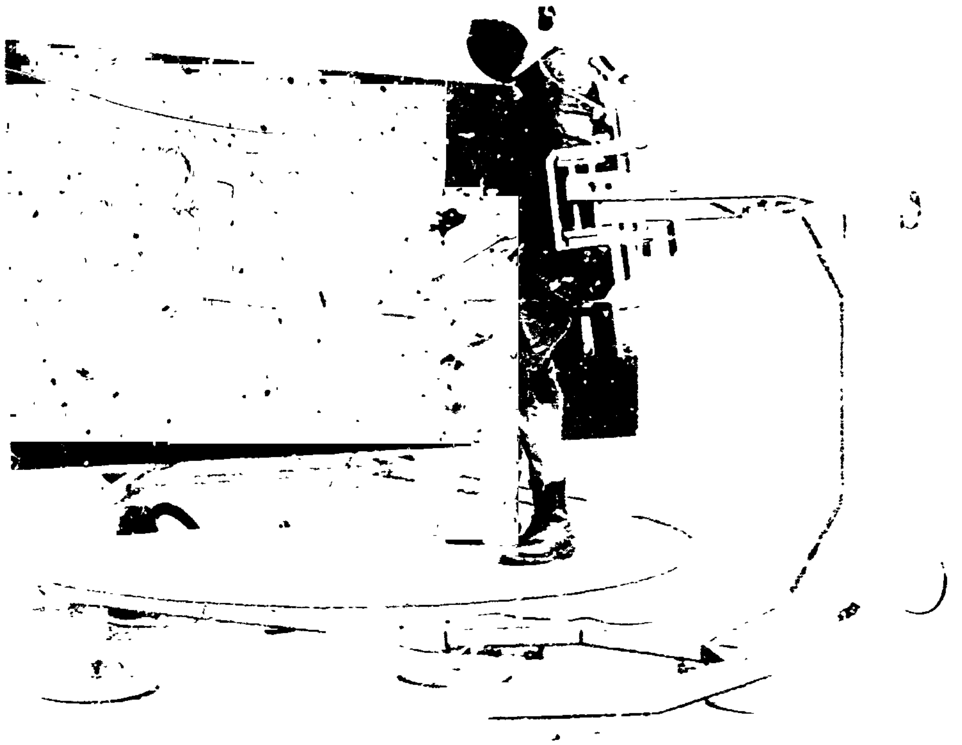


FIGURE 12. LUNAR GRAVITY AND EARTH ORBITAL SIMULATOR AND FIVE-DEGREES-OF-FREEDOM SIMULATOR BEING USED FOR LUNAR GRAVITY SIMULATION WORK

2. Technical Information

The air bearing capture and transfer simulator (Fig. 14) consists of the following major components: frame assembly, walking beam assembly, air bearing pads, air supply system for pads, thruster system (Fig. 15), thruster control electronics package and control stick assembly, transportation casters, and personnel seat assembly.

Frame Assembly. The frame 10 is rectangular in shape, made of square aluminum tubes welded together, and serves as a plenum for the pad air. The front air bearing pads are attached to it.

Walking Beam Assembly. The beam 35 is free to pivot in a plane perpendicular to the frame. This provides the simulator with the same leveling effect on an uneven floor as three air pads but with the load carrying capacity of four pads.

Air Bearing Pads. The pads like the one shown in Figure 16 are used on this simulator.

Air Supply System for Pads. A Black & Decker Manufacturing Company Model 820 EDA heavy-duty central cleaning system unit 12 is used on this simulator.

Thruster System. Six thrusters 60 are used with this system. A high pressure storage sphere 11, fill valve 44, regulator 46, ball valve 50, manifold 56, solenoid valves 57, and tubes 53 and 55 make up the other major components of the system (Fig. 15).

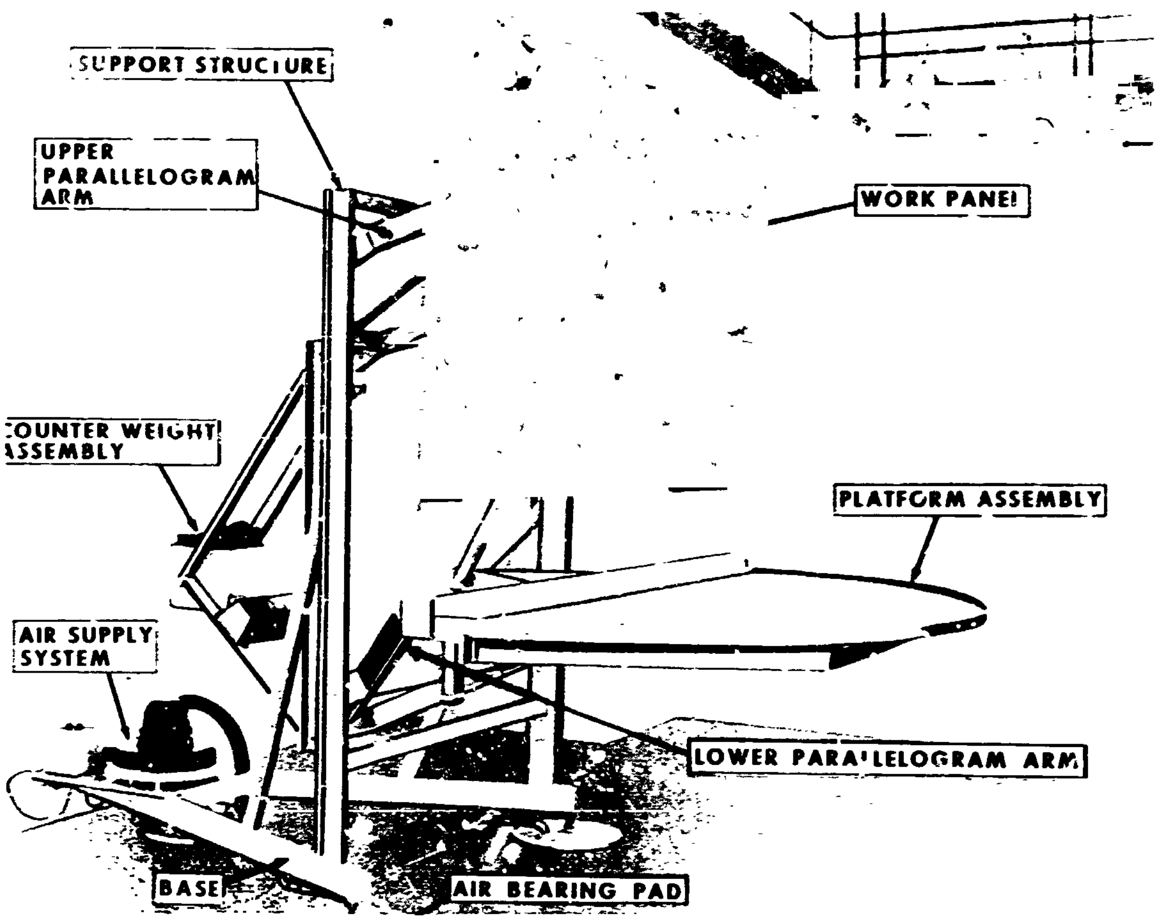


FIGURE 13. LUNAR GRAVITY AND EARTH ORBITAL SIMULATOR

Thruster Control Electronics Package and Control Stick. A military aircraft control stick SA/2-0 electrically actuates relays in the control electronics package, which in turn electrically actuates the solenoid valves 57.

Transportation Casters. These casters 1 provide a means of moving the simulator other than on its air bearing pads. When in use the pads do not touch the floor.

Personnel Seat. The seat support 28 provides a place to attach the air supply 12 as well as support the personnel seat 29.

3. Application.

A grappler developed for use as a means of attaching spacecraft to work surfaces is shown mounted on the simulator. When tests have been completed it can be removed and so the simulator can be used for other tests.

AIR BEARING SYSTEM DEVELOPMENT

MEDIUM INLET PRESSURE AIR BEARING SYSTEM

Air Bearing Pads. A Hovair air bearing pad made by the Inland Division of General Motors Corporation is shown in Figure 17. This pad will support various loads with the pressures and volumes of air described by the load map shown in Figure 18. Inlet pressures of approximately 0.62 MN/m^2 (90 psig) are used with this pad. For the purpose of this review, this is a medium inlet pressure.

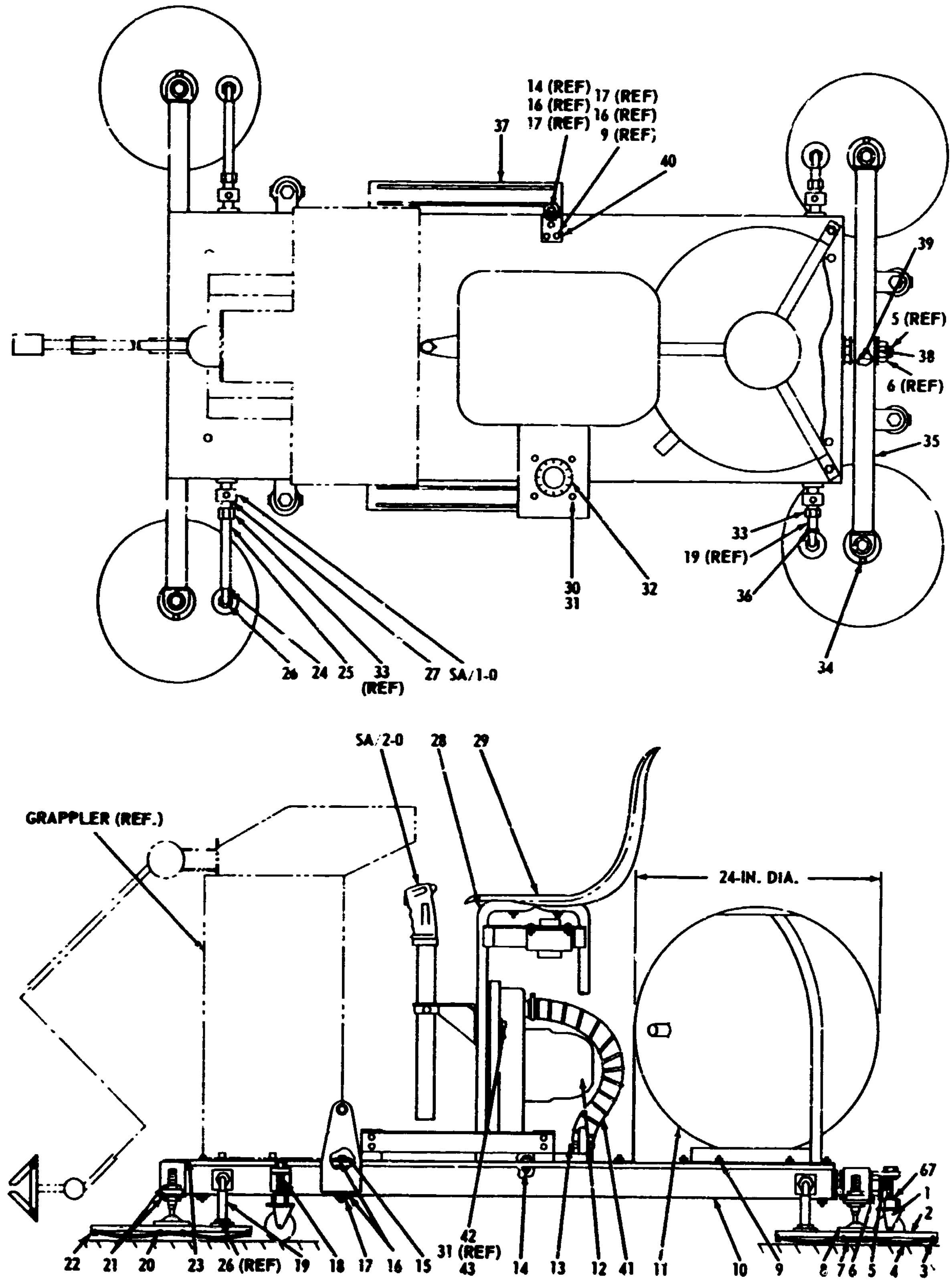


FIGURE 14. AIR BEARING CAPTURE AND TRANSFER SIMULATOR – TOP AND SIDE VIEW

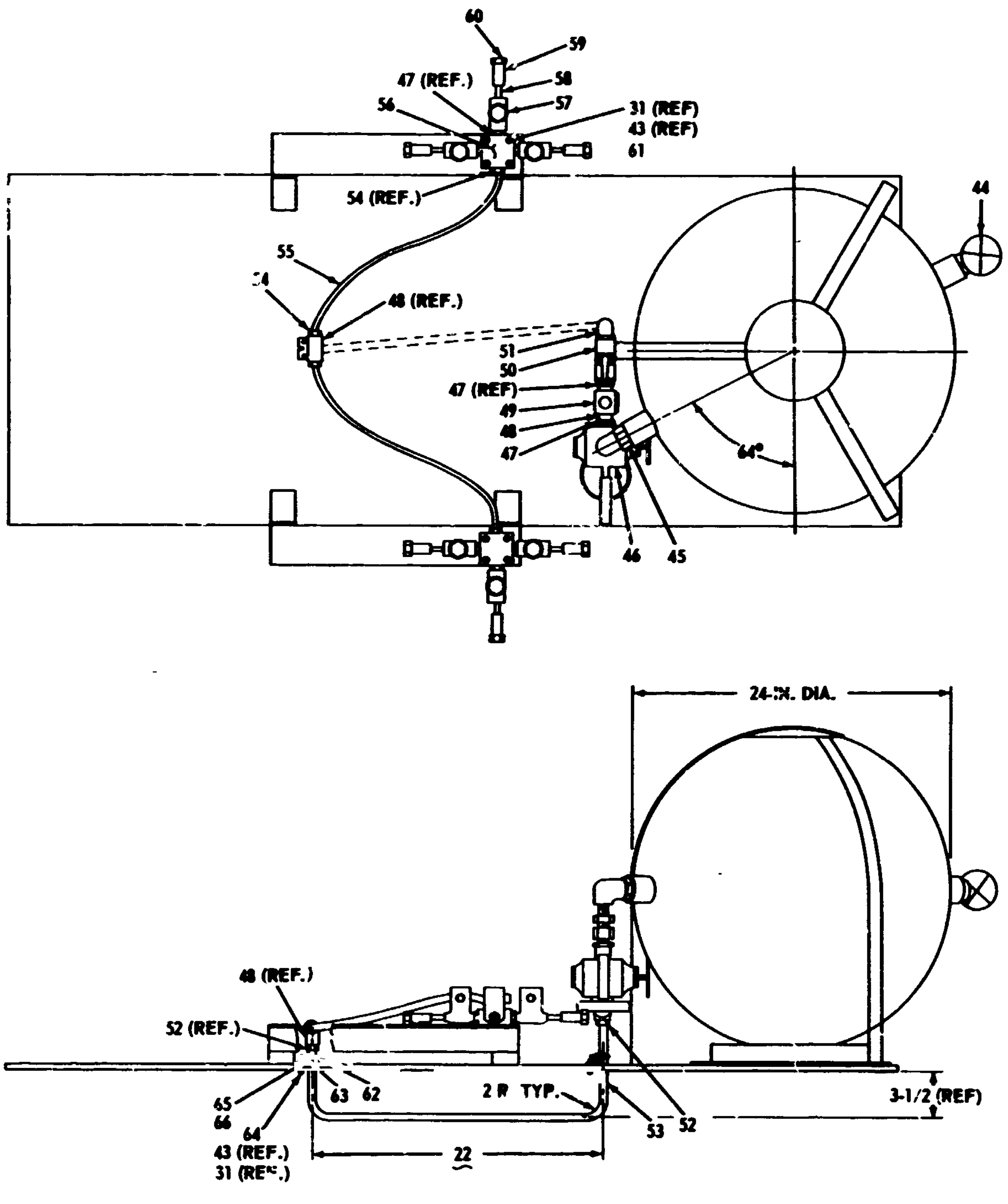


FIGURE 15. THRUSTER SYSTEM FOR AIR BEARING CAPTURE AND TRANSFER SIMULATOR

Air Supplies. There are at least three ways of supplying air at medium pressures. They are the following: mounting a high pressure {2063 N/cm^2 (3000 psig)} sphere and regulating equipment on the device, mounting an air compressor on the device, or attaching a medium pressure air line to the device.

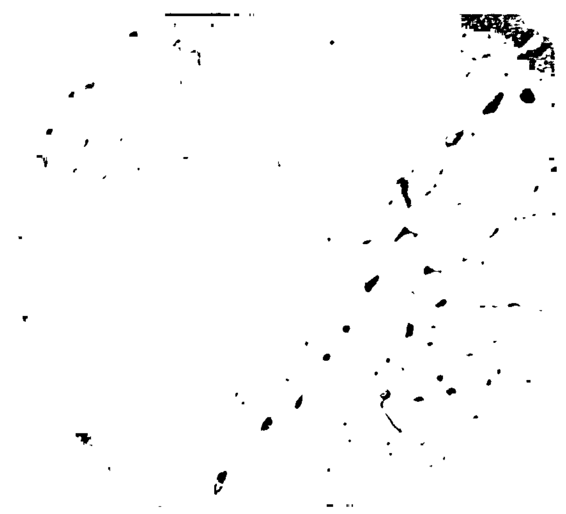


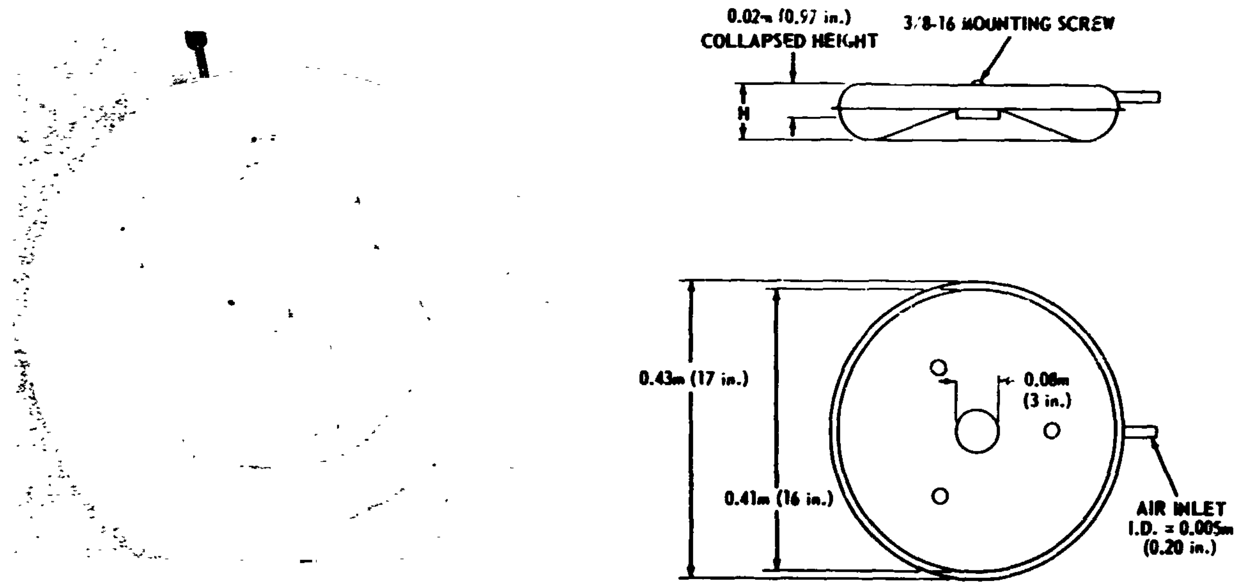
FIGURE 16. LOW INLET PRESSURE AIR BEARING PAD MADE BY MARTIN-MARIETTA CORPORATION

LOW INLET PRESSURE AIR BEARING SYSTEM

Air Bearing Pads. A low inlet pressure pad designed by the Martin-Marietta Company, Baltimore Division, is shown in Figure 16. This cushion operates on inlet pressures and volumes of air obtainable with vacuum cleaner motors described in subsequent paragraphs.

Air Supplies. The two types of air supplies that have been used are described below.

Lamb Electric Model 115250. This is a two-stage direct air flow vacuum motor for domestic canister and tank type vacuum cleaners (Fig. 19). Since the motor is cooled by discharge of the vacuum air from the blower section, this type of unit is not suitable for use in applications where the air flow could be sealed



Specifications

Diaphragm Material	**3032-50 Urethane**	**Support Area @ 5.2 psig**	**192 in.^{2}**
Diaphragm Thickness	**0.050 in. Nominal**	**Seal Perimeter @ 5.2 psig**	**4.10 ft**
Top Plate Material	**Steel**	**Air Inlet**	**1/4 in. Tubing**
Top Plate Thickness	**0.060 in.**		

FIGURE 17. HOVAIR MEDIUM INLET PRESSURE AIR BEARING PAD MADE BY THE INLAND DIVISION OF GENERAL MOTORS CORPORATION

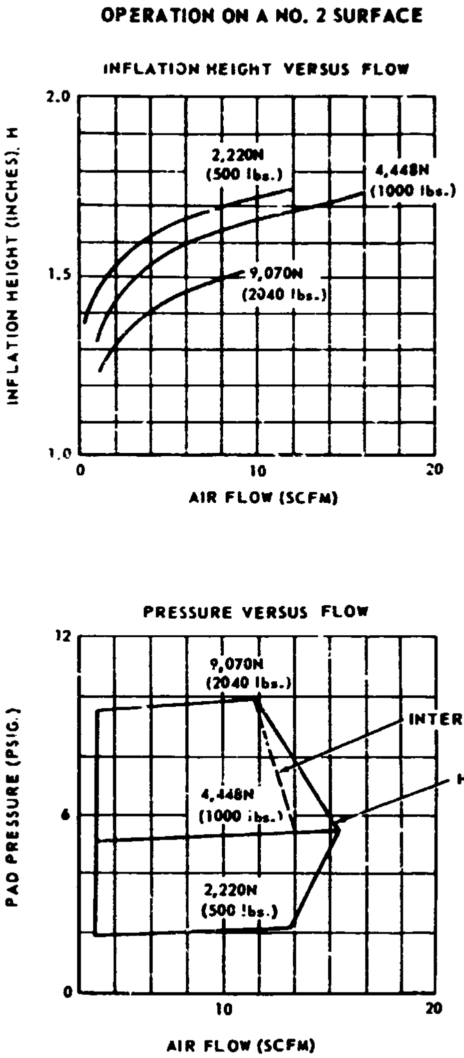


FIGURE 18. LOAD MAP FOR HOVAIR XD 16009 AIR BEARING PAD

off for an appreciable length of time. The motor performance curves are shown in Figure 20. Curve "A" shows the characteristics of a vacuum cleaner which was designed for Lamb's older Model 1S-14750 but has been replaced with Model 115250. Vacuum cleaners having the performance specified in Curve "B" draw the maximum amount of power allowable for #18 SV line cord when using Model 115250.

Black & Decker Manufacturing Company Model 820EDA. This is a heavy-duty central cleaning system 1680 W (2-1/4 h.p. maximum) unit, for medium size homes and apartments (Fig. 21). Since the motor is cooled separately, this type of unit is suitable for use in applications where the air flow could be sealed off.

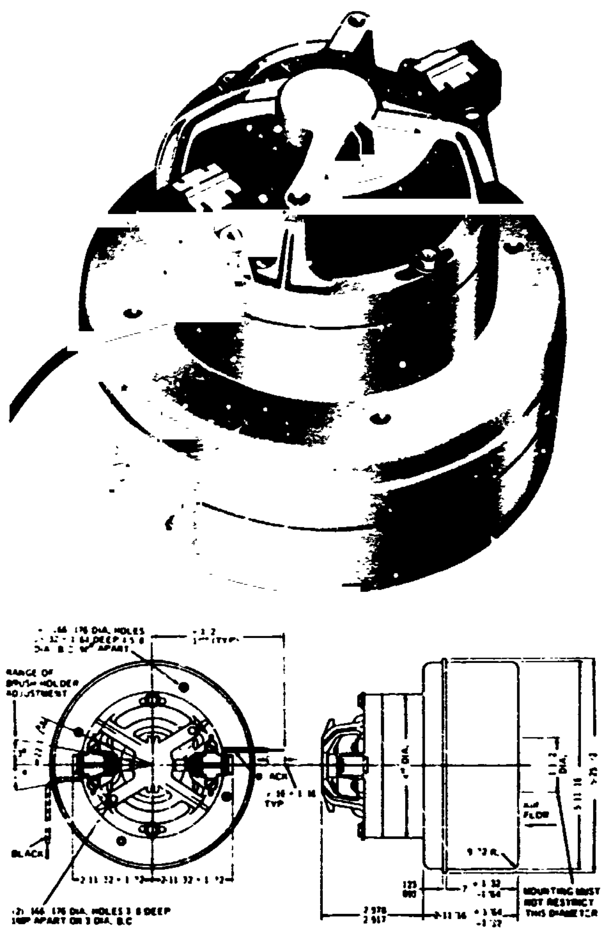


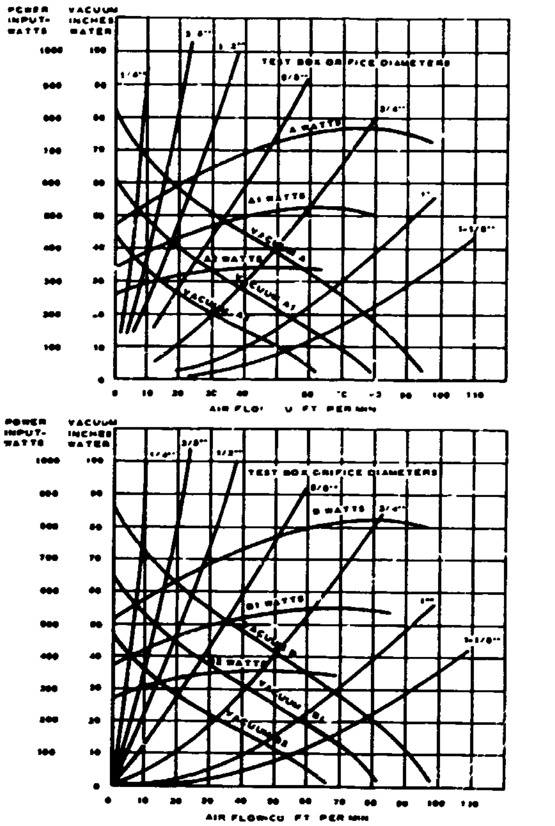
FIGURE 19. LAMB ELECTRIC MODEL 115250 TWO-STAGE DIRECT AIR FLOW VACUUM MOTOR FOR DOMESTIC CANISTER AND TANK TYPE VACUUM CLEANERS

COMPARISON OF MEDIUM AND LOW INLET PRESSURE AIR BEARING SYSTEMS

Mass. Assuming the air supply is mounted on the device, the mass of the low pressure system is lesser of the two. The lower the mass of simulator supporting the subject or work piece, the more accurate the simulation data will be.

Force. When medium pressure shop air lines are attached to the device, the forces required to produce translation are increased, thereby degrading it for simulation purposes.

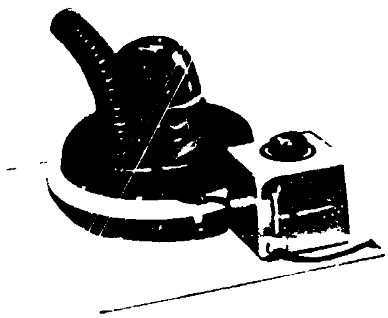
Floor Finish. A smoother floor is required for medium inlet pressure pads (Fig. 17) than for low inlet pressure pads (Fig. 16).



AVERAGE TEST DATA CORRECTED TO STANDARD BAROMETER OF 29.92 IN. HG AND 60°F.

NOTE: CURVES MARKED WITH FRACTIONAL INCH DESIGNATIONS INDICATE AIR FLOW AND VACUUM THROUGH SHARP-EDGED THIN PLATE TEST ORIFICES OF DIAMETER INDICATED.

FIGURE 20. LAMB ELECTRIC MODEL 115250 VACUUM MOTOR PERFORMANCE CURVES



CENTRAL CLEANING SYSTEM UNIT

Horsepower Rating Max.	2 1/4
R.P.M. Rated	12,000
Amperage	10
Voltage	115
Current	AC
Cycles	50-60
Inches sealed water lift	72
Inches Mercury	5.32
C.F.M.	118
Fans	2 stage
Cooling System	by-pass

FIGURE 21. BLACK & DECKER MODEL 820 EDA HEAVY DUTY CENTRAL CLEANING SYSTEM VACUUM MOTOR AND VARIAC SPEED CONTROL

FIGURE 22. METAL TUBING AND FITTINGS USED WITH LOW OR MEDIUM INLET PRESSURE AIR SUPPLY (NOTE BALL VALVE)

FIGURE 23. NONMETALLIC TUBING AND FITTINGS USED WITH LOW INLET PRESSURE AIR SUPPLY

Plumbing. The plumbing required for a medium inlet pressure system costs more than for a low inlet pressure system. Nonmetallic water pipe and fittings can be used with the latter system (compare Figs. 22 and 23).

MANIFOLD AND AIR SUPPLY RESERVOIRS

The frame of the device may be used for both medium and low pressure systems. This produces a significant weight savings.

REGULATION OF THE VOLUME OF AIR SUPPLIED TO EACH AIR BEARING PAD FOR MEDIUM AND LOW PRESSURE SYSTEMS

Ball Valves. Metallic and nonmetallic ball valves can be used with either system (Figs. 22 and 23).

Bleed Off Valve. See Figure 24.

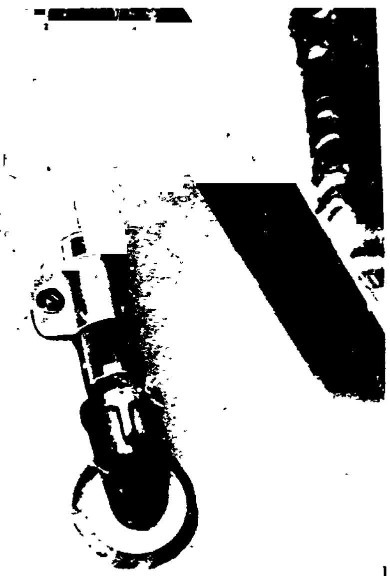


FIGURE 24. BLEED-OFF VALVE USED WITH AIR SUPPLY MOTORS THAT ARE COOLED BY DISCHARGE OF THE VACUUM AIR FROM THE BLOWER SECTION

Comparison of Ball and Bleed Off Valves. Commercially available ball valves generally cost less than the labor and materials required to make bleed off valves. Bleed off valves (Fig. 24) should be used with vacuum cleaner motors that are cooled by the air discharge from the blower section.

AIR BEARING CART THRUST SYSTEM DEVELOPMENT

1. General Information

The design criteria for this system were the following:

Cart Mass. The total mass of the cart less operator and item to be tested was estimated to be 168 kg (11.5 slugs). The masses of the operator and item to be tested were estimated to be 81.3 and 68.2 kg (5.60 and 4.66 slugs) respectively. A total mass of 318 kg (21.8 slugs) for the cart, operator and item to be tested was used for design purposes.

Velocity. The cart is to reach a velocity of at least 1.52 m/sec (5 ft/sec) in less than 6.1 m (20 ft).

Propellant. Select a propellant that has the best specific impulse for the following characteristics: easy to handle, readily available, inexpensive, requires no protective equipment for handling or use, and the exhaust products must not be toxic or nauseating.

Utilities. No propellant lines were to be attached to the cart during operation. One 120 Vac power cable could be attached to the cart during operation to run the vacuum cleaner motor, thruster control system, and item being tested.

Miscellaneous. Select inexpensive, lightweight, commercially available hardware and use it in such a manner as to produce a reliable minimum-maintenance system.

2. Technical Information

Test Setup. The apparatus shown in Figure 25 was designed to test different propellants. Its major components are a high pressure sphere 2, regulator 8, relief valve 23, ball valve 11, solenoid valve 17, and the candidate thruster 18. This apparatus was attached to a platform scale to determine thrust levels for different propellants and thruster designs. The

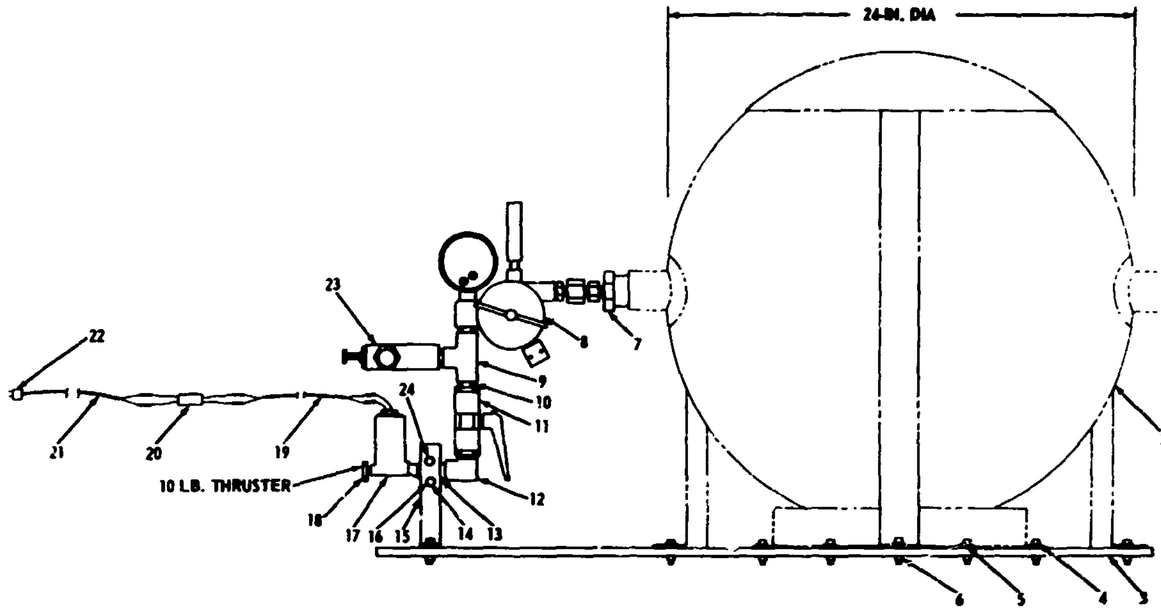


FIGURE 25. TEST SETUP USED FOR SELECTION OF PROPELLANT AND THRUSTER

center line of the thruster was perpendicular to the platform with the exit nozzle pointing upward. For determining the velocity and acceleration obtainable with each propellant, the apparatus was attached to a test cart (Fig. 26).

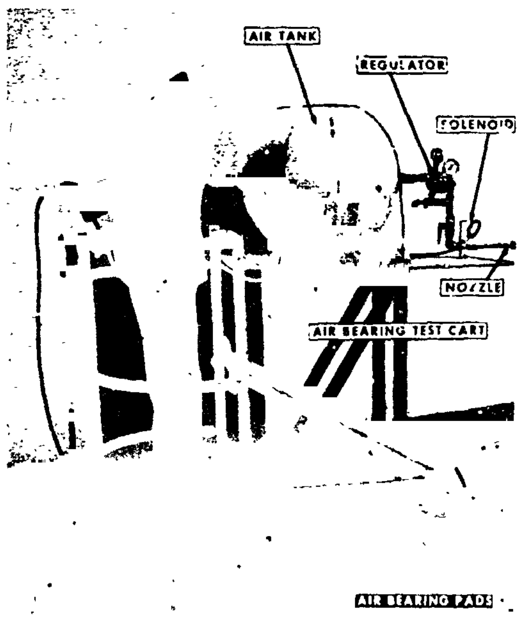


FIGURE 26. TEST SETUP USED TO DETERMINE VELOCITY AND ACCELERATION OBTAINABLE WITH EACH PROPELLANT AND THRUSTER

Test Results. Air was the first and only propellant tested. It produced the desired results after several thruster designs and tubing configurations were tried. The thruster shown in Figure 27 produced the maximum thrust once the flow into the inlet nozzle was changed to laminar. This thruster has inlet and outlet nozzle areas equal to 4 and 2 times the throat area, respectively. The throat area is 3.38×10^{-5} m^2 (0.0523 in.^{2}). In the earlier tests the thruster 18 was screwed into the outlet of the solenoid valve 17 as shown in Figure 25. The solenoid valve passage caused the flow to be turbulent. This turbulence decreased the maximum thrust of the thruster.

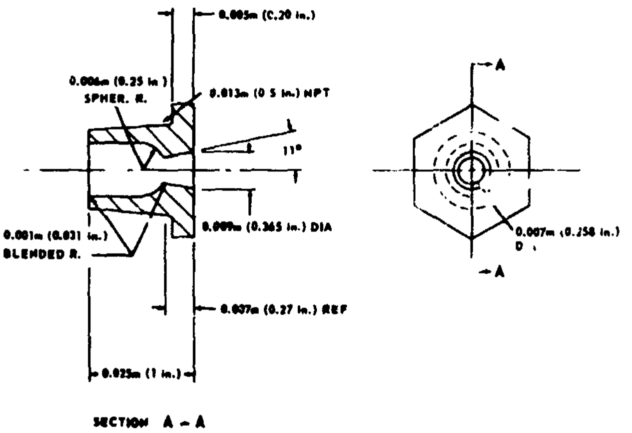


FIGURE 27. THRUSTER SELECTED FOR AIR PROPULSION SYSTEM

The flow into the inlet nozzle of the thruster was changed from turbulent to laminar by putting a pipe nipple 10 times the pipe diameter between the solenoid valve and thruster as shown in Figure 26. Table I data were obtained with the apparatus shown in Figure 25 mounted on the platform scale and a 0.127 m (5 in.) pipe nipple between the solenoid and thruster.

DEVICES TO SUPPORT EQUIPMENT FOR MECHANICAL SIMULATION

1. General Information

The weight of equipment, e.g., hand tools, which a subject uses to perform tasks, cannot be effectively balanced with a simulator balancing system because as soon as equipment is moved, the subject is out of balance. If the subject is balanced without the equipment and then handed the equipment he is out of balance too. To remedy this situation the equipment may be supported with a helium filled balloon. The balloon can be filled to produce a force equal to the weight of the equipment for earth orbital gravity simulation, or 5/6 its weight for lunar gravity simulation.

Equipment designed for earth orbital or lunar gravity work will not necessarily function in an earth gravity, one "g", environment because of its mass, deflections caused by cantilevered members, etc. One way of solving this problem is to support the equipment with a pedestal attached to the platform supported by air bearing pads. These devices are called Air Bearing Platforms.

2. Technical Information

Balloons. In Figure 28 an impact wrench is supported by a 1.83 m (6 ft) diameter weather balloon filled with sufficient helium to provide an upward force equal to the weight of the wrench.

The balloons presently used were made especially for simulation work by the G. T. Schjildahl Company, Northfield, Massachusetts. These balloons are 2.54 m (8 ft) in diameter consisting of 12 gores of two layers of 25.4 μm (0.001 in.) thick polyester bilaminate with 2 plastic inflation valves 180 degrees apart. These balloons will lift a mass up to 8.2 kg (0.56 slugs).

Air Bearing Platform. These platforms are designed to provide a stable base for simulating a near-frictionless environment by means of air bearing pads operating on a smooth level floor. Two configurations of the platform are presently being used: the manned

FIGURE 28. HELIUM FILLED WEATHER BALLOON SUPPORTING THE IMPACT WRENCH

version including a man-seat pedestal and serpentuator attach bracket (Fig. 29), and the unmanned version that includes the serpentuator support bracket (Fig. 30). The components that make up these configurations have the following masses: platform air bearing, 25 kg (1.71 slugs); serpentuator attach post and

TABLE I. AIR BEARING CART TEST THRUSTER ASSEMBLY THRUST TESTS

OCTOBER 13, 1967

TEST NUMBER	TIME CYCLE (sec)	SET REGULATOR AT 215 psi	DYNAMIC PRESSURE 150 psi TIMED	SPHERE PRESSURE PRIOR TO TEST	SPHERE PRESSURE AFTER TEST	INITIAL WEIGHT OF SPHERE PRIOR TO TEST (AT 1 g)	TOP READING OF BLAST TEST	BOTTOM READING OF BLAST TEST	THRUST AT TOP READING	THRUST AT BOTTOM READING	WEIGHT OF SPHERE AFTER TEST (AT 1 g)	ROOM TEMPERATURE
	sec	psi	psi	psi-N_2	psi-N_2	lbf	lbf	lbf	lbf	lbf	lbf	°F
1	7	215	150	2500	2400	179.50	188.00	187.50	8.50	8.00	178.50	68
2	10	215	150	2500	2375	179.50	188.00	186.75	8.50	6.75	178.00	69
3	12	215	150	2500	2350	179.25	188.00	186.50	8.75	7.25	177.75	70
4	15	215	150	2500	2325	179.50	188.00	186.00	8.50	6.50	177.25	73
5	5	215	150	2500	2450	178.50	188.00	187.50	8.75	8.25	178.50	72

OCTOBER 20, 1967

TIME START: 10:00 a.m. TIME END: 11:30 a.m.

TEST NUMBER	TIME CYCLE	SET REGULATOR AT 215 psi	DYNAMIC PRESSURE TIMED	SPHERE PRESSURE PRIOR TO TEST	SPHERE PRESSURE AFTER TEST	TOP READING OF BLAST TEST	BOTTOM OF BLAST TEST	THRUST AT TOP READING	THRUST AT BOTTOM READING	WEIGHT OF SPHERE BEFORE TEST (AT 1 g)	WEIGHT OF SPHERE AFTER TEST (AT 1 g)	ROOM TEMPERATURE	SPHERE TEMPERATURE	DATA RECORDING TIME
	sec	psi	psi	psi-N_2	psi-N_2	lbf	lbf	lbf	lbf	lbf	lbf	°F	°F	min
1	5	215	150	2500	2425	189.75	187.50	11.25	9.00	178.50	178.00	72.0	78.0	6
2	5	215	150	2425	2360	188.75	186.50	10.75	8.50	178.00	177.00	71.0	76.0	3
3	5	215	150	2360	2300	187.75	185.25	10.75	8.25	177.00	176.25	71.5	76.0	2
4	5	215	150	2300	2225	187.00	184.50	10.75	8.25	176.25	175.25	72.0	75.0	1
5	5	215	150	2225	2175	186.50	183.50	11.25	8.25	175.25	174.50	72.0	75.0	1
6	5	215	150	2175	2100	185.25	182.75	10.75	8.25	174.50	173.50	71.5	75.0	1
7	5	215	150	2100	2050	184.50	181.75	11.00	8.25	173.50	172.75	71.5	74.0	1
8	5	215	150	2050	2000	183.25	181.00	10.50	8.50	172.75	171.75	71.5	73.5	1
9	5	215	150	2000	1950	182.25	181.00	10.50	9.25	171.75	170.75	71.0	73.0	1
10	5	215	150	1950	1875	181.50	179.00	10.75	8.25	170.75	170.00	71.0	72.0	1
11	5	215	150	1875	1825	180.75	178.00	10.75	8.00	170.00	169.00	71.0	71.5	1
12	5	215	150	1825	1775	179.75	177.00	10.75	8.00	169.00	168.25	71.5	70.5	1
13	5	215	150	1775	1710	178.75	176.50	10.50	8.25	168.25	167.25	72.0	70.0	1
14	5	215	150	1710	1650	177.75	175.00	10.50	7.75	167.25	166.25	72.0	69.0	1
15	5	215	150	1650	1600	176.50	174.00	10.25	7.75	166.25	165.25	72.0	69.0	1
16	5	215	150	1600	1550	175.75	173.25	10.50	8.00	165.25	164.50	71.0	68.0	1
17	5	215	150	1550	1510	174.75	172.50	10.25	8.00	164.50	163.75	72.0	67.0	1
18	5	215	150	1510	1470	173.75	171.50	10.00	7.75	163.75	162.75	72.0	67.0	1
19	5	215	150	1470	1400	173.00	170.50	10.25	8.75	162.75	162.00	72.0	67.0	1

TABLE I. AIR BEARING CART TEST THRUSTER ASSEMBLY THRUST TESTS (Concluded)

TEST NUMBER	TIME CYCLE	SET REGULATOR AT 215 psi	DYNAMIC PRESSURE TIMED	SPHERE PRESSURE PRIOR TO TEST	SPHERE PRESSURE AFTER TEST	TOP READING OF BLAST TEST	BOTTOM OF BLAST TEST	THRUST AT TOP READING	THRUST AT BOTTOM READING	WEIGHT OF SPHERE BEFORE TEST (AT 1 g)	WEIGHT OF SPHERE AFTER TEST (AT 1 g)	ROOM TEMPERATURE	SPHERE TEMPERATURE	DATA RECORDING TIME
	sec	psi	psi	psi-N_2	psi-N_2	lbf	lbf	lbf	lbf	lbf	lbf	°F	°F	min
20*	5	215	149	1400	1375	172.25	169.75	10.25	7.75	162.00	161.00	72.0	65.0	1
[illegible]	5	215	149	1375	1310	171.25	169.00	10.25	8.00	161.00	160.00	72.0	75.0	1
22	5	215	148	1310	1280	170.50	168.00	10.50	8.00	160.00	159.25	72.0	64.0	1
23	5	215	148	1280	1210	169.50	167.00	10.25	7.75	159.25	158.25	72.0	64.0	1
24	5	215	147	1210	1200	168.50	166.00	10.25	7.75	158.25	157.50	72.0	63.0	1
25	5	215	147	1200	1150	167.75	165.25	10.25	7.75	157.50	[illegible]	72.0	73.0	1
26	5	215	146	1150	1100	167.00	164.50	10.50	8.00	156.50	155.75	72.0	62.0	1
27	5	215	145	1100	1050	165.75	163.50	10.00	7.75	155.75	155.00	72.0	61.0	1
28	5	215	144	1050	1010	165.00	162.50	10.00	7.25	155.00	[illegible]	72.0	60.0	1
29	5	215	144	1010	1000	164.00	161.75	10.00	7.75	154.00	153.25	72.0	59.0	2
30	5	215	143	1000	960	163.00	160.75	9.75	7.50	153.25	152.25	72.0	59.0	1
31	5	215	142	960	910	162.00	160.00	9.75	7.75	152.25	151.50	72.0	58.0	1
32	5	215	142	910	890	161.25	159.00	9.75	7.50	151.50	150.75	71.5	58.0	1
33	5	215	141	890	850	160.25	158.00	9.50	7.25	150.75	150.00	71.5	58.0	1
34	5	215	135	850	800	159.50	156.75	9.50	6.75	150.00	149.25	72.0	57.0	1
35	5	215	135	800	770	159.25	155.75	9.00	6.50	149.25	148.50	72.0	57.0	1
36	5	215	130	770	750	157.50	155.00	9.00	6.50	148.50	147.75	72.0	57.0	1
37	5	215	130	750	700	156.50	154.25	8.75	6.50	147.75	147.00	72.0	57.0	1
38	5	215	125	700	680	155.75	153.25	8.75	6.23	147.00	146.25	72.5	57.0	1
39	5	215	120	680	650	155.00	152.50	8.75	6.25	146.25	145.50	72.5	57.0	1
40	5	215	115	650	600	154.00	151.50	8.50	6.00	145.50	144.75	72.5	56.5	1
41	5	215	115	600	580	153.00	150.75	8.25	6.00	144.75	144.00	70.0	56.0	1
42	5	215	115	580	525	152.25	150.00	8.25	6.00	[illegible]	[illegible]	69.0	55.0	1
43	5	215	110	525	510	151.50	149.00	8.25	5.75	143.25	142.75	68.0	55.0	1
44	5	215	110	510	500	150.50	148.25	7.75	5.50	142.75	142.00	68.0	54.0	1
45	5	215	110	500	480	149.75	147.50	7.75	5.50	142.00	141.25	68.0	54.0	1
46	5	215	105	480	425	149.00	146.75	7.75	5.50	141.25	140.75	69.0	54.0	1
47	5	215	100	425	400	147.75	145.50	7.00	4.75	140.75	140.00	70.0	54.0	1
48	5	215	[illegible]	400	390	147.25	145.00	7.25	5.00	140.00	139.50	70.0	54.0	1
49	5	215	85	390	350	146.25	144.25	6.75	4.75	139.50	138.75	71.0	54.0	1
50	5	215	80	350	310	145.50	143.25	6.75	4.50	138.75	138.25	72.0	54.0	1
51	5	215	80	310	300	144.50	142.50	6.25	4.25	138.25	137.75	[illegible]	54.0	1
52	5	215	80	300	290	143.75	141.75	6.00	4.00	137.75	137.25	72.0	54.0	1
53	5	-	75	290	250	142.75	141.00	5.50	3.75	137.25	136.75	72.0	54.0	1
54	5	-	70	250	220	142.00	140.25	5.25	3.50	136.75	136.25	72.0	54.0	1
55	5	-	65	220	210	[illegible]	[illegible]	[illegible]	3.00	136.25	135.75	72.0	54.0	1
56	5	-	60	210	200	[illegible]	138.50	4.50	2.75	135.75	135.50	72.0	54.0	1

*Sphere started to frost up.

1 g weight of sphere was measured at AEC Oak Ridge Tenn. facility

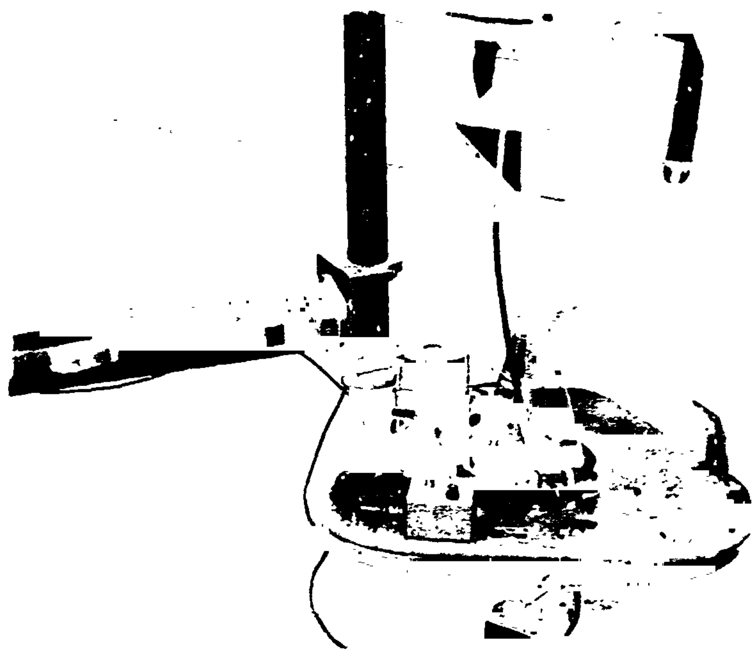


FIGURE 29. MANNED VERSION OF THE AIR BEARING PLATFORM

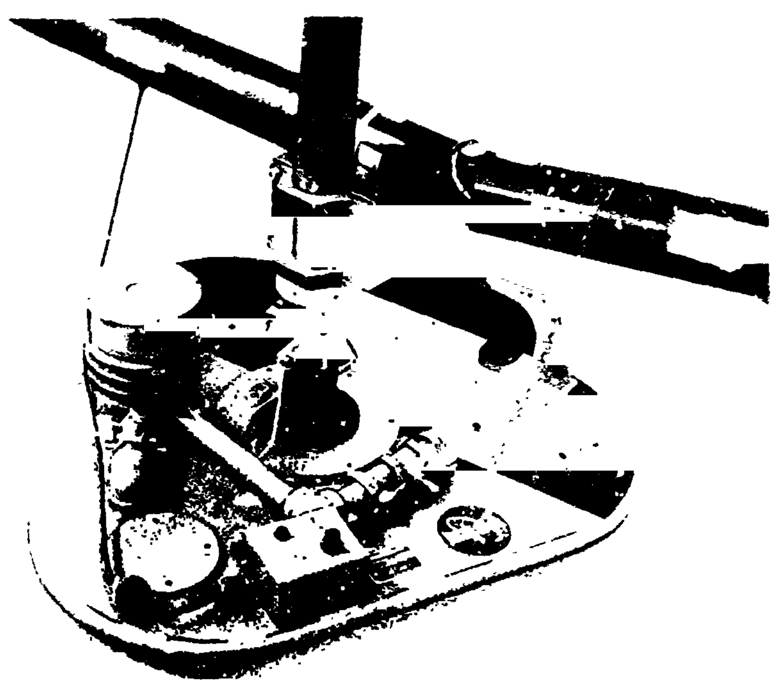


FIGURE 30. UNMANNED VERSION OF THE AIR BEARING PLATFORM

bracket, 4.45 kg (0.373 slugs); serpentuator support, post and bracket, 4.45 kg (0.373 slugs); and the operator's mast and safety yoke, 8.18 kg (0.56 slugs). Each platform will support a nominal load of 273 kg (18.7 slugs). The platform flotation lifting height can be adjusted by controlling the speed of the air supply motor. When unbalanced loads are supported by the platform, it can be leveled by manually adjusting valves which restrict the air flow delivered to each pad. The air supply motor is protected by a 10 A combination circuit breaker/toggle switch mounted in the electrical control box. The unmanned speed control is also contained in the electrical control box. On the manned version the air supply motor runs as long as the foot-treadle is depressed, and the air supply speed control is mounted on the operator's seat yoke. The air supply is capable of maintaining a plenum pressure of 17 220 N/m^2 (2.5 psig) for flows up to 0.438 standard m^3/min (15.5 scfm). The only power requirement is single phase 115 Vac 60 Hz (60 Cps) with maximum current of 10 A.

Application. In Figure 31 the serpentuator and operator are supported by the air bearing platforms described above.

FIGURE 31. AIR BEARING PLATFORMS SUPPORTING SERPENTUATOR AND OPERATOR

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SIMULATION

The main requirement of this simulation project, after programming the system of equations, is to make provision for human logic. The necessary corrective decisions, i.e., when are the thrusters to be turned on and off, must be simulated. This was done through three logic tools: control variables, generated functions, and phases. Each will be defined and an example of their combined use given.

Control Variables. A control variable is a continuous comparison of a parameter of motion with a preassigned constant. The control variable is used to define a condition requiring a specific thrust action. For example, a control variable is used to compare the pitch with a preassigned limit. When the pitch angle becomes greater than the prescribed value, the control variable indicates a need for corrective thrust.

Generated Functions. A generated function is a rule of correspondence between two variables and is set prior to the beginning of a maneuver. The function, usually a combination of step functions, is used to establish a variable physical limit on a parameter of motion. For example, each component of velocity can be continually compared with a preselected function of the distance translated. The generated function is then used to define conditions which demand thrust action to control that velocity.

Phases. The third logic tool, a phase, is a state in which the parameters of motion demand a certain action. Examples of phases are acceleration, coast, and deceleration. The phases also allow coordination between unrelated planes of action. The forward velocity can be interconnected with the lateral translation at any distance. This is demonstrated in the following example.

Example of Command Logic. The three tools described above are combined through Boolean algebra to define all conditions requiring thrust action. An example of the use and combination of the three logic tools in programming a 50-foot translation maneuver follows.

(1) Control variables are defined for all angular displacements, angular velocities, and linear velocities. A few examples are:

Control Variable	Controlled Action
$L\ 1 = \phi > \alpha$	roll right
$L\ 2 = \phi < -\alpha$	roll left
$L\ 3 = \theta > \beta$	pitch up
$L\ 4 = \theta < -\beta$	pitch down
$L\ 5 = \psi > \gamma$	yaw right
$L\ 6 = \psi < -\gamma$	yaw left
$L\ 7 = \dot{\theta} > \dot{\beta}$	pitch up rate
$L\ 8 = \dot{\theta} < -\dot{\beta}$	pitch down rate

For most of this study, the following limits were used: Rotation limits α, β, and γ of .26 radians: rotation rate limits $\dot{\alpha}$, $\dot{\beta}$, and $\dot{\gamma}$ of .06 radians per second; and lateral and vertical velocity limits of 0.5 feet per second.

New control variables can be defined through combination of others, using Boolean algebra notation. "•" means "and"; "+" means "or"; and

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SIMULATION OF UNSTABILIZED MANEUVERING UNITS*

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SUMMARY: A method for predicting the performance of unstabilized maneuvering units has been developed. The equations of motion giving the response of man and maneuvering unit to the thrust produced were written for any unstabilized unit and programmed for an analog computer. The command decisions required of the astronaut were simulated with a hybrid computer by providing corrective thrust whenever certain combinations of the parameters of motion exceed preassigned limits.

INTRODUCTION

It is expected that extravehicular activity will become an increasingly important part of the program for space exploration. Two distinct forms of propulsive devices, stabilized and unstabilized units, are envisaged. The stabilized maneuvering unit includes a gyro-stabilized attitude control which will permit traveling considerable distances, perhaps up to a mile. The unstabilized maneuvering unit contains no attitude control and will permit only local travel, perhaps to 50 feet from the spacecraft.[1]

The purpose of this investigation was to develop an analog computer program which could simulate various types of unstabilized extravehicular activity maneuvering units, using as program input only three properties of the combined astronaut-maneuvering unit system: (1) the mass and center of mass, (2) the moments and products of inertia, and (3) the thrust force and moment vectors. Hybrid analog-digital computer equipment was then used to record the dynamic response during specific simulated maneuvers.

Desired output data included time plots of (1) translational velocity and displacement along three axes, (2) angular velocity and displacement in pitch, roll and yaw, (3) an on-off thrust profile, and (4) a cumulative record of fuel used. From such data the capabilities of individual units can be analyzed and compared.

* This paper is based on a thesis submitted by the first author in partial fulfillment of the requirements for the Master of Science degree at the Air Force Institute of Technology. The views expressed herein are those of the author and do not necessarily reflect the views of the Air University, the Department of Defense or the United States Air Force.

DEVELOPMENT OF GOVERNING EQUATIONS

The combined system of astronaut, maneuvering unit, and environment control system is assumed to respond as a single rigid body. Photographs of extravehicular activity indicate that the posture of the astronaut does not vary greatly during the maneuver. Space suit pressurization of 3.5 psi makes the suit quite rigid during motions characterized by such low accelerations.[1]

The motion of the astronaut is referred to an axis system, X_i, fixed to the orbiting spacecraft. Treating this moving reference frame, X_i, as an inertial frame introduces errors in acceleration which are less than 1% of the accelerations possible with typical maneuvering units. This approximation allows an appreciable simplification in the set of equations which must be integrated. Another set of axes, x_i, having origin at the center of mass of the astronaut-maneuvering unit system are fixed in the body and rotate with it. In order to provide for a logical sequence of rotation corrections by the astronaut, a modified set of Euler angles, ϕ, θ and ψ are defined. The description of body position is then made through the following sequence: beginning with the x_i and X_i axes parallel, (1) yaw ψ degrees about the x_3 axis, (2) pitch θ degrees about the x_2 axis and (3) roll ϕ degrees about the x_1 axis. To realign the body axes parallel with the reference axes, rotational corrections must take place in the opposite sequence. In other words, (1) a roll about the x_1 axis until the x_2 axis is restored to the X_1-X_2 plane, (2) pitch about the x_2 axis to restore the x_1 axis to the X_1-X_2 plane and (3) yaw about the x_3 axis to return to original alignment. All rotation corrections in the maneuvers programmed in this study were made in this sequence. This particular sequence was chosen arbitrarily, but some such choice must be made.

The translational equations give the change in velocity of the system center of mass due to the thrust of the maneuvering unit.

$$\dot{\vec{V}} = \frac{\vec{F}}{m} \tag{1}$$

The mass, m, is that of the entire system, man plus maneuvering unit and other attached equipment. Since the maneuvers to be programmed are short and use limited fuel, no provision will be made for the change in mass resulting from fuel consumption. The velocity $\vec{V}$ is relative to the origin of the reference frame. As the force $\vec{F}$ is known in terms of the body axes, writing the components of Equation 1 introduces the Euler angles. If V_1, V_2 and V_3 are the components of velocity with respect to the reference frame, and f_1, f_2 and f_3 are the components of thrust with respect to the body fixed frame, we find

$$\dot{V}_1 = \frac{1}{m}\begin{bmatrix} \cos\theta \cos\psi \, f_1 \\ +(\sin\theta \sin\phi \cos\psi - \cos\phi \sin\psi) f_2 \\ +(\sin\theta \cos\phi \cos\psi + \sin\phi \sin\psi) f_3 \end{bmatrix} \tag{2a}$$

$$\dot{V}_2 = \frac{1}{m}\begin{bmatrix} \cos\theta \sin\psi \, f_1 \\ +(\sin\theta \sin\phi \sin\psi + \cos\phi \cos\psi) f_2 \\ +(\sin\theta \cos\phi \sin\psi - \sin\phi \cos\psi) f_3 \end{bmatrix} \tag{2b}$$

$$\dot{V}_3 = \frac{1}{m}\begin{bmatrix} -\sin\theta \, f_1 + \cos\theta \sin\phi \, f_2 \\ + \cos\theta \cos\phi \, f_3 \end{bmatrix} \tag{2c}$$

Equating the total moment (about the center of mass) to the time rate of change of angular momentum

$$\vec{M} = \frac{d}{dt}(\vec{\vec{I}}\cdot\vec{\omega}) = \vec{\vec{I}}\cdot\dot{\vec{\omega}} + \vec{\omega} \times (\vec{\vec{J}}\cdot\vec{\omega}) \tag{3}$$

where $\vec{\omega}$ is the angular velocity of the body, and $\vec{\vec{I}}$ is the inertia dyadic computed in body axes. Introducing the inverse of the inertia dyadic, $\vec{\vec{A}}$, leads to Euler's moment equations:

$$\dot{\vec{\omega}} = \vec{\vec{A}} \cdot [\vec{M} - \vec{\omega} \times (\vec{\vec{I}} \cdot \vec{\omega})] \tag{4}$$

The equations relating the rates of change of the Euler angles to the angular velocities are

$$\dot{\phi} = \omega_1 + \tan\theta \sin\phi \ \omega_2 + \tan\theta \cos\phi \ \omega_3 \tag{5a}$$

$$\dot{\theta} = \cos\phi \ \omega_2 - \sin\phi \ \omega_3 \tag{5b}$$

$$\dot{\psi} = \sec\theta \sin\phi \ \omega_2 + \cos\phi \sec\theta \ \omega_3 \tag{5c}$$

where the components of the angular velocity vector, ω_i, are with respect to the body frame. It should be noted that the pitch, θ, must be less than $\pi/2$. In summary, the nine equations to be integrated are: three equations (2a, 2b, and 2c) for the translational velocity, three equations (4) for the components of rotational velocity, and three equations (5a, 5b, and 5c) for the Euler angles. These, together with the required maneuvering unit properties -- mass, inertia, and thrust, and the necessary initial conditions for each of the unknowns determine the motion of the man-maneuvering unit system.

A block diagram, indicating the procedure to be followed in integrating this set of equations is given as Figure 1.

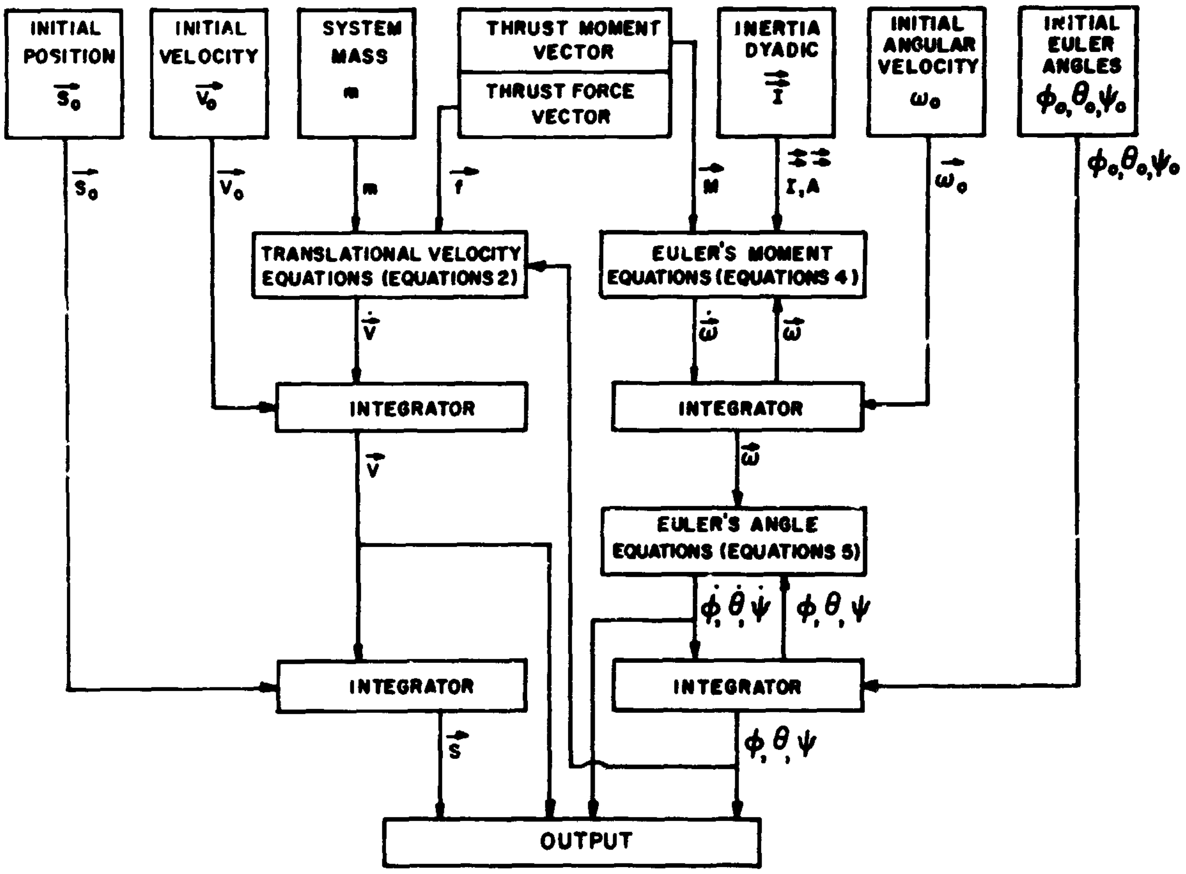

FIGURE 1. FLOW DIAGRAM OF SEQUENCE OF COMPUTATION

"'" means "not." Such combinations are L 21 = L 1 + L 2 + L 3 + L 4 and L 22 = L 21' · L 6' · L 5'. Thus, L 21 is true if excessive roll or pitch does exist, and L 22 is true if excessive roll, pitch or yaw does not exist.

(2) Functions used to limit forward velocity and lateral and vertical translations are defined. Typical functions, F1, F2, F3, and F4 are shown in Figure 2. These functions are generated by the computer.

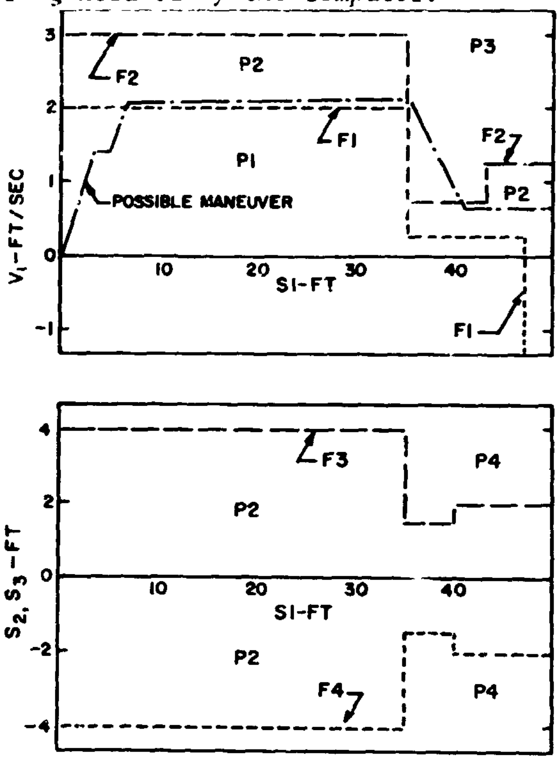

FIGURE 2. TYPICAL FUNCTIONS AND PHASES

(3) Phases are defined for use in combined thrust command logic. Some examples follow.

Phase	Definition
acceleration	$P1=[(V_1<F1)\cdot P4'\cdot(S_1<35)]+[(V_1<F1)\cdot(S_1>35)]$
coast	$P2=(V_1<F2)\cdot(V_1>F1)\cdot P4'$
deceleration	$P3=[(V_1>F2)\cdot P4'\cdot(S_1<35)]+[(V_1>F2)\cdot(S_1>35)]$
vertical or lateral correction	$P4=(S_2>F3)+(S_2<F4)+(S_3>F_3)+(S_3<F_4)$
STOP	$STOP = S_1>50$

These are interpreted in the following manner.

"Phase 1 exists when V_1 is less than F 1 and when not in Phase 4 and S_1 is less than 35 feet or Phase 1 exists when V_1 is less than F 1 and S_1 is greater than 35 feet."

(4) Control variables and phases are combined into a command logic which determines, for all possible conditions, when a thruster should be turned on. For example:

Thrust Action	Symbol	Command Logic
translate forward	TF	= P1·L22
translate back	TB	= P3·L22
pitch down	PD	= L1'·L2'·L3·L8'
pitch up	PU	= L1'·L2'·L4·L7'

When conditions demand a particular action, the appropriate thrust force and moment vectors are injected into the equation system of the computer program by the command logic. The dotted line in Figure 2 shows a possible forward velocity profile of a maneuvering unit controlled by this logic.

Of special significance is the flexibility in programming allowed by the use of the logic tools described. The generated functions can be easily preset so as to impose any desired limits on a maneuvering astronaut. It can be seen from the phase definitions and the command logic that simultaneous corrections can be allowed or prohibited as desired. In this example, simultaneous forward or back, vertical and lateral translation is permitted in the last 15 feet of the maneuver but vertical and lateral translation takes precedence over forward or back in the first 35 feet.

The system of equations developed in Section II and the command logic were programmed by personnel of the Analog Computation Division (SESCA), Wright-Patterson Air Force Base, Ohio. A Reeves System Dynamics Simulator and an Applied Dynamics Hybrid Computer were employed.

MANEUVERING UNITS

The two maneuvering units for which data was obtained in this study are the Astronaut Maneuvering Unit (AMU) and the Hand-Held Maneuvering Unit (HHMU).

Astronaut Maneuvering Units

The Astronaut Maneuvering Unit[2] consists of a backpack propulsion unit, an environment control system chestpack, an alarm system and communications and telemetry equipment, and was designed as a stabilized unit for use in the Gemini program. Thrust required for stabilization about three axes and translation along two axes is provided by twelve thrust chambers arranged to operate in either primary or alternate mode. In order to provide for lateral translation parallel to the remaining, or x_2-axis, a thrust chamber was added to each side for this study. These extra thrusters were considered as being placed 2.7 inches behind the center of mass of the man-maneuvering unit system. In this study of unstabilized units, it is assumed that the AMU thrusters are operated only in the manual mode; no attitude stabilization is included.

The weight of the AMU is 177 pounds including 25 pounds of fuel. Additional weight is added by the 40-pound chest pack and a 25-pound space suit. A 50th percentile man of 162 pounds was chosen from available anthropometric data.[3] The total mass is 12.55 slugs.

Each thruster produces 2.3±0.2 pounds of thrust, or a maximum acceleration of 0.42 ft/sec^2. No thrust growth or decay has been provided for; nor has allowance been made for decreasing thrust as propellant is expended.

Hand-Held Maneuvering Unit

The Hand-Held Maneuvering Unit[4] is an integral thrust device that contains two pressurized gas bottles, a pressure regulator, two spring-loaded poppet valves, a rocking trigger inside the handle, and three thrust nozzles. The two cylindrical gas bottles together contain less than a pound of oxygen. For more extensive use, the propellant gas, either oxygen or nitrogen, can be supplied through the astronaut's umbilical. Each of the two forward thrust nozzles produces about one pound of thrust and the front facing nozzle produces approximately two pounds of push or braking thrust.

For this study, the umbilical configuration was chosen to allow sufficient gas supply for the programmed maneuvers. The total mass of astronaut, space suit, gun and chestpack is 7.2 slugs.

Selecting a body position for

the Hand-Held Maneuvering Unit is more arbitrary than for the Astronaut Maneuvering Unit because of the possible variation in position and method of performing a maneuver. In the position chosen, the arms are held perpendicular to the torso, the lower legs parallel to the torso and the upper legs inclined at 45°. The center of mass and the inertia matrix were computed using data given elsewhere.[5, 6]

Comparison of Performance

Table I shows the time, fuel consumption, and number of corrections made during a series of simulated 50-foot translations by the Astronaut Maneuvering Unit and Hand-Held Maneuvering Unit with various initial conditions. Simultaneous translation corrections were allowed throughout the runs for the Astronaut Maneuvering Unit, but prohibited for the Hand-Held Maneuvering Unit. A one second delay was imposed between corrections. A logic for lateral corrections which attempts to represent the recommended procedure of facing the target when translating was introduced into the simulation of the

Table I
AMU versus HHMU

$\alpha = \beta = \gamma = 15°$ except HHMU $\alpha = 30°$ — 50-foot Translation

Run	Initial Conditions	UNIT	Time (sec)	Fuel (lb_m)	Number of Corrections	Program Limits Exceeded
1	None	AMU	39.6	.491	7	
		HHMU	34.2	.425	7	S_3=3ft
2	V_3=0.5 ft/sec	AMU	38.2	.503	9	
		HHMU	26.1	.365	5	V_1=2ft/sec,S_3>7ft;θ out of limits for 8 sec.
3	V_3= -0.5 ft/sec	AMU	31.8	.476	11	
		HHMU	48.5	.468	14	Rides TF-TD limit
4	S_3= -2 ft	AMU	35.0	.491	12	
		HHMU	30.2	.450	9	
5	ω_1=0.05 rad/sec	AMU	38.2	.548	14	
		HHMU	42.0	.384	8	S_2<-5ft,S_3=3.4 ft
6	ω_2= -0.05 rad/sec	AMU	37.6	.294	4	
		HHMU	40.8	.468	12	S_3= 4.2 ft but correcting
7	ω_1=ω_2= -0.05 rad/sec	AMU	37.3	.365	12	
		HHMU	36.5	.408	10	S_2= -2.4ft,S_3=2.4 ft

Hand-Held Maneuvering Unit.

Comments on Results

(1) The one second delay between Hand-Held Maneuvering Unit thrust corrections detracts from its ability to keep angles within limits as shown in run number 2.

(2) While the Hand-Held Maneuvering Unit may appear to require fewer corrections and use less fuel, it should be noted that the Astronaut Maneuvering Unit makes more timely corrections and remains within maneuver limits. The Hand-Held Maneuvering Unit usually does not remain within the specified limits on lateral displacement (x_2-x_3 plane) to the end of a 50-foot translation. Also the Hand-Held Maneuvering Unit frequently does not provide sufficient deceleration opportunity to end the maneuver below the desired V limit of 1.25 ft/sec.

(3) Simulated 10-foot translations were run for both units, using a generated function which allowed acceleration to 1 ft/sec and required deceleration to .5 ft/sec at a distance of five feet. No lateral correction was provided for in the Hand-Held Maneuvering Unit. Both units performed satisfactorily over the same range of initial conditions used in the 50-foot runs.

(4) Results of an extensive series of runs and further comparisons between the two units are given elsewhere.[7]

CONCLUSIONS

A computer program utilizing analog and hybrid computer equipment has been devised to simulate and compare unstabilized maneuvering units designed for extravehicular activity. This program can accommodate any maneuvering unit for which the mass and the position of the center of mass, the moments and products of inertia, and the thrust force and moment vectors are available.

By using computer components as comparators and by generating appropriate functions on the computer, a flexible logic has been devised to simulate man's physiological ability and psychological choice in making thrust corrections during a maneuver. The logic limits thus imposed can be varied between computer runs to observe the effect of the limits on maneuvers. A full range of initial conditions, such as a rotation rate or a translational velocity, can be imposed on a maneuver to observe the capability of a particular maneuvering unit to recover and reach the target.

Program output includes time plots of the translational and rotational parameters of motion, thrust on-off profiles and fuel consumption. The number of thrust corrections made and the general character of a maneuver can be determined from these results. Over 350 simulations of maneuvers with the Astronaut Maneuvering Unit and the Hand-Held Maneuvering Unit were performed, and some points of interest have been investigated and comparisons made between the two units. Although the evidence cannot be considered entirely conclusive because of the many parameters which have been varied in this study, the Astronaut Maneuvering Unit demonstrates some advantage over the Hand-Held Maneuvering Unit in a 50-foot translation. The Astronaut Maneuvering Unit usually reaches the target within programmed limits while the Hand-Held Maneuvering Unit frequently exceeds the programmed velocity and displacement. For 10-foot translations, the Hand-Held Maneuvering Unit and the Astronaut Maneuvering

Unit are equally successful in reaching the target. To this comment, it must be added that the Hand-Held Maneuvering Unit simulation is handicapped by the lack of satisfactory correction logic for X_2-axis translation. If it can be further demonstrated in space or by simulation with this program or in a manned simulator that the Hand-Held Maneuvering Unit is equally successful at longer ranges than 10 feet, such factors as cost, weight and simplicity demand its consideration for maneuvering in the vicinity of a spacecraft.

In the development and use of this computer program, it was found that programmed velocity and rotation rate limits must be compatible with displacement and rotation limits. Large displacement limits require, and can tolerate, large velocity limits so as to keep unwanted motion within the desired range without an excessive number of corrections and without precluding other necessary corrections. If an angle is assigned a large limit without adequate correction rate, other corrections are precluded or delayed or result in large unwanted translations and rotations. This is also true of all other limits on displacements and velocities. For excessive rotation limits, such as ±35°, the unstabilized units become generally uncontrollable. For extremely small limits, such as ±5°, an excessive number of corrections are required. Through a more intensive study of the interrelation of rotation limits and rotation rate limits, optimum or suggested values could be found to aid in the selection of units for particular missions and to assist the astronaut in his training for these missions.

The results of several computer runs show the importance of having the line of action of the resultant thrust pass through the system center of mass. This should be emphasized both in the design of maneuvering units and in the training of astronauts. Control improved markedly after the Hand-Held Maneuvering Unit was lowered two inches closer to the astronaut's center of mass. Asymmetrical thrust has also been considered in a series of Astronaut Maneuvering Unit computer runs. When oppositely paired Astronaut Maneuvering Unit thrusters are set at 2.1 and 2.5 pounds of thrust respectively, maneuver control is seriously degraded. For a stabilized unit, this situation results in greater fuel consumption; an unstabilized unit becomes extremely difficult to control.

It is evident that the complexities of simulating a maneuvering astronaut in zero-gravity conditions are great. However, the type of flexible computer program developed in this study can be used to investigate many specific points of interest concerning unstabilized maneuvering units.

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13. ABSTRACT

This report contains a presentation of technical contributions summarizing the status of current and significant research in the fields of Space Maintenance and Extravehicular Activities. The report is based upon the discussions at the Second National Conference on Space Maintenance and Extravehicular Activities held August 6-8, 1968 at the Stardust Hotel, Las Vegas, Nevada. The conference transactions have been arranged in the order of presentation during the seven sessions of the conference.

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